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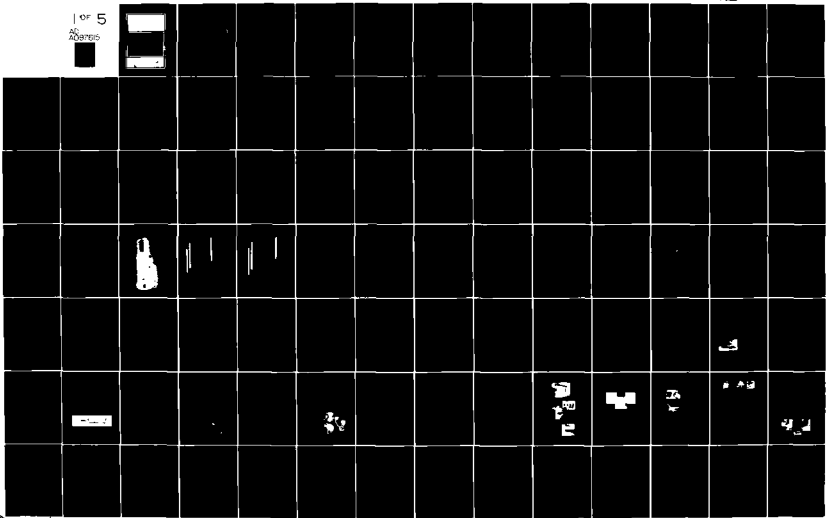

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TECHNICAL EVALUATION REPORT

by
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This symposium, held in Turino, Italy, was the 56th of the Propulsion and Energetics Panel of AGARD. The symposium was important, since the increasing demands on turbine engines performance, durability, safety and pullution emission require an improvement in test methods. The aim of the symposium was to provide better test methods to the engine research and development engineers and to meet the manufacturers', the buyers', and the users' test requirements for engine delivery, reliability, economy and maintenance.

At first a comprehensive survey on testing requirements for engine qualification and development was provided for both military and civil engines. Engine component testing and complete power plant testing were discussed. Various aspects were accounted for; performance assessment with transient and angle of attack effects; complementation of advanced analytical prediction techniques by testing; and comparison of test data and actual flight data.

Finally, trends for future engine testing were considered. New test techniques and additional test facility capability may be required to cope with the changing test needs.

In order to build a framework against which to examine more detailed requirements, the symposium opened with a review of differences between approaches to existing codes, emphasizing differences in reliability and durability testing requirements. The influence of mission profiles and environmental condition impacts were discussed. The possibilities of international coordination to the point of obtaining a common requirement were discussed. The amplification of the details and requirement expansions, and the need for these expansions as they applied to current military engines being developed by the United States Navy opened the view of testing requirements further. Also broadening the view was the system of qualification and certification used in the Federal Republic of Germany. A difference is created by the fact that the approving agency is concerned only with airworthiness, free from demands of schedule, finances and other user nuances, and is true in nearly all commercial aviation. Then some of the peculiarities created by usage were covered, such as helicopters, efforts combined with two countries participating in the certification on a commercial effort.

An additional problem is created by the fact that some sections of the propulsion system, (inlet, air exhaust, their interactions and driven systems) are integrally connected to the airframe, and testing of the total propulsion and power system must therefore be a combined and highly cooperative activity. These problems were pointed out by two air frame manufacturers. Because of these technical aspects and the fact that both air frame and engine manufacturers have great assets at risk, the documentation of the description, schedule, performance, etc., must be carefully defined, as well as the basis for any measurement defined in great detail.

One of the really new testing elements that must be addressed has, in effect, been adapted to military usage in theory and some field trials held. The basis of this new technique is having knowledge of failure rates based on the failure mode which may be used to extract the life built into the engine. There are two predominant failure modes, i.e. stress-rupture and low cycle fatigue. Another mode, high cycle fatigue, seldom reaches service use, since it can exist only minutes; or, abuse being random, can hardly be designed out of the engine. The defined process can aid supply agencies in buying the correct distribution of parts, even when circumstances force a major change in mission or usage. If durability is to be programmed, the testing program must provide proper and adequate testing in the development, in terms of time, hardware, money and facilities. What is really being called for is treatment of the aircraft as an "engineering" system, rather than a "management" system. When adequate data is accumulated this system makes possible a concept on testing objectives which may well be worth an AGARD workshop in 12 to 18 months. The testing to be done includes complete engine testing, environmental testing, component testing, and bit and piece testing (including maintenance effects).

The UK programme of engine in-flight data recording and analysis in service, which is gaining as an essential requirement (for the support of an on condition maintenance policy), is the initial model of data necessary for the system outlined in the foregoing. The minimal system, which began operation in 1974 accruing data for engine life usage information, has been expanded and results put to use.

The remainder of the tests have been found to be necessary and in general incorporated in most programs currently active. These include testing to determine airflow patterns as created by the inlet during steady state and during maneuvers - a costly process because of the data analysis effort required. Also required is an expensive instrumentation arrangement, thus comprising a program which must be well planned or it is useless, or much more expensive. Another element difficult to measure accurately - that is, measure with sufficient accuracy to be useful - is Radar Cross Section, which frequently requires large sections of the aircraft to be attached.

Infra-red radiation is just as difficult, but knowledge of survivability characteristics require both measurements to be made. Emissions are another headache. these include smoke in the military cases, and chemical emissions in the civil sector. Many of the measurements are in parts-per-million, and approach the capability of accurate measurement.

The Auxiliary Power Unit (APU) offers another differing demand for testing, since the life of the unit between overhauls is nearly always determined by low cycle fatigue. It is frequently operated unattended and its life in general is depleted by the start-stop cycle. It is frequently placed where it is difficult to inspect, yet it is complex and must be dependable. The certification test on the APU probably has the widest variation (dependent on use) of any set of engine certification tests.

Another problem which requires testing in the development program is pointed out by our commercial friends. Since the engines for a fleet of aircraft lose efficiency with usage, the loss or lapse rate needs to be known. One airline's data indicates loss of 3/4% per year for the older engines, while the newer, high pressure, low fuel consumption engines lose about 1¼% per year. The differences are not that striking if absolute increases are used. The performance deterioration has been studied in depth, as reported by NASA for both CF6 and JT9.. The damage mechanisms and how they contribute to the performance deterioration include seal rubs, erosion, surface roughness and thermal distortion. If these are the causes of the deterioration, then it is dubious that the current AIDS system can do much to pick out offending modules, and in fact, an AIDS system accurate and adequate to pick up such elements causing deterioration could well cause more maintenance problems than they cure, even if the user were willing to pay for the more complicated equipment.

Another testing requirement which has been added in recent years requires demonstration of a minimum degree of tolerance to foreign object damage. Past field experience indicates that the effect of various hardness objects is great, yet the rules currently are concerned only with ice, rubber and birds, and therefore this is essentially a limited test.

Recently a new form of testing has been demanded by some military purchasers of aircraft engines. This general form of testing has been in use for some time in the civil world. A current name for this testing is Accelerated Mission Endurance Testing, and it sets up cycles to simulate the mission in use, then removes the long time, steady state, easy (low life extraction) portions to give a situation that by adjustment can be made to duplicate some service failures. In general, one specific test arrangement will test one group of parts realistically, but not all parts. It is therefore necessary that there be a way in which to evaluate the value of the test. This can be done with field experience. Evaluating the value of a test is a common problem in engineering, and a function often overlooked. The presentation gave a discussion of the various types of failure modes which are currently not detectable in AMT testing.

The importance of testing the entire propulsion system, which includes engine inlet, inlet control, exhaust, nacelle de-icing system, and other power systems, has long been contemplated and discussed and studied, and is the main reason for free-jet testing. It requires about two times the air flow that a direct-connect arrangement requires. While direct-connect requires between 15 and 20 corrections, free-jet is more complex yet, particularly in the sub-sonic case. The results can be worth the effort, as shown in the NCTE paper.

The technique of inlet compatability testing has been developing over a 15-year period and has been brought to a relatively high state of effectiveness. The method described has proven highly successful in assessing compatability.

Testing engine transients has become of increasing importance regarding durability effects and prediction of stall, surge and other specific problems. The development of the necessary high speed instrumentation and techniques for using it will vary somewhat with the engine being tested and the purpose of the test, but the principles remain constant.

Experimental Verification of Turbobladlng Aeromechanics is a new class of testing required by the increasing diversity of modern aircraft and missions, causing increased blading complexity and requiring consideration of extended and variable operational environment of the current and growth flight regime. Aeromechanical behavior and evaluation must include practical operational effects and sensitivities. Prediction of vibratory responses of forced vibration and fundamental mode instability margins is presently inadequate. The theory has been proven incomplete, and there is certainly opportunity for development of a fundamental theory which responds to the necessary parameters. These dynamic responses are influenced by pressures and temperatures which define the environment of the part. The instrumentation becomes very difficult and complex, particularly for the inner spool of a multi-spool engine when operating in the engine.

The thrust load on bearings is an example of other forces which need to be verified and are very difficult to measure; yet remarkable capability in terms of electronics exists to accomplish these measurements.

Special purpose or broadly differing usage creates many specific testing problems The problems arise due to additional data being necessary, additional components being introduced, or the design life of the engine being so very short for single purpose that frequently an engine will not last long enough to obtain the complete data set. In other cases the importance of data elements is altered by a gross change in usage.

These variations were discussed in terms of a VTOL engine which is required to vector its thrust, change thrust rapidly, and accept large bleed flow some of the time an advanced reheated turbofan engine where the performance was to be measured of determined in flight; and various validation methods used for determining aircraft performance change caused by the variation in engines caused by production tolerances. Also covered were the procedures and specific tests for helicopter transmissions and the specific problem in dynamic testing of interface problems, as well as development testing of a limited life engine, such as the microturbo TRI60 turbojet.

A relatively small role in the program covered component testing. This type of testing is in general necessary to flow match the components, match of RPM for spools of a multi-spool engine, optimization by adjustment of stagger and camber the flow and efficiency of turbine and compressor, optimization of clearances, roughness, fits and a myriad detail of design and construction definition. This class of testing included large compressor testing, concentrating on the facility and instrumentation; full annular combustor test facility, again concentrating on the facility and instrumentation. Also includes procedures, low pressure turbine testing and the accuracy attainable. Also addressed in the paper was supporting bidimensional cascades, rotating cascades, cold flow rigs and the total engine; a two-stage turbine rig used to investigate the effect of the tip clearance on efficiency, Reynolds number effect and cooling flow aerodynamic effects; mechanical testing, such as overspeed, fatigue, bird ingestion and blade containment; and the need for controlling environment during this testing, instrumentation and additional facility needs.

The final touch was two papers predicting the type of facilities needed, but not now available. I will leave it to the historians to show the correctness of these predictions.

The symposium created an excellent examination of the various types of testing which have been found necessary in the propulsion area. All of this testing can in general be divided into four basic categories, which I have elected to call Proof Testing, Capability Testing, Design Testing, and Trouble Shooting. There were two other categories: one endeavoring to correct historic details of test programs to include testing and acquisition of data that will provide aid and comfort to the functions of maintenance, supply and cost-consciousness, an omission long needing correction; the second predicting facilities that will be needed in the future.

It was noted that no paper (and very little mention) appeared on data accuracy or confidence, despite the fact that the primary purpose of testing is to obtain data which can be confidently accepted and used. An uncertainty report on the data is considered as important in testing as the data compilation. Considering performance the most accurate data over the long run is about $\pm 1\frac{1}{2}\%$, which includes force balance and momentum balance methods at the same time. The worst (one-time) set of data seen by the author is $\pm 17.0\%$. It is apparent that much skill exists, but if not properly applied the test has zero value. Life testing has much scatter (largely due to variation of material properties) and can only be as accurate as prediction of usage.

Proof testing concerned about 50% of the papers, and since this is the payoff test, there is great temptation by the manufacturer to aim at this test, fix bayonets and charge without adequate testing in the other categories of testing. Of the 18 papers, 15 were on engine tests, 8 on the various procedures used, and 7 on qualification of special cases. Three were on engine tests as part of their aircraft, one of which was a special case. Many of the specifications call for a large number of tests in addition to a complex and long engine endurance test, which includes many throttle movements, altitude testing, several fuels, etc. Tests such as removing and replacing the oil tank cap 12,000 times, a proof test that the control system will function properly in a very severe electro-magnetic field, a test of the fuel pump with very dirty fuel, an oil interruption test, etc.,are also required. There are between 75 and 100 such tests, and only when all have been successfully completed can the engine be properly called Certified.

Capability testing is usually conducted on an engine, but occasionally on a component. If a component is used, it is usually for one of two reasons; (1) a failure is likely to occur and it becomes far less expensive with a component because secondary damage is restrained; or (2) many of the instrumentation problems with an engine can be eliminated. Seventeen precent of the papers were on such testing.

The newest test form for the military is "accelerated mission testing" and is quite effective for life determination after the usage definition is known. The limitations must be understood for it is not a cure-all. Such things as the effect of errors in adjusting the engine, lapse rate of performance with usage, maneuver limitations as caused by inlet distortion, rate of precession that the engine can absorb and the airframe can provide, limits of afterburner light in altitude and speed, relight limits, surge and stall limitations, low cycle fatigue life component by component, flutter boundaries, stress rupture life and repair tolerances, as well

as techniques, are some of the elements that capability testing will be required to cover.

Design testing usually uses major components or small bench set-ups, and these results are necessary to provide validation of computed designs; to provide corrections to computed designs; and to optimize such elements as cooling passages, camber, turning angle, deviation angles, boundary layer correction factors, temperature profiles of the gas (working fluid) and hot parts, transient heat flow, material property variations caused by manufacturing processes, and heat treatment. They are currently used to optimize variable geometry schedules, adjust designs for 3-dimensional flow effects and determine minimum clearances due to transient temperature and stress conditions. The usual approach is to design the parts based on experience and theory and check the elements out as bench elements or components to the maximum degree possible to ensure profiles, flows, clearances, pressures, temperatures, etc., are as they should be, insofar as possible with the transient conditions. This class was addressed by 19% of the papers covering mechanical parts and elements, turbines, compressors, helicopter trans-missions, combustors, and the facilities and instrumentation necessary to conduct some difficult forms of this class.

The paper which could be considered as "trouble-shooting" covers an important example. In this testing function it is necessary to be able to duplicate the failure not only in failure mode, but in cycles or time that relates back to the point in the engine's life where the failure occurred. These tests lean heavily on experience of the test engineer as well as clever deduction by an analytical design engineer. This paper accounts for about 3% of the testing papers.

One view of the shortcomings of the testing which follows the historical approach may be shown by the following:

1. It provides inadequate data to translate life expectancy at one mission to another mission, an event which will usually occur one or more times in military usage.

2. It provides inadequate data to let the Supply Agency do a good job of purchasing the spectrum of spare parts.

3. It does not provide a logical life growth system in its makeup.

4. It provides a format too rigorous to permit proper aircraft system engineering - as opposed to system management.

5. It does not adequately address the problems of trimming in maintenance.

There were four papers, or 11%, which voiced methods and improvements in the testing approach which would alleviate, if not solve, the foregoing problems. These particular papers - since they seek to solve problems rather than report events, methods procedures in use today, or about to be in use - are deemed important beyond their number. We seem to have forgotten several facts:

1. For every new engine built there are 5 or more "overhauled" engines built, many containing repaired parts, and the bulk of the parts are "used", with all that this word implies.

2. Every Chief Engineer should have to personally go into the field and trim one of his little gems (the most recent) so that he can know without equivocation the kind of mess his trim procedure is. To cite a few examples: some require measurement of a pressure ratio to the third decimal place - a value which is influenced by a breeze of modest magnitude and breeze direction; some engines take as much as 8 hours and several men to do the trim effort; some engines must be retrimmed when ambient temperature changes a modest amount.

3. All too often we find that we have in supply excess spare parts whose failure rate is much less than predicted, and no spare parts for those whose failure rate is much higher than predicted. Naturally, only the latter case creates the havoc (particularly in these days of long lead time) of grounded aircraft.

4. Engines, like people, are not born mature, nor are they likely to be. One of the main reasons is that the aircraft, mission, usage and maintenance environment are not correctly defined when the engine is "born".

5. The required operating characteristics of an aircraft increase in severity during the development process for a number of reasons. Some reasons are: (1) addition of missions, and hence equipment; (2) correction of problems; (3) reduction of cost, such as substituting aluminum or steel for titanium and causing large weight increases; (4) errors in drag estimation; (5) desire for more range, etc. These changes require the engine to alter as dictated by reasonable system engineering. There are cases where the weight of the vehicle has doubled, but system management has not permitted the engine to alter nor the schedule to slow.

These four papers are, in a practical sense, probably the most important of the symposium, if only they stir some thought, but more important, some action -- even a little action.

Conclusions:

I conclude that the symposium did an excellent job in (1) defining procedures, instrumentation, and tests for the currently new requirements; (2) pointing out areas of change needed in the testing program approach; and (3) outlining some areas where some research is needed, such as a basic computational system for blade flutter and instrumentation for emissions, strain and other high frequency devices.

Recommendations:

I recommend that the Panel consider means of causing improvement in content of engine testing programs by (1) providing a logical growth in life element in the sequence; (2) adding requirements to support the logistic functions; and (3) recommending a basic research program on blade flutter.

OVERVIEW OF ALL CIVIL AVIATION ENGINE CERTIFICATION/DEMONSTRATION REQUIREMENTS AND RATIONALE I.E. FAA, CAA ETC.

by

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INTRODUCTION

The Western World scene is presently dominated by two outstanding codes for civil certification of aircraft turbine engines, one adopted in the US, FAR 33, and the other widely adopted in Europe, JAR "E".

The JAR "E" is a fairly recent code adopted by the European Joint Airworthiness Requirements Committee which technically corresponds to British BCAR Sect C but does not include helicopter engines. The present situation is representative of the technical and scientific expertise on both sides of the Atlantic where the most important engine Manufacturers have built their most successful products that became worldwide accepted. Other engine Manufacturers in different countries have followed one of the two codes to certificate their products mainly according to the area of influence of their respective markets. We wish to point out that since many of the successful engines of the western world have been certificated according to both codes, the substantial equivalence of the airworthiness level "de facto" afforded by the two sets of regulations is established.

Since 1970 US (FAA) and British Airworthiness Authorities (CAA) have agreed on the mutual acceptance of engines built in respective countries, provided that a technical evaluation is performed with particular respect to the areas of new features of the design itself. This has been a valuable agreement and has saved time and money for both the Authorities and industries concerned. It is, however, subject to two disadvantages. Firstly, it does not include countries other than US and UK and because of its dependence on ". . . the service record of engines accepted against these codes", it can only really be used by countries with an established and proven record on engines. Secondly, it is theoretically renewable every time a change is made to the requirements on either side, and although this has never led to its cancellation, there have been areas about which the words implying that there ". . . is a general balance of the two codes . . . " have been dangerously stretched.

In more recent years substantial work has been performed by an "Ad hoc" US, UK and France committee trying to establish the existing differences and to reach agreement on common solutions in different problem areas.

Harmonization appears to be a different task owing to different existing philosophies and procedures and different manufacturers positions on both sides of the Atlantic, and the road to reach the ultimate goal appears to be rather long at this point in time.

For many years the FAR 33 format was kept rather short. Basically only the acceptance requirements were written in broad form while policy material to comply with the formal requirements (usually a few sentences if not only one) was laid out in rather extended Advisory Circulars.

BCAR Sect C have included in the body of the regulations also CAA accepted policies spelling out the details of various testing procedures and acceptable means of compliance in great extent.

It was often said that building successful engines is an Art more than a Science and the logical consequence was that to the "Artist" (the Manufacturer) much freedom was left in performing his duty to comply with the rules. The difference between the amount of endurance testing required and the engine overhaul life, which now is about 20 or more than the officially demonstrated period, can be explained only considering economics and technology improvements which have pushed engine durability far beyond expectations.

In the whole framework of the certification tests the mechanical, gasdynamic and thermodynamic properties of the turbine engine receive proper assessment together with all engine related parts and related systems. FAR 33 makes specific reference to those parts of fuel and oil lines that are integral parts of the engines in assessing their endurance, vibration, and fire proof characteristics etc.

BCAR Sect C in addition to system components that are physically attached to the engine makes also reference to other system components that are part of the aircraft system with an overall view to engine aircraft integration and interface. Some requirements that are currently in BCAR Sect C cover also parts of the aircraft systems that in the FARs given in different parts outside FAR 33 (i.e. FAR 25). The same approach is used in respect of accessories that are covered in a more detailed fashion defining the requirements to be included in the specifications such as:

Vibration effect on the operation
Acceleration effect on the operation
Gyroscopic loads as appropriate
High energy rotating parts containment
Resistance to fire or explosion
Magnetic and radio interference
Electrical insulation
Climate variation such as humidity etc.

while FAR 33 refers broadly to their reliability under expected flight and atmospheric conditions.

Owing to the practical difficulties to present a paper dealing with all engine certification requirements, some of the most significant items have been identified and are discussed extensively with the purpose of detailing a comparison between the two sets of rules as follows:

1. Power ratings
2. Endurance testing
3. Stress and fatigue aspects of rotating parts
4. Foreign object ingestion
5. Ice protection
6. Engine fault analysis

Only general comments are then given for other requirements. Some views are then expressed concerning other aspects of engine final certification that although outside the airworthiness certification cannot be ignored since they are dictated by existing environment rules, such as noise and exhaust pollution requirements.

1. POWER AND/OR THRUST LEVELS

The performance data provided by Engine Manufacturers and approved by the Airworthiness Authorities give the power and/or thrust produced by an Engine under specified conditions (i.e. intake efficiency, forward speed, atmospheric temperature) when operating within the limitations approved for the defined conditions of power and/or thrust in terms of RPM, Torque and EGT.

The conditions of power and/or thrust as defined in the two sets of rules under examination differ to some extent and the following differences are pointed out.

While the definitions for rated maximum continuous power and/or thrust do not differ in the two basic codes (for unlimited period of time) the take off power and/or thrust is limited to 5 minutes duration according to FAR 33 and can be used up to 10 minutes in multiengined aeroplanes when one power unit has failed according to BCAR Sect C.

According to both codes when one power unit has failed during take off and balked landing a greater power setting can be used in addition to the T.O. power and/or thrust the maximum contingency power and/or thrust (for a period of 2½ minutes). FAR 33 however allows the usage of the 2½ minutes level for the augmented thrust case.

Only BCAR however introduce the definition of intermediate contingency power and/or thrust that also can be applied for a period of unrestricted duration when one power unit is shut down (normally higher than max continuous power). FAR 33 speaks of 30 minutes power rating that is practically the same level of power as above but limited to a period of 30 minutes (only for helicopter engines). For single engine helicopter BCAR defines in lieu of take off power the one hour power condition.

2. ENDURANCE TEST

Power rating based upon standard atmospheric conditions with no airbleed for aircraft services and with only the accessories installed necessary for engine functioning must be established before and after the endurance test in order to reach assurance that no detrimental change has occurred in respect of engine performance (calibration tests).

When speaking of endurance test one may think of a typical engineering evaluation based upon sound practice that stems from old applied philosophy and that has not changed much through the years. In fact this basic requirement of the certification test has also followed the engine evolution throughout the years. Basically every endurance testing is directed to assess the extent of wear in mechanical parts, the impact of vibrations and effects that may cause distress in the hot section. The duration of the endurance tests is still limited to 150 hours in both codes plus a limited amount of hours to be decided by the Airworthiness Authority.

In the civil field there are examples of endurance testing for a longer period of time (i.e. 200 hours ground testing of transport helicopters) but this "magic" figure of 150 hours that dates back to old piston engine practice has remained the same. As it is generally recognized a factor of at least three can be assumed in assessing the equivalent life of an engine in flight operation that allows to grant the engine an initial overhaul life of about 500 hours. This is considered together with other prescribed specific tests for every aspect of the certification testing to be sufficient as a minimum level of acceptance from an airworthiness standpoint.

If the Manufacturer wishes (universally accepted practice today) he can demonstrate by extensive bench testing and endurance cycles much greater figures for the initial engine overhaul life. Before and after official endurance tests the engine running according to the established or more severe schedule takes place very extensively to meet current civil market requirements.

An important requirement was recently added to FAR 33 concerning the "overhaul test" defined as the test run to simulate the conditions in which the engine is expected to operate in service, including start stop cycles. The period of time of the test is established taking into account limitations on operation prior to the first overhaul.

Following the aeronautical progress, requirements have been continuously improved to keep pace with new developments (e.g. for endurance testing of dual and trispool engines). Special requirements have then been added for supersonic engines to account for air inlet temperature increase during testing, for thrust augmenters operation and variable area exhaust nozzles operation.

If one compares the typical 6 hours block tests (6 x 25 = 150 hours) of BCAR Sect C (JAR "E") and FAR 33 requirements for subsonic aeroplane turbine engines as per Figure 1 both endurance schedules look practically the same.

However the 2 hours and 30 minutes period at incremental power is divided into 15 steps with different criteria with respect to significant peak vibration regimes. According to BCAR not less than 10 hours (but not exceeding 50% of the incremental period) shall be run with RPM varied continuously in the critical vibrational range while FAR 33 quotes only the maximum period to be run under those conditions (also 50%).

On the same Figure 1 the corresponding 6 hours block for gas turbine engines for helicopter use is also plotted to indicate the differences when 2½ power rating or 30 minutes power rating approval is requested. No attempt is made to show differences with corresponding parts of BCAR concerning helicopter turbine engines since the main purpose is to discuss aeroplane engines and many differences exist.

Differences exist also in the testing program on the thrust reverser. An example of the schedules is given in Figure 2 and Figure 3. If the thrust reverser is to be used in flight additional testing is required. Generally speaking the equivalence of the respective testing schedules in both codes is considered.

In Table 1 a comparison is presented between the endurance tests of a turboprop engine plus the propeller according to FAR 33 and BCAR Sect C.

3. STRESS AND FATIGUE ASPECTS OF ROTATING PARTS

3.1 Centrifugal and Thermal Stresses

Besides the requirements concerning the proper design and functioning of the engine control system that must prevent the engine from exceeding the limitations affecting turbine, compressor rotor structural integrity, the requirements concerning actual stress margins of rotors are somewhat differently laid out in FAR 33 and BCAR's Sect C.

According to FAR 33.27(c) only the turbine and compressor rotor sustaining the highest operating stresses at maximum limiting RPM (the "heaviest") must be tested, while according to BCAR each rotor must be tested.

The test schedules of both codes are in Table 2.

For a period of 5 minutes and at maximum operating temperature (except for test No.3) the rotor must be tested according to FAR 33. In addition FAR 33.88 requires a test of 30 minutes duration with gas temperature 75°F higher than the maximum operating limit on the complete engine.

Following each test, each rotor must be within the dimensional limit allowed by the type design and may not be cracked.

BCAR Sect C calls for the most critical temperature conditions that the turbine rotor can attain in the event of failures of the cooling air supply but is less specific in accepting the test article after the endurance test. It appears that the requirements for centrifugal stress are more severe in BCAR while thermal stresses requirements are more stringent in FAR 33. For engines equipped with a free turbine BCAR requires an overtorque test for 15 minutes to substantiate a 20 seconds limit. The overtorque is assumed at the maximum engine declared overtorque or 3% in excess of the maximum torque of the engine whichever is greater. No corresponding requirements in FAR 33 exists.

3.2 High and Low Cycle Fatigue Aspects

A survey of vibratory stresses of rotating parts (including discs and shafts together with blades) represents the corner stone of high cycle fatigue life assessment. The influence of air inlet distorsion or other installation related aspects (e.g. when the use of a propeller is contemplated) is accounted for during the alternating stress survey for the purpose of measuring the peak values in critical parts. FAR 33 requires all stator and rotor blades* to be surveyed but only to 103% of maximum take off RPM and does not include carcass nor fault conditions to be investigated (according to BCAR 105% of Maximum RPM or 2% in excess of maximum overspeed must be investigated but only critical blade rows are selected).

The peak amplitude of the stressess must be shown to be below the endurance limit of the material. If not, for steel part infinite life is considered as attained when at peak amplitude of the stresses, 18 million stress reversals have been sustained without fatigue failure by the test specimen.

It is worth to note that notwithstanding the requirement for safe life demonstration of blading both codes require also blade containment capability from the engine case. This requirement is certainly dictated by other considerations than purely fatigue aspects (such as foreign object induced damage) but certainly represents an additional safety aspect when embodied into the engine in respect of damage deriving from fatigue failures (see Table 3).

3.3 Low Cycle Fatigue (Start Stop Cycle) (see Figure 4)

The average non containment rate from all causes (world wide) is 1 per 10^6 aircraft hours (fairly constant value during the past 10 years). Of all non containments about half involved a disc failure of which about 1/8 have resulted in the release of debris approaching or equal to a third of the disc.

Possible alternate solution to avoid non contained failures would be a reduction of RPM, while a reduction of 5% is meaningful, in respect of centrifugal stresses (10%), its effect on engine thrust is also very important. On the other hand, total containment except for small engine, is not a feasible solution as the weight penalty is too high (for an engine of 20 ton thrust the weight of the shield needed is of the order of 50% of the bare engine weight).

At the present state of the art a partial containment (e.g. 1/20 of bladed disc mass) is a possibility and research is going on including the control of the direction of crack propagation and optimum case design for containment.

Both codes require an operating limitation be established for each rotor disc and each rotor spacer of the compressor and turbine with regard to the number of start stop stress cycles. The start stop cycle is then defined as accelerating the engine after starting to its maximum rated power or thrust and maintaining the power setting until temperatures are stabilized, after which the engine is stopped and disc and spacer temperatures are reduced to a significant amount.

A factor of three is prescribed by FAR 33 between the service life assumed and the number of cycles tested without failures. An increase of the service life so determined can be granted if at least three samples of the discs that have been operated in actual service through the service life are tested for a number of cycles at least twice the requested life increase.

BCAR define the predicted safe life as:

$$P = \frac{KNi}{y}$$

"Ni" is the number of test cycles completed at the test rig, "K" is a test factor of equivalence between test cycles and flight cycles in terms of fatigue damage and "y" is the scatter factor usually taken as 4.

The initial value of the service life should be established on the test available data. Since the program does not cover all possible factors which may affect the lives of service components (corrosion, fretting, high cycle fatigue stresses) on a new engine usually P/3 is retained always less than 5000 cycles.

Higher time in service discs can then be subsequently granted, assisted by further analytical and experimental stress determination on components in order to increase service life.

Subsequent service life increase of approximately one fourth of the predicted maximum life is suggested.

Both criteria adopted for the establishment of disc life during engine certification tests appear substantially conservative. However present criteria certainly need further improvements in respect of areas such as statistical analysis, fracture mechanics and quality control (production aspects).

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* and shafts

4 FOREIGN OBJECT INGESTION

4.1 Bird Ingestion

The problem is very serious as indicated by available service data. From service experience of nearly all current commercial engines, *the ingestion of a single medium size bird will result in damage in approximately 60% of the* incidents, followed by inflight shut down, in approximately 30% of the incidents.

For seven of the major commercial turbine engines the average shut down rate due to bird ingestion is less than one shut down for every 100,000 of aircraft flights. For high by pass ratio turbofan the rate of engine shut down following bird ingestion is approximately 6 times higher than for other seven engines considered, but service experience has shown no significant hazard on those engines.

Chances to have shut down due to medium size birds on two engines are of the order of 1 per million aircraft flights.

The major concern about those data is that the absence of a definite inlet area relationship since the proposed linear relationship between inlet area and the number of medium birds ingested is not supported by available statistical data.

For small aircraft powered by smaller engines it appears that the small bird ingestion is also a problem and might provide a greater hazard than the medium birds.

The following classification of birds and relevant type tests associated with them is taken from BCAR Sect "C" (JAR "E") but is corresponding to definitions of FAR 33:

Large birds (those having 4 lbs (1.8 Kg) of weight)
It is required that ingestion into engine does not give rise to hazardous conditions to the aeroplane as a result of the damage that may be caused. Impact and ingestion are assumed to occur at max true airspeed in service up to 2500 m of altitude.

Medium birds (those having 1½ lbs (.7 Kg) of weight)
Small birds (those having 2 – 4 oz (55 – 110 gr) of weight)
It is required that at the maximum true airspeed during climb immediately after take off no unacceptable immediate or ultimate engine performance loss occurs nor serious increase of engine operating temperature or deterioration of engine handling characteristics and no dangerous physical damage after bird strike (see *Table 2*).

4.2 Hail and Water Ingestion

This occurrence is now being handled in equivalent fashion by both regulations (see Table 3). For water ingestion test, introduced to cover the danger of damage arising from casing distortion or contraction when subjected to sudden chilling effects of entry into heavy rain and to avoid engine flame out, both high power test and low power conditions need to be investigated. BCAR and FAR prescribe both an ingestion test of 4% water concentration of the intake air mass flow at flight idle speed and at take off power for 3 minutes. No unacceptable reduction of engine performance should occur after water ingestion tests nor dangerous mechanical damage or deterioration of engine handling characteristics after hail ingestion tests.

5. ENVIRONMENT EFFECT

5.1 Engine Icing Protection

One of the basic decisions of the JAR Joint Airworthiness Committee was to accept FAR 25 Appendix C as the basic icing atmosphere. The engine test program itself does not actually depend on the atmosphere definition since test conditions are defined elsewhere in the BCAR. However this point was very important in achieving standardization. Tests on both codes are made to establish that the engine will function satisfactorily when operated in the atmospheric icing conditions prescribed in FAR 25 Appendix "C" without unacceptable (see Figures 5/(a)/(b) and 6/(a)/(b)):

Immediate or ultimate reduction of engine performance
Increase of engine operating temperature
Deterioration of engine handling characteristics
Mechanical damage

However guidance contained in FAA Advisory Circular 20 – 73 are still of general nature and tests mentioned cover only testing at sea level under static conditions. The problems of surge and flame out occurring after ice shedding however have caused some concern for JAR in flight under natural icing conditions since sea level testings do not provide in their opinion adequate covering. Tests have to be conducted with representative intake and propeller (for turboprops) since intake distortion due to incidence or ice formation on the intake and propeller must be taken into account together with shedding of ice into the engine or icing of engine sensing device contained in the intake.

In JAR "E" (BCAR Sect "C") the following table is presented to prescribe repetitions of either

28 Km Continuous/5 Km Intermittent Maximum Conditions for 30 minutes conditions

or

6 Km Continuous/5 Km Intermittent Maximum Conditions for 10 minutes conditions.

The test points have been indicated within Figures 5/(a)/(b) and 6/(a)/(b) for the two cases mentioned above and for the 10 minutes tests respectively at T.O., MC, 75% MC, 50% MC and idle according to AC 20-73.

During testing two minutes delay in initiation of operation of ice protection systems are considered representative to demonstrate that engine characteristics are not unacceptably affected.

Each test should be run at minimum power declared by the Manufacturer and at the end of the specified period the engine should be run at maximum power conditions corresponding to test altitude to demonstrate any effects of ice shedding.

5.2 Ground Freezing Fog or Freezing Rain

These conditions are a problem which has been experienced in service and FAA requires a separate test with an atmospheric liquid water content of 2.0 gr/m^3 while JAR E (BCAR Sect C) requires a test of 30 minutes duration at minimum selected RPM with an atmospheric liquid water content of .3 gr/m^3. At the end of the period the engine should be accelerated at Maximum T.O. power without suffering unacceptable damage or power loss.

5.3 Ice Crystal Conditions

Sect. E of JAR (BCAR Sect C) requires additional testing for engines designed with reverse flow intakes, or having intakes involving considerable changes of air flow direction. This schedule is not reported here for sake of brevity.

6. FAILURE ANALYSIS

The failure analysis is becoming more and more a useful tool in designing aircraft equipment and in recent years also engine requirements have been updated to cater for this basic issue that certainly had a great role in assessing safety for supersonic aircraft and spacecraft. Indeed on the aircraft side the method has been applied to systems for a great number of years.

In FAR Part 33 a requirement (para 33.75) is laid out concerning safety analysis and calls for analysis of any probable malfunction of single or multiple failures that could cause the engine to:

catch fire
burst (penetrate in its case)
generate loads greater than those specified for the engine
lose the capability of being shut down.

The intent of this requirement is to give in engine terms the interpretation of what may constitute "hazard to the aeroplane" and this often cannot be judged only by the engine Manufacturer. It has been generally accepted that aircraft design should be aimed at catastrophic failure rate for all airworthiness causes not exceeding an extremely remote probability.

An event having an extremely remote probability to occur during the life of single aircraft becomes a remote probability if one considers the number of hours flown during the lifetime of all transport aircraft having the same type of engine. This obliges the aircraft Manufacturer to introduce costly modification and heavy reinforcement to the aircraft structure to withstand disc fragment penetration without catastrophic consequences or engine fire. The present definition of remote event is defined as an event having about 10^{-6} probability of occurrence while a very remote event is associated to a probability in the 10^{-8} – 10^{-9} range. Predictions based on the present state of the art often fell short of design goal due to shortcomings of the present testing methodologies. It is recognized that the probability of prime failures of certain mechanical rotating parts (e.g. rotor disc, shaft) cannot be estimated in numerical terms.

The difficulty of the problem to reproduce during fatigue testing of engine discs the loading sequence that occurs in real operation is mainly due to the presence of the blade induced stresses together with centrifugal and thermal effects.

The final acceptance of the level of safety of the design shall be based on "engineering judgement and previous experience combined with sound design and test philosophies" according to BCAR Sect C which also admits an absolute proof is not possible in those cases.

Having mentioned the first difficulty encountered in failure analysis we shall recall Table 4 which summarizes the current definitions of the events as per BCAR Sect "C" in accordance to the principle "the more dangerous the less probable".

It is very important to mention that the failure analysis includes investigation of all manual and automatic engine controls, cooling systems, gas temperature control, engine governor, overspeed limiters, thrust reverses (or propeller) system etc. When the expected failures can be of dormient type important consequences are drawn with respect to the maintenance controls and related check periods. In many cases failure analysis will depend on installation conditions and it is very difficult to define when the Engine Manufacturer responsibility ends in certain cases.

The failure analysis when significant doubts exist on the effect of failures and likely combination of failures (including induced failures) has to be substantiated by tests when a hazardous consequence is suspected (e.g. effect of unbalance arising from a large blade failure or bearing failure).

The "fall out" of the failure analysis are also very important and invest together with areas strictly related to engine design and quality control of manufactured parts and equipments also different areas such as engine operating instructions (crew emergency procedures) and as indicated before maintenance procedures and checks intervals.

7. FINAL REMARKS

After reviewing the differences between the two sets of rules in some important areas we wish to point out that many of them can be attributed more to specific choice dictated by arbitrary will than to technical reasons.

It is difficult to establish where in most cases the best solution is. In fact the willingness to work toward a common set of rules could simplify the task of achieving standardization in the future. Some other aspects of final engine approval are also existing which presently are handled as separate issues from basic airworthiness certification. Their importance is a relatively recent addition to the general framework of engine certification due to environment requirements such as noise and exhaust emissions.

As an example we will recall the relevant rules to be followed in US for emission control. FAA has issued the Special Federal Aviation Regulation (SFAR) 27 to ensure compliance with aircraft and aircraft engine emissions standards and related test procedures issued by the Environment Protection Agency (EPA) under Part 87. EPA Part 87 covers together with engines of new design still to be certificated also engines in operation (such as Pratt & Whitney JT3D and JT8D families). EPA rules define the "SMOKE NUMBER" together with Hydrocarbons, Carbon Monoxide and Nitrogen Oxide percentage in weight not to be exceeded for engine exhaust. The main impact of those requirements affect the Combustor design of the engine.

As far as noise control is concerned the situation is less clear for gas turbine engines since existing noise rules (ICAO Annex 16 and FAR 36 in the US) apply only to aircraft in flight defining the noise levels not to be exceeded at the three measuring points as a function of the maximum weight of the aircraft. However the most important noise source of aircraft noise can still be identified in engine noise. The noise standards have had a strong impact in the latest generation of engines and will continue to greatly influence the overall design of future engines. Drastic reductions of engine noise have been achieved through new design of compressors, fans, nozzles and engine sound proofing.

In order to reach the best design compromise the Manufacturer performs noise surveys on a test stand with the complete engine but no official confirmation of final results is given.

If one looks at the airworthiness implications of all environmental aspects the basic issue which appears evident is that every structural modification requirement to basic engine components dictated by environment rules should be embodied in the engine configuration that will undergo airworthiness certification testing. If not, costly repetition of certification tests will take place and will be unavoidable. Obviously the test article should be fully representative of final engine configuration (Type Design).

The test schedule should include the overall and comprehensive overview of all tests needed before final engine approval (not only airworthiness certification). The proper succession of relevant official tests should therefore be planned with the aim to establish the link between environmental requirements and airworthiness testing.

BIBLIOGRAPHY

1. Federal Aviation Regulation Part 33C up to and including Amendment 33-8.

2. British Civil Airworthiness Requirements Section C, *Engines and Propellers*, (Issue 11). Joint Airworthiness Requirements Sect E, *Engines*, (including Amendment 4).

3. Federal Aviation Administration
   Advisory Circulars
   - 33-1 Turbine engines foreign object ingestion and rotor blade containment type certification procedures
   - 33-2 Engine Type Certification Manual
   - 33-3 Turbine and compressor rotors type certification substantiation procedures.

TABLE 1

Turboprops Endurance Test
(Engine Propeller Combination)

FAA Schedule	*CAA Schedule*
150^h Endurance test on engine fitted with propeller (variable pitch)	150^h Endurance test on engine fitted with propeller (variable pitch)
Feathering 85 cycles	**Feathering** 50 cyclees
Reverse Pitch 175 cycles of 30″ each from idle 25 cycles of 30″ each from MC power	**Reverse Pitch** 200 cycles of reverse from fine pitch then returning to coarse pitch of 1′ each
Negative Torque and Thrust 25 cycles from MC power	**Overspeed** 10′ at max RPM + 5%

TABLE 2

FAR 33		*BCAR Sect C*	
120%	of maximum limiting RPM (rotors equipped with blades or weights)	125%	of maximum speed (RPM) to be approved
115%	of maximum limiting RPM if on engine	110%	of highest speed (RPM) resulting from engine component failure or system
120%	of RPM (while cold spinning) at which max temperature and RPM induced the same stresses	105%	of highest speed (RPM) resulting from engine component failure + other failure (that cannot be detected by crew)

TABLE 3

Foreign Object	*Quantity*	*Speed*	*Engine Operation*
Small birds 3 oz	One per 0.0032 m^2 of inlet area and fraction thereof up to max 16	Lift off speed	Take off
Medium birds 1.5 lb	One per 0.2 m^2 of inlet area, additional one per each 0.65 m^2 up to max 8	Initial climb speed	Take off
Big birds 4 lb	1	Maximum climb speed	Maximum cruise
Ice	Qty accumulated after 30 seconds delay	Sucked in	Maximum cruise
Hail	If inlet area < 100 in^2 One hailstone of 1 in If inlet area > 100 in^2 One hailstone of 1 in and one 2 in hailstone for each additional 150 in^2	Rough air speed	Maximum cruise at 15000 ft
Water	8% weight of engine airflow	Sucked in	Take off and flight idle
Broken heaviest compressor or turbine blade (80% of the blade)	One	Sucked in (fan can be tested separately)	at take off 15 seconds delay before shut down
Mixed gravel	1 oz per each 100 in^2 of inlet area	Sucked in	Take off (over 15 minutes period)

TABLE 4

(From BCAR Sect "C")

Type of Engine Failures	*Type of Event*	*Probability of Occurrence*	*Estimated Figure*	*Failure Analysis*	*Engine + Systems*
"Safe" engine loss of power	Minor effect	Reasonable Probable	10^{-3} – 10^{-4}/hour	NO	
	Major effect	Remote	10^{-5} – 10^{-7}/hour	YES	
(a) Disc failure	Hazardous	Extremely remote	10^{-8}/hour	YES	
(b) Thrust in opposite direction					
(c) Inability to shut down					

TABLE 5

Other Systems Requirements Comparison

Subject	*FAR 33 More Severe*	*BCAR Sect C More Severe*	*Comment*
Duplicate Ignition	X		BCAR Permit single ignition
Contaminated Oil	X		FAR requires testing of engines with contaminated oil
Titanium Fires		X	BCAR to exclude titanium stators unless compensation factors are provided
Purity of Bleed Air		X	BCAR consider it part of engine certification
Rubbing of Rotating Parts		X	BCAR requires specifically an investigation of position of rubs
Equipment Drives		X	BCAR covers failure cases
Tear Down Inspection	X		FAR requires parts to conform to type design and manual limits
Shaft Failures (Gear Boxes)		X	BCAR more precise in requiring demonstration of safety in failure case or defined integrity.

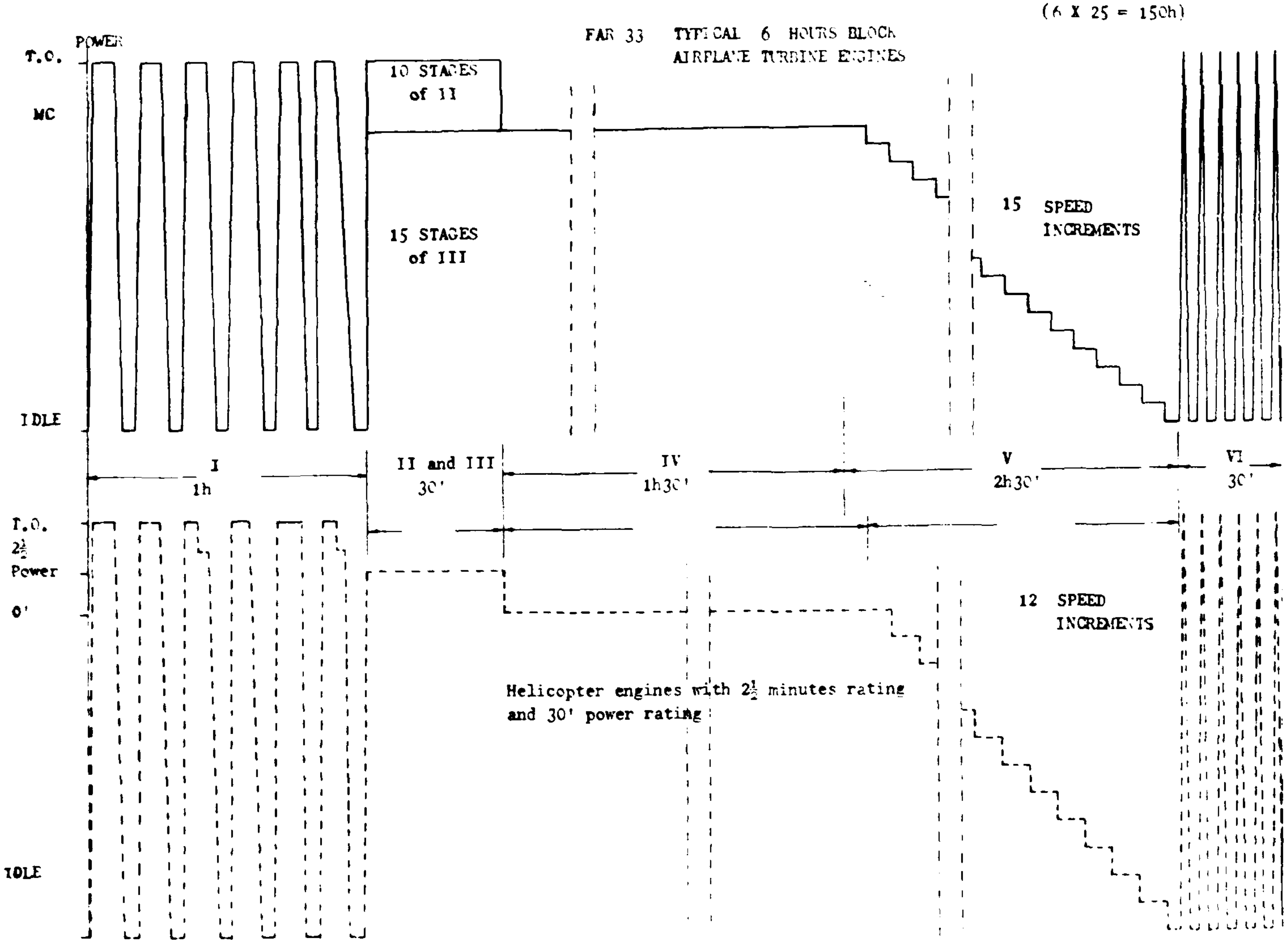

FAR 33 TYPICAL 6 HOURS BLOCK
AIRPLANE TURBINE ENGINES
(6 X 25 = 150h)
POWER
T.O.
MC
IDLE
10 STAGES
of II
15 STAGES
of III
15 SPEED
INCREMENTS
I
1h
II and III
30'
IV
1h30'
V
2h30'
VI
30'
T.O.
2½
Power
0'
IDLE
12 SPEED
INCREMENTS
Helicopter engines with 2½ minutes rating
and 30' power rating

Figure 1

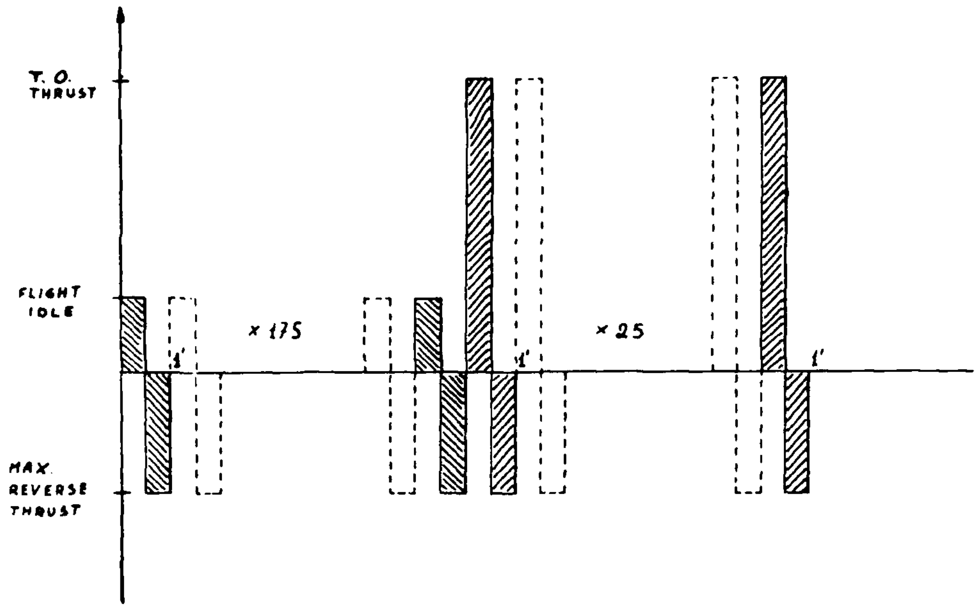
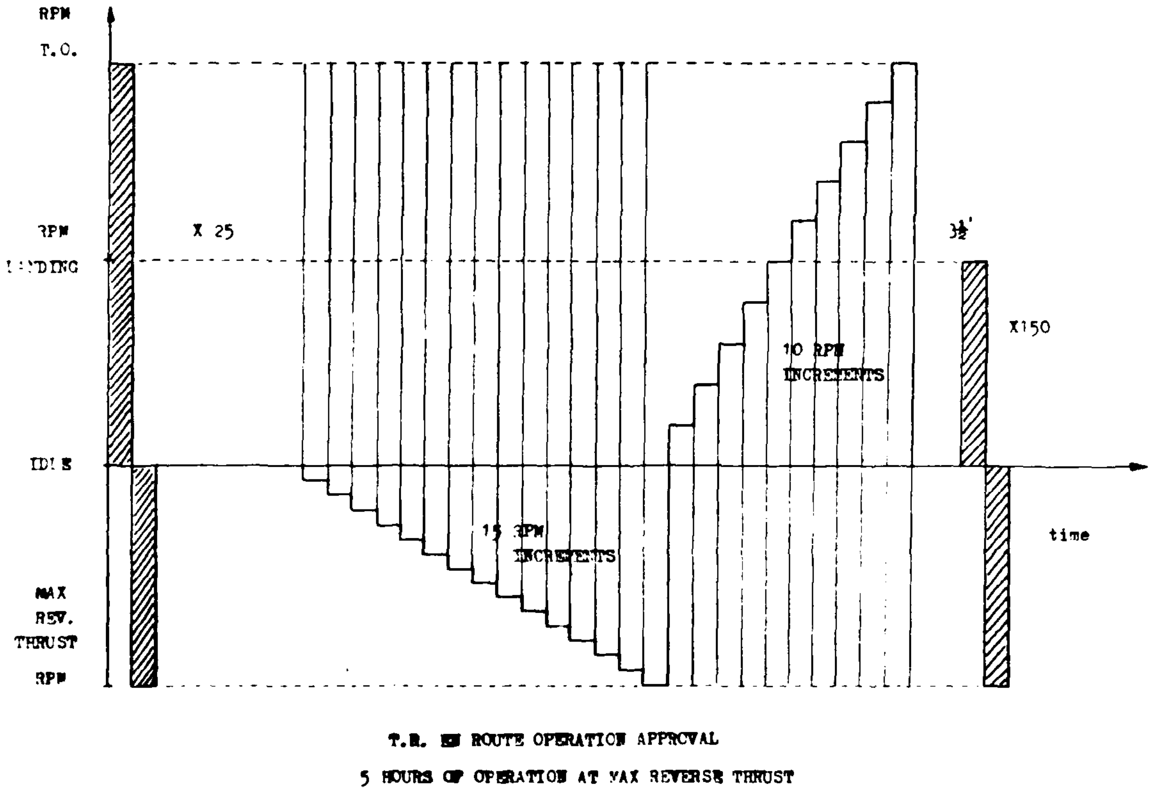


Fig.2 FAA thrust reverser endurance test



T.R. EN ROUTE OPERATION APPROVAL

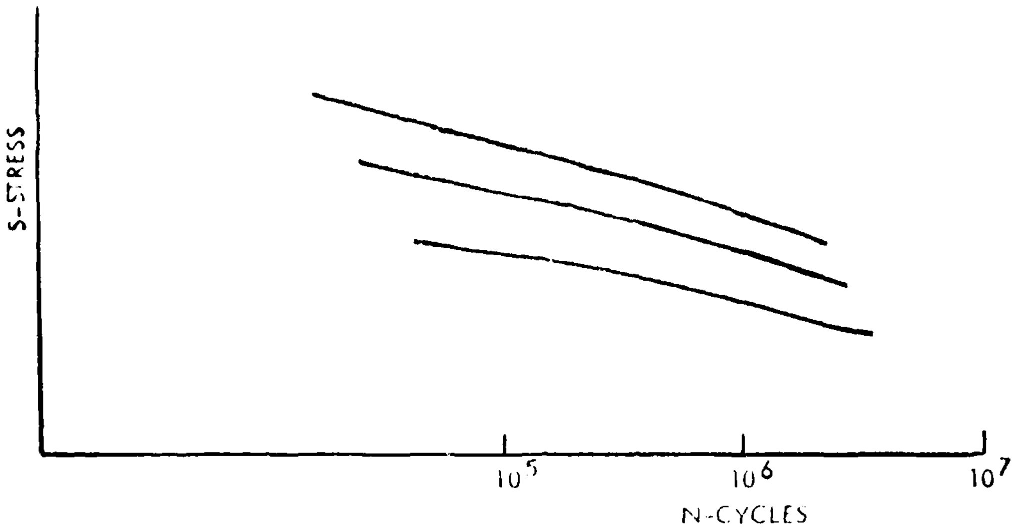
5 HOURS OF OPERATION AT MAX REVERSE THRUST

(50% OF LANDING REVERSE THRUST)

+ 30 OPERATIONS

Fig.3 CAA thrust reverser endurance test (landing)

S-N CURVE



LOW CYCLE FATIGUE
S-N CURVE

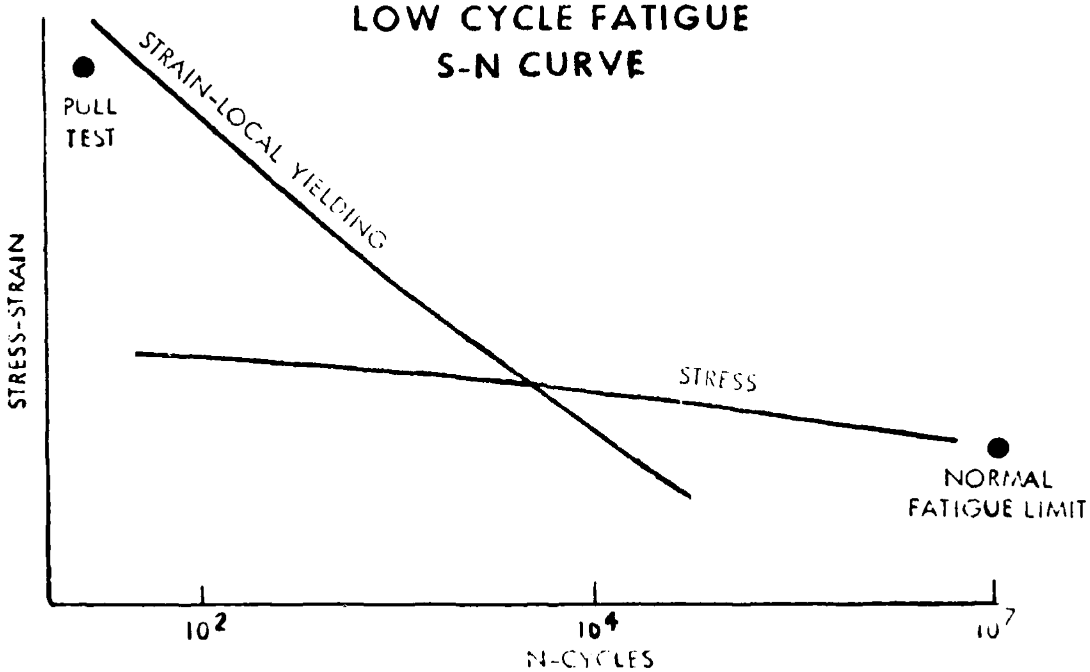
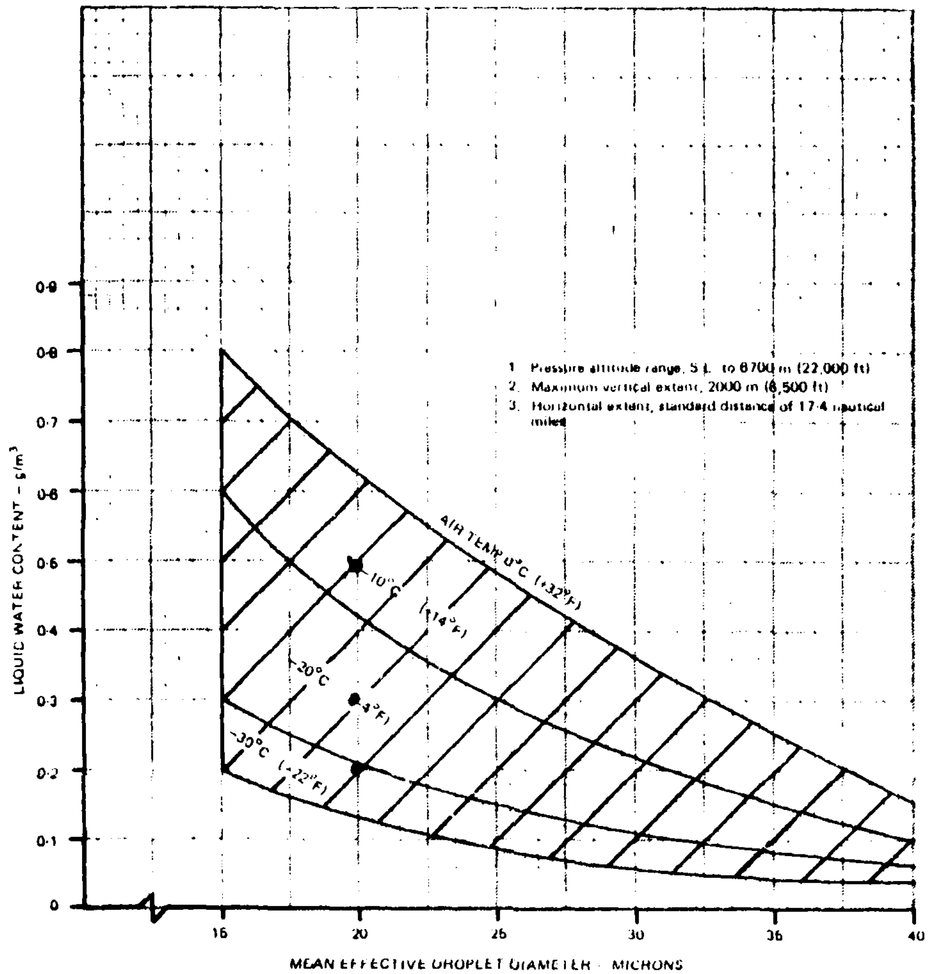


Figure 4

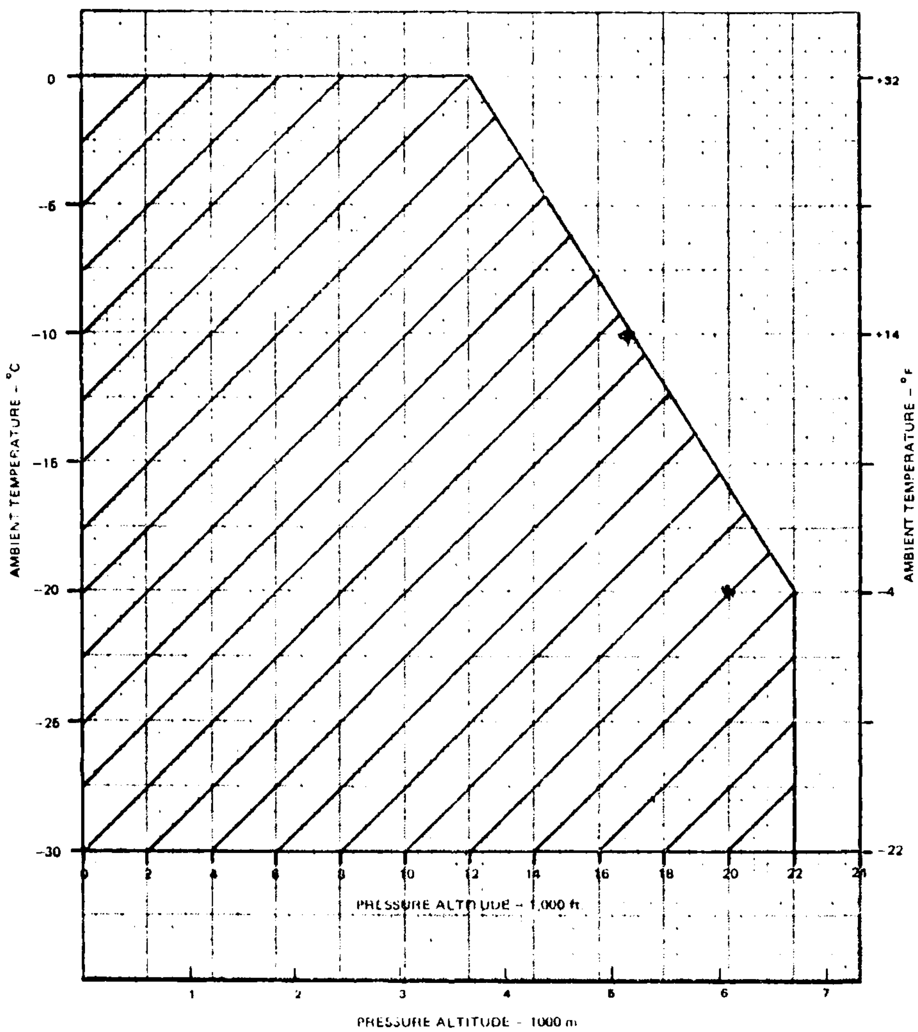


● TEST POINT

CONTINUOUS MAXIMUM (STRATIFORM CLOUDS)
ATMOSPHERIC ICING CONDITIONS
LIQUID WATER CONTENT VS MEAN EFFECTIVE DROP DIAMETER

NOTE: Source of data – NACA TN No. 1855, Class III-M, Continuous Maximum.

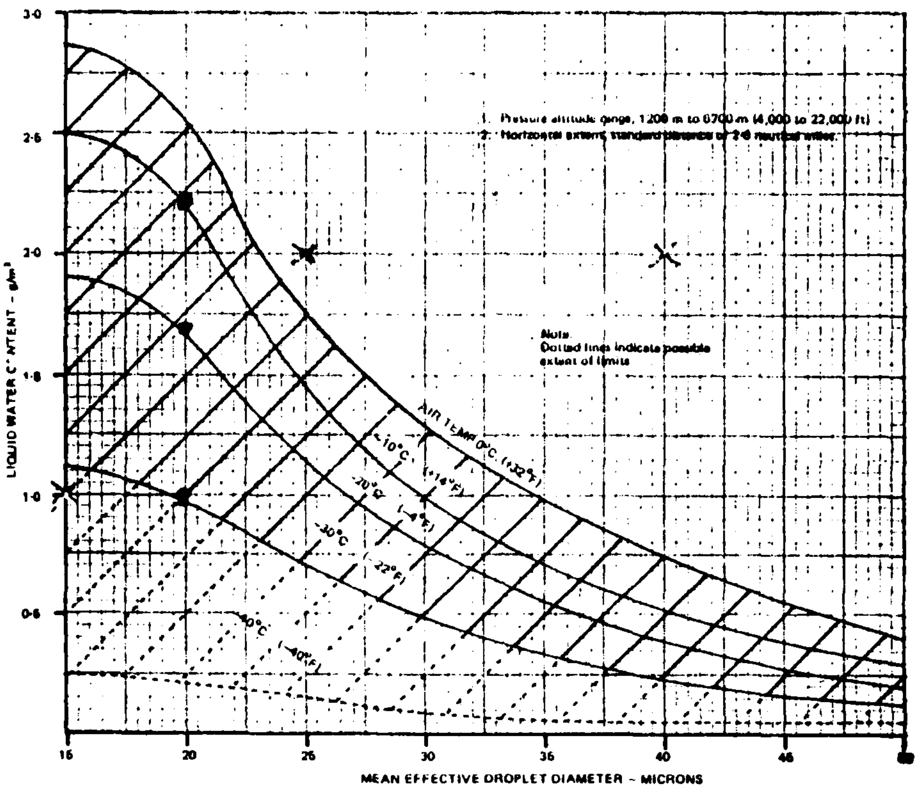
Figure 5(a)



CONTINUOUS MAXIMUM (STRATIFORM CLOUDS)
ATMOSPHERIC ICING CONDITIONS
AMBIENT TEMPERATURE VS PRESSURE ALTITUDE

NOTE: Source of data NACA TN No. 2569.

Figure 5(b)

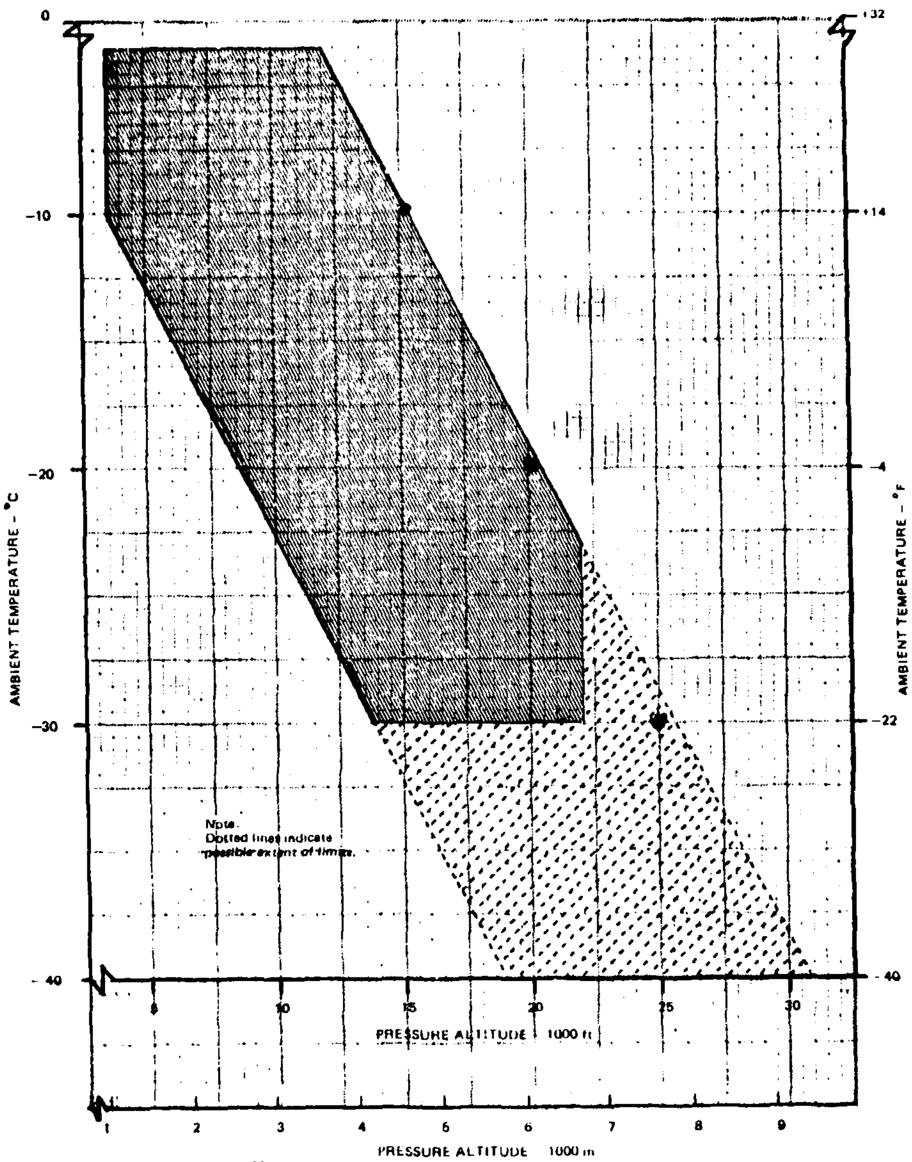


● TEST POINT (JAR E)
x TEST POINT (AL 20-73)

INTERMITTENT MAXIMUM (CUMULIFORM CLOUDS)
ATMOSPHERIC ICING CONDITIONS
LIQUID WATER CONTENT VS MEAN EFFECTIVE DROP DIAMETER

NOTE: Source of data—NACA TN No. 1855, Class II-M, Intermittent Maximum.

Figure 6(a)



● TEST POINT

INTERMITTENT MAXIMUM (CUMULIFORM CLOUDS)
ATMOSPHERIC ICING CONDITIONS
AMBIENT TEMPERATURE VS PRESSURE ALTITUDE

NOTE: Source of data NACA TN No. 2569.

Figure 6(b)

DISCUSSION

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Could the lecturer please amplify on the difference between transient thrust requirements for British and American certification? It should be noted that small differences in the definition of initial and final thrust settings have a major effect on the acceleration time; this is fundamentally due to the large increase in rotor time constant at lower rotational speeds. The initial setting, therefore, has a considerable effect on the acceleration fuel schedule, which, in turn, affects the temperature levels and cyclic thermal damage. The response requirements for civil and military aircraft, and for helicopters and fixed wing aircraft are quite different and these differences must be considered in specifying response rates.

Author's Reply

By looking to the two sets of rules considered (within the Amendment status of the two sets of rules (as per Bibliography), five (5) seconds time delay is a common requirement for the acceleration rate, but I must confess that this important item was not included in the list of items considered in the comparison I made.

SPECIFICATION REQUIREMENTS FOR U.S. FIGHTER ENGINES

by

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SUMMARY

Military Specification MIL-E-5007D, a general specification for the development of turbofan/turbojet engines, has been used by the U.S. Government for the procurement of new engines since 1973. This specification was tailored and applied by the U.S. Navy to the F404-GE-400 engine (F-18 aircraft) development program.

This paper briefly discusses the military general specification philosophy for the procurement of turbojet/turbofan engines using MIL-E-5007D. With heavy emphasis on technical requirements and assurance tests related to engine durability, the paper describes (1) improvements in MIL-E-5007D over previous specifications, (2) experiences in applying MIL-E-5007D to the F404 engine development program and (3) assurance test approaches being considered for a revision to MIL-E-5007D.

MIL-E-5007D: U.S. General Military Engine Specification

Description of the General Military Specification for Aircraft Engines

The U.S. military services have used the MIL-E-5007 military specification series for turbojet/fan engines for 30 years (similar documents, MIL-E-8593 series, have been used for turboshaft/prop engines for 25 years). The current turbojet/fan specification, MIL-E-5007D, was published in 1973 (reference 1). Typical of other U.S. military specifications, the document is divided into the following sections:

1. Scope
2. Applicable Documents
3. Requirements
4. Quality Assurance Provisions
5. Preparation for Delivery
6. Notes

Sections 3 and 4 represent the major thrust of the specification, providing, respectively, the technical requirements and the verification/validation tests required to provide reasonable assurance that the technical requirements have been achieved by the engine contractor prior to introduction of the engine into service.

Various revisions of the military engine specification have intended to: (1) improve the clarity of requirements, (2) be more reflective of engine technology advances, (3) incorporate "lessons learned" from problems in previous engine development programs and (4) consider improved test methods/approaches. Each revision has tended to result in a document with more requirements than the previous version and increased testing. The military general engine specification forms the basis for specifying the technical requirements and assurance tests for any new engine development program.

Since the military services provide the engine to the airframer as "government-furnished equipment," the specification requires the engine contractor to define interface information which permits the airframe and engine contractors to integrate their products. In addition, MIL-E-5007D provides performance, functional operation and structural/durability requirements. Although some requirements influence the engine design, the specification generally does not provide specific design guidance.

How the General Specification is Used

MIL-E-5007D is used by the military services as the basis for specifying the engine requirements in any new engine development program contract negotiations. The specification, as published, is "general" in that it covers numerous requirements for the engine contractor to describe the characteristics and features of his engine (e.g., the control system and function of each control unit, maximum operating limits, interface requirements with the aircraft). The military services are also required by the specification to provide specific direction to the potential engine contractors. Paragraphs requiring a military service (Using Service) input usually incorporate such phrases as "when required by the Using Service" or "as specified by the Using Service." When the engine is to be developed solely for a particular weapons system application, the Using Service tailors certain requirements to that specific application (e.g., the 5-hour windmilling requirement may be reduced for a fighter application, requirements to operate in a hostile environment may be reduced for a trainer application, etc.). The tailored requirements are usually furnished in a document which supplements the general specification, as a part of the Request for Proposal (RFP).

In the competition phase for the full scale development program, each potential engine contractor is required to submit an engine model specification in response to the general specification and the tailored requirements document. This model specification is written in the format of the MIL-E-5007D general specification document and reflects the contractor's intent to comply with the Using Service's requirements or provides a vehicle for the contractor to show a lesser or even greater offer against government requirements. The government and contractor will then negotiate requirements which mutually satisfy each party considering economic, life and performance trades. The final negotiated model specification defines the end product which will eventually be introduced into service, and becomes a major contract requirement when the full scale development contract is awarded.

During the engine development program, verification tests may uncover problems which the contractor cannot overcome within the funding, time and technological constraints of the program. Where these problems result in engine capabilities different than the originally negotiated requirements (usually reduced capabilities) the contractor may propose requirement changes to the Using Service by the Specification Change Notice (SCN) process. The SCN process may also be used to update model specification information based on actual test results, to make editorial corrections or to incorporate new technical requirements desired by the Using Service. Approval of an SCN by the Using Service for a different requirement or capability than originally agreed upon may result in a negotiated financial or other settlement between the contractor and the Using Service.

Improved Military Requirements

The requirements of MIL-E-5007D have formed the basis for the development of the new generation of U.S. military jet fighter aircraft engines. The F404-GE-400 engine (F/A-18 aircraft) model specification was negotiated against a Navy-tailored MIL-E-5007D; the F100-PW-100 (F-15/F-16 aircraft) engine specification was negotiated by the Air Force against a number of requirements from which MIL-E-5007D was developed. Since the F404 engine was negotiated directly against a tailored MIL-E-5007D specification, this paper will emphasize the requirements for this engine as evolved from MIL-E-5007D.

MIL-E-5007D is unquestionably a stronger technical document than its predecessor, MIL-E-5007C (reference 2), published in 1965. New requirements, in the areas indicated in figure 1, are intended to better define the engine interface characteristics, and provide a more safe, maintainable and durable engine upon completion of the full scale development program. In addition, lubrication, electrical, starting and control system requirements are more specifically defined compared to MIL-E-5007C.

The structural requirements of MIL-E-5007D offer some of the most important changes from MIL-E-5007C. Structural requirements in earlier MIL-E-5007 documents were relatively nonexistent. In MIL-E-5007C, requirements were added for containment and rotor structural integrity, fatigue life and low cycle thermal fatigue life. Durability requirements were stated as:

> "All metallic or para-metallic parts of the engine shall be designed to at least 5000 hours inherent life within the environmental conditions specified herein and within the power limits, critical speeds, and qualification test schedules specified in the model specification and the vibration survey."

Unfortunately, no test or verification requirements relative to this requirement were specified. A review of all MIL-E-5007 specifications prior to MIL-E-5007D shows that these documents tended to emphasize the interface and performance aspects of the engine. As borne out by service operation, structural problems which affected safety, operational readiness and durability often were not discovered until new development engines reached service. Fortunately, redesign efforts were usually successful, but these efforts delayed maturation of the engines during the production process and caused high engine life cycle costs.

In 1969, the U.S. Air Force developed a structural program for the B-1 bomber engine (F-101 engine). Reference 3 states that this program, referred to as ENSIP (Turbine Engine Structural Integrity Program):

> "is an organized and disciplined approach to the structural design analysis, development, production, and life management of gas turbine engines with the goal of ensuring engine structural safety, increasing service readiness, and reducing life cycle costs through substantially reducing the occurrence of structural durability problems during service operations."

The details of this program are beyond the scope of this paper, but reference 3 provides a good explanation of ENSIP. Most of the structural requirements in MIL-E-5007D evolved from this program. For comparison purposes, the structural requirements of MIL-E-5007C are summarized in Table 1 and the structural requirements of MIL-E-5007D are summarized in Table 2. It is apparent from the tables that MIL-E-5007D structural requirements are much stronger than those of MIL-E-5007C.

In addition to the increased technical requirements of MIL-E-5007D, major changes/ additions were incorporated into the specification for quality assurance testing. Table 1 includes the structural test requirements for MIL-E-5009D (reference 4), the companion specification for MIL-E-5007C, which provided the engine test requirements prior to the development of MIL-E-5007D. Table 2 includes the structural test requirements for MIL-E-5007D. The following tests which influence structural design were also introduced into MIL-E-5007D:

1. Oil flow interruption - operate 30 seconds without oil
2. Oil reservoir pressure - cyclic pressure/fatigue and proof pressure tests
3. Fire - fire tests of flammable fluid lines, fittings, components
4. Ice ingestion - hail and sheet ice ingestion tests
5. Generator/alternator - overspeed, load, containment
6. Foreign object damage - operate with a stress concentration factor of 3 applied to three first stage blades

The qualification endurance test has always been considered the final "proof" that an engine possessed sufficient durability for initial service use. (Unfortunately, for fighter and attack aircraft, this initial service use often amounted to only 350 hours before engine hot section problems occurred.) Table 3 provides a comparison of the major features of the MIL-E-5009D and MIL-E-5007D qualification endurance tests. Each endurance test engine in the MIL-E-5007D program is required to undergo a 300-hour test composed of 50 6-hour cycles similar to the MIL-E-5009D cycles. By doubling the MIL-E-5009D 150-hour test, the new endurance test has more than doubled engine exposure to both high internal gas temperatures and cyclic operation. The increased hot time and cyclic operation required by MIL-E-5007D test is believed to better simulate service operation. The number of required starts has been nearly doubled, also. The increased time at high temperature and increased cyclic operation for the MIL-E-5007D qualification endurance test is expected to cause engine contractors to utilize new or alternate design methods to be assured of passing this more realistic test, with the result that a more durable product will be introduced into service.

In addition to the increased structural test requirements of MIL-E-5007D, other engine and component tests have been added to the quality assurance program. Previously, many of these tests had been run as "good engineering practice" by the contractors. MIL-E-5007D has merely formalized these tests. Excluding the new structural tests, these new formal tests include a Pre-Flight Rating altitude test (formerly included as a contractual requirement), and armament gas ingestion, exhaust emissions, radar cross-section and generator/alternator tests.

Other details on the various engine and component tests required for U.S. military engines are available in MIL-E-5007D, and are far too extensive to discuss here.

<u>Increased Testing</u>

Engine contractors have indicated their concern with the number and length of formal tests required by MIL-E-5007D compared to previously used general specifications. The Aerospace Industries Association pointed out in reference 5 that the number of engine and component qualification tests had increased from 22 to 44, and test time had increased from 543 to 830 hours. The figure of 830 hours did not include time required for LCF testing, too.

The U.S. military services recognized that MIL-E-5007D would require more extensive formal testing than MIL-E-5009D. It was also recognized that engine contractors tended to design to pass these quality assurance tests, and many of the technical requirements in MIL-E-5007C were possibly being overlooked because tests were not required to verify them. As discussed earlier, assurance of a more durable product upon introduction into service requires more complete (and cost-effective) structural verification testing. Considering that past engine development programs have generally required approximately 10,000 engine test hours, including both formal and contractor in-house tests, the increased assurance testing in MIL-E-5007D substituted for contractor in-house tests is expected to provide more productive test hours for the development program.

<u>F404-GE-400 Engine Development</u>

The F404-GE-400 engine has been developed generally against the structural requirements of MIL-E-5007D and required to pass the specified structural tests. Table 4 indicates the structural tests and analyses required in the F404 engine model specifications.

In addition to the structural tests required by MIL-E-5007D, the F404 engine was contractually required to complete other endurance tests consistent with Navy engine development milestones introduced in 1974. These milestones will be discussed more fully later, but they include a Low Production Release (LPR) milestone, which replaced the "qualification" milestone term, and a High Production Release (HPR) milestone. (In the following discussion of the F404 engine program, the term "qualification" will be used vice "LPR" to reflect MIL-E-5007D terminology.) The additional contractual tests resulted in the accumulation of well over 2000 total hours of engine simulated mission endurance testing and well over 2000 total hours of engine accelerated mission testing.

The following paragraphs concerning the F404 engine development program will show the results of applying the MIL-E-5007D structural requirements to the program. They will show that problems were uncovered which otherwise might not have been revealed until the engine reached service. Obviously, a good development program is intended to uncover and correct problems in advance of service operation. It will also be shown that substantiation and verification tests proved that computer programs, in some cases, did not adequately predict stresses expected under operational conditions, and allowed corrections to be made in a timely manner. The total engine development program has provided both the Navy and the engine contractor with more complete engineering information than any previous Navy engine program. This information will be useful in analyzing possible engine problems in service in the future.

Description of the Engine

Prior to a discussion of the F404-GE-400 engine structural integrity program, a brief description of the engine is appropriate. The engine (see figure 2) is a low bypass, augmented type with a 3-stage fan, 7-stage high pressure compressor (each driven by a 1-stage turbine), a throughflow annular combustor and an afterburner. The inlet guide vanes of the fan and the stator vanes of the first three stages of the high pressure compressor are variable. Continuous fan bypass air is provided to the fully modulating afterburner. Five bearings support the rotors. The engine, manufactured by General Electric Company (GE), has a thrust-to-weight ratio in the 8:1 class and provides maximum thrust in the 16,000 pound class.

Low Cycle Fatigue

For qualification purposes, the hot parts tested against the specification low cycle fatigue requirements were the 1-stage high pressure turbine (HPT) disk, 1-stage low pressure turbine (LPT) disk, combustor casing and afterburner casing. The parts were required to complete the equivalent of 1750 sea level static LCF cycles (each cycle defined by a start-IRP-stop) to satisfy the MIL-E-5007D LCF general requirement for fighter engines. The parts were run in various engine builds and tested using Accelerated Mission Test (AMT) cycles to accumulate the equivalent of 1750 LCF cycles. The AMT cycle was developed to relate to the major cyclic and high temperature operation expected by the F404 engine in the F-18 aircraft in service. Each AMT cycle, designed to a time period of 36 minutes, equated to one full LCF cycle and nine full thermal cycles (idle - at least 95 percent IRP - idle) and accelerated service operation by 3.33:1. The nine partial cycles for each AMT cycle were equated analytically to an equivalent number of full LCF cycles to accumulate the required 1750 full LCF cycles for each disk and casing tested. The LPT disk assemblies, the HPT disk assemblies and the afterburner casing successfully met the LCF requirement without incident.

A combustor casing completed 4785 equivalent start-stop cycles, but exhibited a small crack (0.2 in. max. length) at each of six strut outer leading edges. Unfortunately, the casing had not been disassembled for inspection after the required 1750 start-stop cycles because a major disassembly would have been required and there was a desire to accumulate time rapidly on other engine parts in the test vehicle. The records of nine other casings were examined to determine if these cracks were a common problem. One casing had no cracks at 1916 cycles, one had a crack at 1205 cycles and all others had no cracks, although none had accumulated more than 1105 start-stop cycles.

As a result of this investigation, GE ran a comprehensive thermal gradient test of the entire casing and updated its stress model to more accurately reflect the true stresses. Because crack propagation appeared to progress minimally after initiation and offered no problems, the combustor casing was considered qualified by the Navy.

The 3-stage fan rotor and 3-stage forward compressor rotor assemblies successfully completed 3500 start-stop cycles, as required by the specification, in spin pit tests conducted at the Naval Air Propulsion Center (NAPC). It was estimated by the contractor that in 4000 hours of expected F-18 aircraft service operation the F404 engine would accumulate approximately 2000 start-stop cycles. Considering that the fan rotor and forward compressor assemblies were designed for a 4000-hour life, the components were tested to nearly two times their design lives relative to start-stop cycling.

Upon agreement by the Navy and GE, spin pit testing of the assemblies was continued past the 3500 cycle specification requirement. After 6518 cycles the fan first stage disk failed; the contractor had expected the fan assembly to reach 10,000 cycles without failure. The high pressure compressor assembly successfully completed 14,000 cycles. Although the compressor assembly met the specification requirement, the contractor redesigned the first stage fan disk to provide increased LCF life.

To gain more information on the LCF life of the engine rotating parts, an extensive component test program is currently in progress at GE and NAPC. Demonstrations of three times the expected 1000-hour mission requirements are planned.

Engine Pressure Vessel/Case Design

A combustor casing was subjected to a pressure of two times its maximum operating pressure without rupturing to meet the specification requirement. The pressure was increased to 2.22 times the maximum operating pressure before slight local yielding occurred at a stress concentration point.

Strength and Life

A detailed strength and life analysis was performed by GE during the Pre-Flight Rating Test (PFRT) and Qualification Test (QT) programs. Most parts were considered adequately designed to meet expected F-18 aircraft service/mission life requirements relative to stress rupture, fracture mechanics and low cycle fatigue. However, the analysis during the PFRT phase showed that the stage 1 fan disk, stage 3 fan disk, aft fan shaft, inner balance piston seal, high pressure turbine nozzle and combustor would not meet desired LCF life requirements. As a result of the analysis, the stage 1 disk, stage 3 fan disk and aft fan shaft were redesigned by GE for introduction later in the development program. Analysis showed that the HPT nozzle, supposedly designed for a 500-hour life, would show cracks after 350 hours, but was not expected to require repair until 500 hours. Based upon successful testing, GE later changed nozzle life predictions to 1000 hours. The combustor, designed for 1000 hours, was analytically predicted to exhibit crack initiation at 540 hours, but these cracks were not considered by GE to limit the combustor service life (without repair) to less than 1000 hours.

The updated strength and life analysis during the QT phase supported the earlier PFRT analysis and also showed deficiencies in the HPT disk, LPT disk and LPT forward seal based on the materials being used for the early flight test engines. The QT analysis showed that these parts fell short of the minimum LCF design life of 4000 hours required for expected service operation. Material changes for production hardware were expected to correct these LCF deficiencies.

The QT Strength and Life report established the LCF test duty cycle and duty time required for an AMT required by the F404 engine contract. As discussed earlier, a 36-minute cycle was developed. For the AMT, 2000 of these 36-minute cycles, or 1200 hours of test operation, were considered equivalent to 4000 hours of expected F-18 aircraft service operation. Later the contractual AMT was changed to a 600-hour test. The number of thermal cycles in each 36-minute segment of the test was increased from 9 to 12, producing an acceleration factor considered to be at least 4:1.

Finally, the strength and life reports analyzed expected creep, or stress rupture, usage by the engine parts relative to 4000 hours of expected service usage. The fractional life used by each critical part for the time spent at the various power settings was calculated; no major problems were uncovered by this analysis.

As part of the strength and life analysis program, a vibration and stress analysis was performed during the PFRT phase. The vibratory characteristics of the fan, compressor and turbine blades were analyzed to check their mechanical design adequacy. The airfoils were examined by the GE TWISTED BLADE OR MASS computer programs. The dovetail attachments were evaluated by GE using their SINGLEHOOK, MULTI-TANG or ROTOR programs. The analysis showed that vibratory characteristics (vibration-stress distributions, critical frequencies, vibration modes, etc.) were within acceptable design limits.

To verify the vibration and stress analysis, an engine test was conducted with the rotors intentionally unbalanced to at least their maximum limits. The test concentrated on examining vibration and stress levels of shafts, bearings, some disks and other critical components using accelerometers and strain gages. Spectrograms were developed for both velocity and acceleration. Engine running consisted of starts, slow accelerations and decelerations, chops and bursts. The tests showed that all vibration and dynamic stress levels were below maximum allowable levels. In addition, the test history of the number 1 and 3 thrust bearings was reviewed to provide further assurance that bearing thrust loads were consistent with life expectations.

Another engine vibratory test was conducted to concentrate on bearing loads and rotor deflections for an engine dynamically equivalent to the qualification engine. No unforeseen major problems were discovered, although it was known early in the F404 engine development program that there might be some risk associated with the number 4 roller bearing. As a result of high vibrations during early F404 engine testing, the bearing clearances were tightened. This led to excessive edge loading and reduced bearing life. Number 4 bearing failures occurred in the development program despite attempts to provide minor corrections to design/assembly of the bearing. Finally, a larger bearing (same outside diameter, but smaller inside diameter) was designed and was being tested at the time this paper was written.

Creep

A creep analysis showed that the creep-limited components of the engine would not exceed their 0.2 percent creep limits for the desired mission lives of the parts (4000 hours for most parts). For the turbine blades, their rupture lives were required by GE to exceed their stated mission lives (500 hours predicted at that time for HPT blades and 1000 hours for LPT blades); the analysis showed that the mission lives would indeed be exceeded by the rupture lives. For information, the predicted lives of the HPT blades were extended to 1000 hours later in the program due to successful testing.

The 150-hour qualification endurance test, to be discussed later, was also used as a basis for assuring that creep would not be a major engine problem. For the high pressure turbine blades, the 150-hour qualification endurance test was considered a severe test, since the blades were supposedly designed for 500 hours and the 150-hour qualification endurance test required afterburner and intermediate power operational time well above the

total high temperature time expected in 500 hours of service operation. For parts designed for 4000 hours, the 150-hour test was not considered severe. All engine parts successfully completed the 150-hour qualification endurance test without excessive creep.

Containment

Containment tests were conducted during the qualification phase of the program for the fan module, high and low pressure turbines and forward compressor rotor assembly. In each case, one of the highest energy stage blades was notched in the airfoil just above the attachment platform to cause failure at maximum allowable transient speed. The casings successfully contained the damage resulting from each blade failure.

Rotor Integrity - Overtemperature and Overspeed

During the PFRT phase, the engine was tested to satisfy the specification overtemperature requirement of five minutes operation at a T_{5H} temperature (low pressure turbine inlet temperature measured at the engine electrical harness) of $75^{\circ}F$ above the maximum allowable steady-state temperature. Both the fan and gas generator rotors were required to operate at or above the maximum speed limits. The test was conducted for five hours, 15 minutes, at or greater than the specification required conditions. No damage was sustained and engine part dimensions were within acceptable limits.

Compressor and fan rotor assemblies were operated in NAPC spin pits to meet the specification overspeed requirements. They were required to operate for five minutes at a speed simulating 115 percent of maximum allowable engine steady-state speed at the corresponding disk metal temperatures. The disks were actually run at speeds higher than 115 percent to compensate for conducting the spin pit tests at room temperature rather than at the appropriate disk metal temperatures.

Neither the fan or compressor had to repeat the overtemperature and overspeed tests for qualification because the PFRT and QT parts were similar. The fan stage 1 disk, stage 3 disk and fan aft shaft were improved for LCF life during the qualification phase of the program, but the production parts were expected to have at least the same overtemperature and overspeed capabilities as the PFRT parts.

Disk Burst

Disk burst tests were not required because the overspeed test requirements of 115 percent of maximum allowable speed corresponded to the F404 engine specification disk burst requirements.

Vibration

Engine vibration measurements were taken in the PFRT phase of the program under steady-state and transient conditions. Using vibration accelerometer pickups, no destructive vibrations were evident in the operating range of the engine. Other extensive vibration and stress surveys conducted were discussed earlier under "Strength and Life."

Externally Applied Forces

Prior to performing a static load test of the engine, all static structural elements which transmit to the engine mounts inertial, gyroscopic, and thrust forces resulting from flight maneuver conditions were analyzed. The GE MASS, CLASS/MASS and VAST computer programs were used in the analysis. The CLASS/MASS program analyzed flanges, shells and other axisymmetric members; the VAST program analyzed the complete engine system. The analyses, performed in the PFRT phase of the program, developed margins of safety for the parts analyzed. For a number of parts, test data were also compared to the analysis. In some of the cases, the tests revealed significantly higher stresses than the computer analysis, indicating some imperfections in the computer programs.

Next, the engine static structures were subjected to loads simulating the maximum combinations of inertial, gyroscopic and thrust forces, and the resultant mount reactions. The specification required the parts to withstand the limiting maneuver loads without permanent deformation and 1.5 times the maneuver loads with no evidence of cracking, buckling or permanent deformation. Generally, the stresses for all parts were well below the material yield strengths up to the 150 percent maneuver load operating conditions. However, the turbine exhaust frame mount ring support links, although acceptable for the 100 percent load case, were not acceptable for the 150 percent case. As a result, the links were redesigned to increase strength, and the new design proved to offer good load margin when tested later during the qualification phase. A repeat of the PFRT engine static load test was not required during the qualification phase because of similarity of PFRT and QT parts.

Gyroscopic Moments

An analysis was performed during the qualification phase of the program to determine the effect of the maximum expected maneuver loads on engine rotor-to-stator deflections and clearances. The analysis showed that the cycling gyro loads were not expected to result in any parts failure or produce severe enough rubs to affect satisfactory engine operation (rubbing was predicted under some limiting conditions). Generally, good stress margin was indicated for engine parts. One significant problem was uncovered by

the analysis, however. Stresses at the low pressure turbine torque cone rim bolt holes exceeded the minimum -3σ material vibratory allowable stress limits. The flange was later thickened to correct the problem.

F404 Engine Qualification Endurance Test Results

The qualification endurance test has always been considered the most significant test for durability in the qualification program. MIL-E-5007D requires two 150-hour tests on an engine. The F404 engine was required to complete one 150-hour test on each of two engines, each engine using a different fuel and lubricant. (Later, due to financial constraints, the Navy changed the formal requirement to one engine test on JP-5 fuel, the Navy's primary fuel. A 150-hour AMT was used as a vehicle to qualify the engine on JP-4 fuel). The F404 engine was also contractually required to run an additional 150-hour endurance test on one of the engines for demonstration purposes. MIL-E-5007D requires the test to be conducted at sea level conditions, although some operation is required (one-third of the test time) at high inlet pressures and temperatures corresponding to certain maximum flight conditions. Nearly half of the F404 test was required to be conducted at inlet temperatures and pressures corresponding to high Mach number conditions at altitude. Table 5 presents some of the significant qualification endurance test requirements for the F404 engine program for comparison to the MIL-E-5007D requirements in Table 3.

Although the MIL-E-5007D qualification endurance test was never perceived to represent a particular number of engine hours in a service environment, there is a temptation to compare the F404 endurance test against expected service operation. One survey presents expected service usage information for the F404 engine for 1000 hours of service operation in the F/A-18 (reference 6). The engine is expected to perform 525 starts, 5990 partial stress, or thermal, cycles (idle - intermediate - idle) and operate 120 hours at maximum temperature. Considering that the F404 engine was required to operate at 17°C above the rating temperatures for the various engine ratings during the qualification endurance test, the stress rupture capabilities of the engine were tested fairly well. In the area of centrifugal and thermal stress low cycle fatiuge, however, the F404 engine endurance test was not expected to be a good representation of predicted service usage for 1000 hours.

No major structural problems were uncovered by the endurance test, but some part failures did occur during the test. The most significant problems were a cracked fuel nozzle pigtail (the pigtail is located between the fuel manifold and nozzle), cracked variable exhaust nozzle secondary seals and an afterburner pump seal failure. Fixes have been initiated to correct each of these problems. The post-test inspection of all engine parts generally showed that all parts were in acceptable condition and within serviceable limits, except for the high pressure turbine nozzle. One of the HPT vanes showed excessive trailing edge distress, believed to be caused by some plugged cooling holes. A change to the processing and inspection procedure during manufacturing was expected to correct the problem.

Where Do We Go From Here?

Since the publication of MIL-E-5007D in 1973, some key factors have occurred to provide impetus for the development of a new revision of MIL-E-5007. The factors include:

a. New engine development milestones have been defined.

b. Concepts of Accelerated Mission Tests (AMT) and Simulated Mission Endurance Tests (SMET) have been developed.

c. Aircraft mission usage profile definition capability has been developed.

d. Office of Management and Budget Circular No. A-109, Major Systems Acquisitions, of April 5, 1976 was published.

e. A Task Force on Specifications and Standards Improvement was chartered as a panel of the Defense Science Board in 1974 and published a final report with specific guidance in April 1977.

In the following paragraphs, some of these factors will be discussed and explored more fully.

New Engine Development Milestone - High Production Release

By the late 60's, both the U.S. Navy and Air Force recognized that old engine design methods were not going to be so easily applied to engines utilizing the newer, lightweight materials and designed with the newest engine technology, including more complex control systems and variable geometry systems. Previously, engine performance was the major consideration on all new engine development programs, and any durability problems could usually be overcome with reasonable effort either in development or production. With the newer technology engines, a different approach to assuring that the engine development programs would continue to produce reliable, durable engines for service became necessary. Fortunately, many new design concepts and assurance test requirements related to engine durability were incorporated into MIL-E-5007D. Still other concepts and approaches have been developed over the past few years and should be considered in a new MIL-E-5007 document. In 1974, the Navy introduced a new engine development program phase and

milestone, and a new endurance test. Engine development programs have classically used the PFRT and QT milestones to clear the engine for flight tests and for production, respectively. The military general engine specification provides the assurance test requirements which must be satisfied to achieve these milestones.

To increase the assurance that an engine would be sufficiently durable and reasonably problem-free upon introduction to full service operation, the Navy added a High Production Release milestone to the development program. The HPR phase of the development program was constructed to include a 1000-hour SMET and an engine LCF test, each test to be performed on an engine produced by production tooling. The SMET was intended to be composed of basic missions expected to be flown by the aircraft in service. The missions were to be mixed in the appropriate proportion for 1000 hours of service operation. An Accelerated Service Test (AST) was also introduced into the HPR phase. The AST was designed as a 1000-hour service type test of the engine(s) in an aircraft. The engine was to be maintained by the contractor for the first 500 hours of the test and by the Navy for the second 500 hours. The test was aimed primarily at assessing maintainability, reliability, durability, maintenance manuals, service limits, etc., in typical service operation.

Since the HPR phase of the engine development program was designed to approve the engine for full production, the former QT phase was redesignated as the Low Production Release phase. An engine, upon completion of the LPR phase, would be permitted to be produced in a limited manner for Navy aircraft assurance tests and limited initial service operation. A typical development program utilizing these milestones and tests is shown in figure 3. These concepts were generally incorporated into the F404 engine development program, although the SMET requirement was for 750 hours rather than 1000 hours. A revision to MIL-E-5007D should consider this approach for engine durability assurance.

The concept of a SMET as the final durability test requirement in an engine development program seems appropriate; it offers a final "proof" that the developed engine, manufactured with production tooling, can perform satisfactorily utilizing the same missions as service engines. Obviously, a sample of one engine provides little statistical assurance and confidence that all production engines will reach 1000 hours in service with no durability problems. Additional approaches to assurance testing to augment one-to-one SMET testing for durability will be discussed later.

Mission Data Development

Mission-oriented testing requires one major ingredient - definition of the expected service missions (and the proportionate mix of missions over some period of service life). To acquire mission definitions for Navy fighter aircraft, Navy and industry (Pratt and Whitney Corporation) representatives surveyed Navy fighter pilots in 1974. Typical fighter missions and a mission mix were developed and were utilized to define the initial F404 engine SMET. In recent years, the Navy has developed a more detailed and systematic approach to acquiring and updating mission information for all types of aircraft (e.g., fighter, attack, patrol, trainer, helicopter). The approach has been to: (1) interview pilots and instructors, (2) acquire engine flight test data by flying similar aircraft against the general missions stated by the pilots, (3) store and continually update mission information in a computer system and (4) utilize the data to develop mission-oriented endurance tests. Reference 6 describes the program more fully and provides a good description of some of the typical missions flown by Navy aircraft.

With this ongoing system for maintaining knowledge of Navy aircraft/engine operational use, the Navy has developed a tool which will be integrated into future engine specification revisions.

Accelerated Testing

The SMET concept which requires test time equivalent to operational mission time on a one-for-one basis is a costly test, considering the cost of fuel and other resources, and requires a lengthy test period for its accomplishment. In the last few years, accelerated mission testing approaches have been developed to compress test time. As discussed earlier, the F404 engine development program has utilized the AMT, sometimes referred to as an Accelerated Simulated Mission Endurance Test (ASMET), approach for some of its assurance testing. An AMT is a shortened SMET intended to reveal creep and LCF problems in the hot section of the engine; time at engine non-damaging internal temperatures and minor speed cycles are deleted from the SMET to form the accelerated test.

Extensive experience with SMET and AMT testing has been gained by the Navy as a result of TF30 engine testing over the past six years. The NAPC performed a 750-hour SMET on a TF30-P-412A engine in 1974, a 400-hour ASMET on a TF30-P-412A in 1976 and a 1000-hour SMET on a TF30-P-414 in 1977. Pratt and Whitney Aircraft Group also performed testing and analyses during this period. This total experience and utilization of F-14 aircraft mission data resulted in the development of three generalized AMT mission cycles for the TF30-P-414 engine. These mission cycles are being used in the appropriate proportion by Pratt and Whitney as the basis for accelerated testing to uncover durability problems before these problems occur in service (i.e., "lead-the-fleet" concept). Accelerated mission testing up to 1000 equivalent flight hours has produced a good correlation of engine hardware condition with TF30-P-412/414 engine service experience.

Reference 7 describes the concept of accelerated mission testing in depth and discusses Air Force work with the TF34-GE-100 engine (A-10 aircraft) utilizing the AMT approach. The TF34 engine tests utilized an acceleration factor of 2:1. The Air Force has also performed extensive AMT testing with the F100-PW-100 engine, utilizing an acceleration factor of 3:1. Because of the large amount of operational time at high engine temperatures and the number of large speed cycles, fighter and attack aircraft AMT's are difficult to accelerate by more than 1.5-3:1. It is obviously desirable to accelerate endurance test time to as high a degree as possible. At least one company, Pratt and Whitney, is currently studying accelerated-accelerated test (A^2MT) approaches for turbine airfoil evaluations, which could conceivably reduce the time to run an AMT by 75 percent for some evaluations.

It is recognized that the AMT (and SMET) have some potential shortcomings for uncovering hot section problems. As discussed in reference 8, some of the problems associated with the development of the mission-oriented tests are:

1. Sufficient operational data are lacking, so pilot survey data are used as the basis for the development of generalized missions.

2. Engine data acquired from flight tests are generated in a rather controlled environment by experienced flight test pilots (the Naval Air Test Center generates data for the Navy) as opposed to operational usage with less experienced pilots in an uncontrolled environment.

3. Environmental effects are not factored into the missions.

4. SMET/AMT testing is usually accomplished in a sea level static test facility; the effects of high altitude operation on the durability of the engine are not completely duplicated in the test cell (F404 engine testing, discussed earlier, did include the effect of high ram conditions at altitude during the SMET and AMT).

5. Mission data, usually developed from older aircraft/engines used to fly the missions, often has to be analyzed and revised to be more representative of the new weapons system.

These shortcomings have not appeared to be crucial in the case of the TF30 engine. Although the AMT as currently run does not show the magnitude of hot corrosion, or sulphidation, found in service engines, correlation between the condition of test engine and service engine parts has been good. Hopefully, for new development engines, the projected, or expected, service environment (missions and mission mix) will permit good correlation between SMET, AMT and service engines.

Rotating cold parts, considered as parts which are not in the hot gas path of the engine, are most often life-limited by LCF. SMET and AMT tests are usually of insufficient time duration to reach the service life for the cold parts because cold parts are designed to last 4000 - 8000 hours, and SMET and AMT tests are aimed at uncovering earlier hot part problems. Since temperature cycling is not a major consideration in the life usage of cold parts, accelerated test methods can be employed which subject the parts to only the major stress cycles for the appropriate expected mission life of the hardware. An Accelerated Low Cycle Fatigue Test (ALCFT) of the complete engine or of components in a spin pit can reduce test time by a 10-20:1 ratio (i.e., 8000 service hours can be represented by 400 - 800 hours of testing). For pure LCF testing, hardware (e.g., engine disks and pressure vessels) should be cycled to include engine start-IRP-stop and idle-IRP-idle cycles. MIL-E-5007D requires an engine LCF test for a start-max. rpm-stop cycle which the contractor is required to develop. The specification also requires that the individual engine components (e.g., fan, compressor, combustor, turbine) which are LCF life-limited be tested to two times their lives. Spin pit tests for these engine disks appear to be the cheapest, most practical approach to testing these parts for LCF resistance. These engine and component specification tests are still considered appropriate, but mission information, now available, should be utilized to develop the cyclic test requirements.

Earlier, cyclic and temperature features of the MIL-E-5007D 300-hour qualification endurance test were described. It was pointed out that the general cycle used for the test was heavily weighted towards checking creep and stress-rupture of hot parts. The test also requires the demonstration of various limits, such as maximum accessory loading maximum customer bleed airflow, anti-icing valve actuation, maximum allowable gas temperatures, etc. The Navy is presently reviewing the 300-hour qualification endurance test requirements. One proposal under consideration is to substitute the MIL-E-5007D endurance test with two tests - an AMT equivalent to a specified service interval for the engine and a Durability Proof Test (DPT) which would exercise various engine functions and demonstrate various limits. An AMT would impose both creep and LCF conditions on the hot parts as assurance of their durability. The DPT, a test conceived to be approximately 60% of an AMT, would be used to demonstrate specification limits and assure that operation at those limits for a reasonable time period would not detrimentally affect the parts.

Reference 8 provides additional information on a Navy "Revised New Look" program which requires accelerated assurance testing. Figure 4 depicts this program with a few modifications. Note that a new development milestone, Developmental Release (DR), has been added to provide additional assurance early in the program that sufficient progress has been made towards achieving durability requirements.

These significant changes are being discussed within the Navy now and a position will eventually be coordinated with the other services to develop a revised military engine general specification. The Navy is currently developing the details of some of these tests in specification format.

The Navy intends to require that all future engine development programs include AMT and accelerated LCF tests from early in the development program to the end of it. Specific engines will be assigned solely to perform these durability tests to uncover failure modes, track crack progression of hot parts and develop some assurance that design predictions and methods are correct.

MIL-E-5007D Revision

The preliminary stages leading to a formal revision of MIL-E-5007D, now seven years old, are underway. Each of the three services is studying the total specification, and the coordination process among the services should begin soon. Impetus for a revised specification has resulted from engine technology improvements, especially in the understanding of failure modes, durability problems and associated test techniques. (Some recommended specification improvements relative to durability testing have been discussed previously in this paper.) Also, there is a desire to make the military general engine specification more "tailorable" as a result of conclusions drawn by the Task Force on Specifications and Standards Improvement, which published its findings in 1977. The Task Force considered the U.S. military specifications as "technically superior to their commercial counterparts" and "essential" for procurement purposes. However, the Task Force recommended a joint government/industry effort to effectively tailor the application of specifications and standards. Reference 9 provides a more comprehensive general discussion of the Task Force's findings.

Closing Summary

MIL-E-5007D, as the general specification used by the U.S. military services to procure turbojet/fan engines, has provided significantly improved requirements over previous military specifications. Major improved durability requirements have paid dividends on the U.S. Navy's F404-GE-400 engine development program. Some potential design weaknesses, uncovered during the development program through required structural analysis and testing, may not have appeared until the engine reached service if MIL-E-5007D requirements had not been adhered to. In recent years, however, the development of better, more complete aircraft mission data and a better understanding of engine failure modes has provided impetus for a new specification revision. Specification improvement must be a continual process which keeps pace with engine technology advances and "real-world" experience with service engines. Maintaining engine specifications current will assure that engines developed in the future will be durable and operationally effective.

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TABLE 1

MIL-E-5007C/MIL-E-5009D Structural Requirements

Structural Considerations	Technical Requirement	Test Requirement
Durability	Design for 5000 hrs. inherent life of metal and para metallic parts within environments, limits, qualification test schedules	None
Low Cycle Fatigue (LCF)	LCF life of at least 12,000 hours of endurance under service usage	Complete 1000 cycles prior to endurance test
Fatigue Life	10^7 cycles for steel parts; 3×10^7 cycles for non-ferrous alloy parts	150 hr. test to approach 10^7 cycles for steel parts and 3×10^7 cycles for ferrous alloy parts at various engine speeds in flight envelope - concentrate at critical speeds and high stress
Containment	Compressor and turbine cases shall contain failed rotor blades	None
Rotor Integrity	Strength to withstand 115% max. allowable steady state speed at max. temp. for 5 min.; 42°C above max. gas temp. at max. allowable steady state speed for 5 min.	'Substantiation' required
Vibration	Free of destructive vibration	Test (or analysis, if test is impractical)
Flight Maneuver Forces Simulated Flight Maneuver Loads	No permanent deformation with conditions specified in MIL-E-5007C; no failure at static loads of 1.5x max. conditions specified	None
Gyroscopic Moment	Withstand gyro moment of 3.5 rad./sec. in yaw at +1g for 15 sec.	None

TABLE 2

MIL-E-5007D Structural Requirements

Structural Considerations	Technical Requirement (1)	Test Requirement
Structural Life	Consistent with system life	-
High Cycle Fatigue	Same as MIL-E-5007C fatigue life requirements	-
Low Cycle Fatigue	Consistent with cycles in spec.	Component test - 2x life requirement; Engine test 1 life
Engine Pressure Vessel/Case Design	2x max. operating pressure without rupture	Test engine case and gas pressure loaded components against requirement
Strength and Life	Submit analysis report	Use stress report for duty cycle and time for engine LCF test; vibration and stress test
Design Material Properties	-3 sigma value	
Creep	Analysis	Endurance and LCF tests
Containment	More extensive than MIL-E-5007C; must contain fan blades, too	Full scale engine or spin pit tests
Rotor Integrity	Same as MIL-E-5007C	Overspeed and overtemp. tests
Disc Burst Speeds	No failure up to 122% max. allowable steady speed	Spin pit testing of all critical rotating disk components
Vibration	Similar to MIL-E-5007C, but more detailed and specific	Engine vibration survey, including acceleration spectrograms
Critical Speeds	[illegible] margin above max. operating speed	
Externally Applied Forces	Same as MIL-E-5007C flight maneuver loads and simulated flight maneuver loads	Engine static load test
Gyroscopic Moment	In addition to MIL-E-5007C requirement angular velocity of 1.4 rad./sec. and max. load for infinite life	Gyro test to demonstrate 3.5 rad./sec. for 15 sec.

Note: (1) For the structural considerations which are unchanged from MIL-E-5007C, the technical requirement listed replace the MIL-E-5007C requirement unless stated otherwise.

TABLE 3

Summary of Qualification Endurance Test Requirements (1)

	MIL-E-50[illegible]	MIL-E-5[illegible]
No. of engines	2 (2)	2
No. of test hours/engine	15[illegible]	[illegible]
Total test hours for program	[illegible]	[illegible]
Time at intermediate rating or higher (hrs.) engine	[illegible]	1[illegible]
Time in afterburner operation (hrs./engine)	[illegible]	[illegible]
Power lever movements/engine	1[illegible]	[illegible]
Thermal cycles/engine (idle - IRP - idle)	35[illegible]	[illegible]
No. of starts (3)/engine	1[illegible]	[illegible]

(1) Assumes engine with modulated afterburner
(2) Two tests are required if the engine is to be qualified on two primary fuels (one test if only one fuel).
(3) MIL-E-5007D requires that all starts not accomplished during the endurance test be followed by an acceleration to max. continuous thrust, dwell for 30 seconds, then shut down; this was not a MIL-E-5007C requirement.

TABLE 4

F404 Engine Structural Test/Analysis Requirements

Structural Considerations	Test/Analysis Required by Model Specification
Low Cycle Fatigue	Engine test for one equivalent engine life; component tests - [illegible] start/stop cycles for certain cold parts and 175[illegible] for hot parts (actual tests each part [illegible])
[illegible] Pressure Vessel Case Design	Test combustor casing [illegible] x max. operating pressure without rupture
Strength and Life	Stress analysis and vibration and stress tests (including engine system vibration tests, blade and vane aeromechanical tests and accessory gearbox tests)
Creep	Creep analysis and qual[illegible]
Containment	Analysis and spin pit tests
Rotor Integrity	Overspeed and overtemp [illegible] requirement
[illegible]	Spin testing to 115% max. allowable [illegible]
Vibration	Vibration test survey at [illegible] acceleration spectrum
Externally Applied Loads	Analysis and static tests [illegible]
[illegible]	Analysis

TABLE 5

Summary of F404 Engine Qualification Endurance Test Requirements

No. of engines	2
No. of test hrs./engine	150
Time at IRP and higher (hrs)	78
Time in afterburner (hrs.)	47
Thermal cycles	·375
No. starts	387

Interface - PTO, inlet, armament gas ingestion, bleed connections, transient airflow, fuel

Maintainability

Environmental - fungus, humidity, exhaust smoke, invisible emissions

Design and Construction - adhesives, self-retaining bolts, securing of fasteners, clamps, screw recesses, drives

Safety

Structural Performance - life, pressure balance, pressure vessel/case design, strength and life analysis, design material properties, creep, disk burst speeds

Major Component Characteristics - Engine Control

Electrical Systems

Instrumentation

Oil

Exhaust Nozzle

Thrust Reverser

Water Injection

Fig.1 New paragraphs/requirements

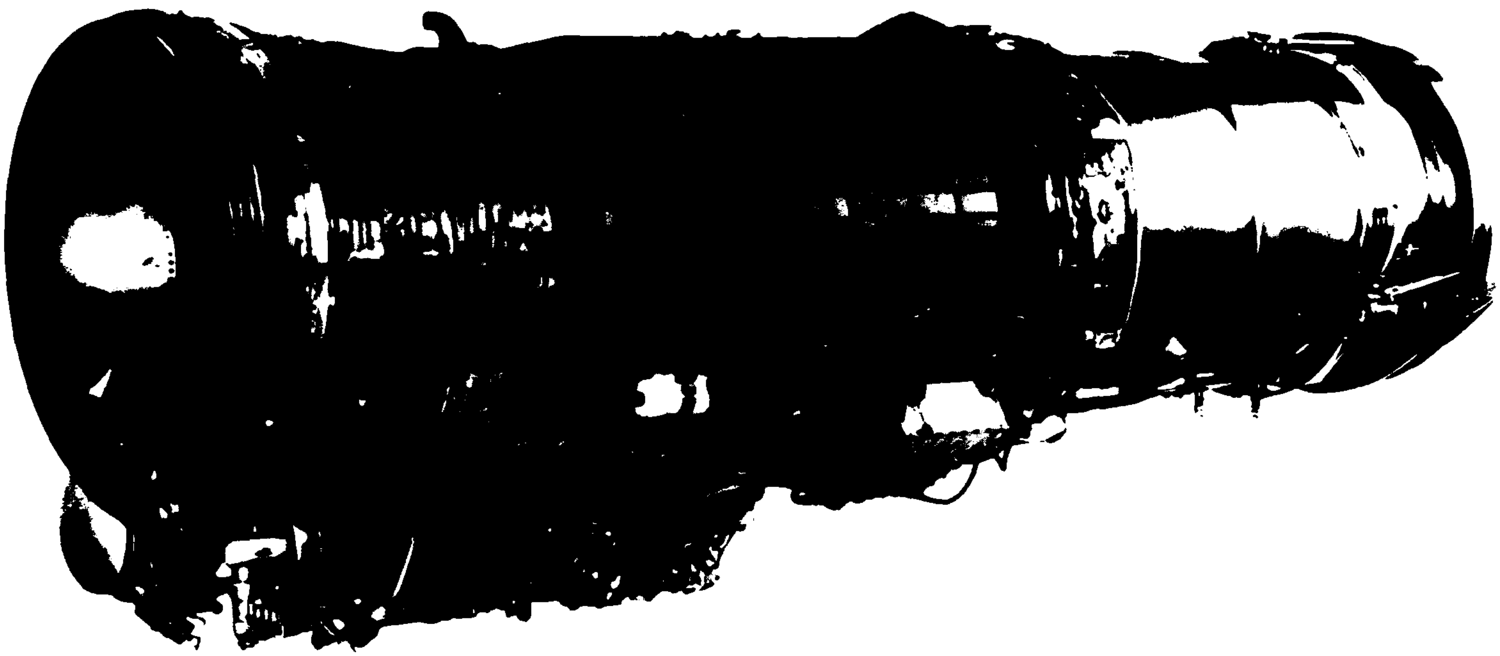

Fig.2 F404-GE-400 engine

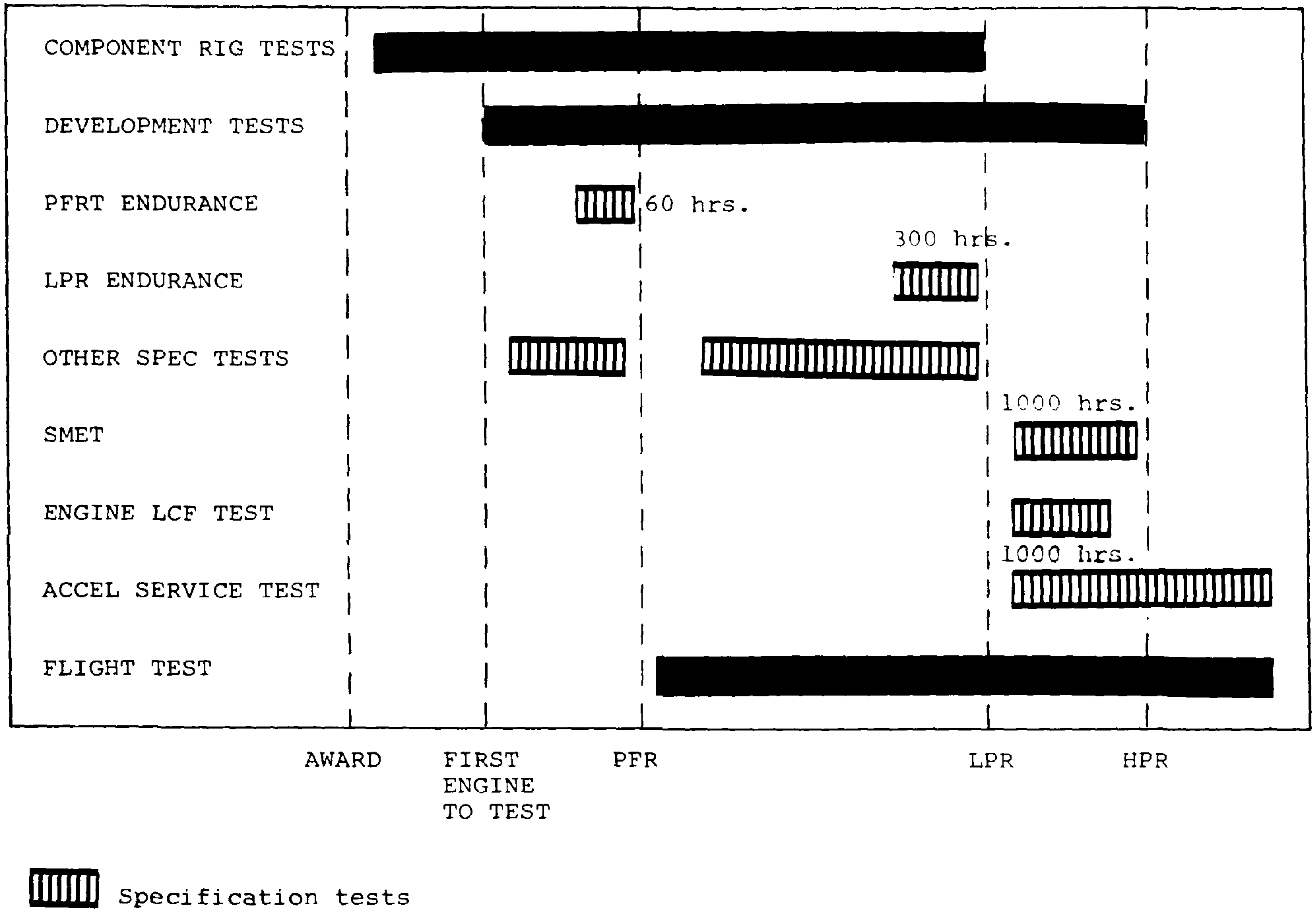

Fig.3 US Navy new look

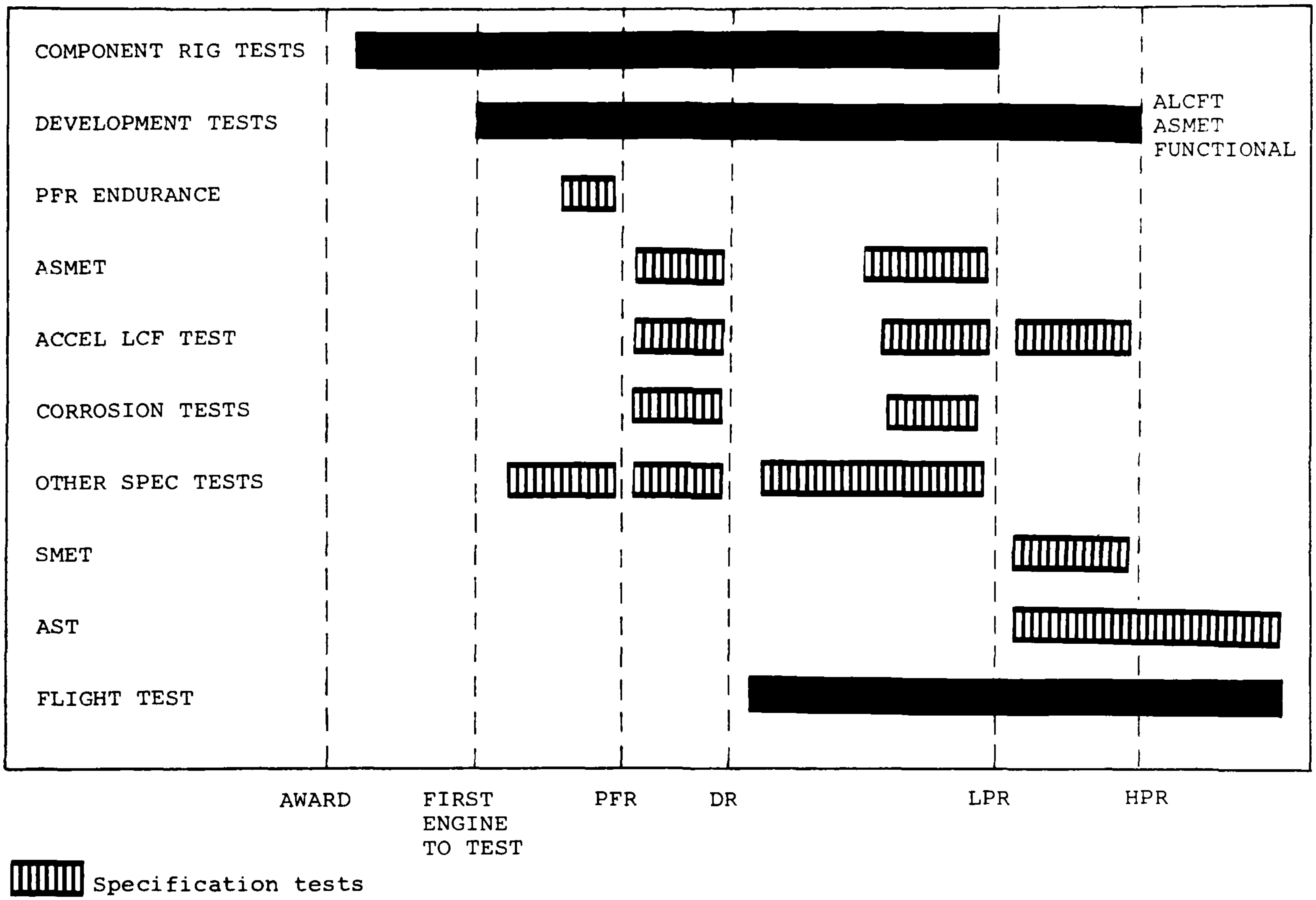

Fig.4 US Navy revised new look

DISCUSSION

J.Fresco, Turbomeca, Fr

On various test requirements of MIL-E-5007D such as rotor integrity test, it is requested to run at maximum temperature.

What does maximum temperature mean with respect to ratings when one engine failure ratings are envisaged?

Author's Reply

It is my understanding that this question relates to the problem of testing to consider a contingency, or emergency, engine rating. The Navy has not normally accepted such a rating. On a recent program where it was permitted, a demonstration of the rating was required a certain number of times during the QT endurance test.

M.D.Paramour, Ministry of Defence, UK

I understand that the US Navy has introduced a new specification NAPC-T-79002. Does this specification include the proposed revision to MIL-E-5007D outlined in your paper, and what is the current status of the new specification?

Author's Reply

NAPC-P-79002 is an interim Navy specification which incorporates some specification revisions which were developed as a result of "lessons learned" in the 1975 1978 time period. The revisions have been applied mainly to Section 3 of the specification. A more extensive revision of Section 3 has been prepared and is being coordinated within the Navy. Most of the test concepts introduced in the paper have not been incorporated into Section 4 yet, although effort in underway.

The Air Force, custodians of MIL-E-5007D, are presently developing a revised specification which will be eventually coordinated by the Military Services.

P.F.Ashwood, National Gas Turbine Establishment, UK

What proportion of the certification programme requires the use of altitude test facilities, and do you use such facilities for endurance testing?

Author's Reply

Official tests which are usually conducted by the Navy in altitude facilities include: PFR and QT altitude, inlet distortion and armament gas ingestion tests. Other tests requiring the use of special facilities (e.g. conditioned inlet air) are: low and high temperatures, corrosion, sand and dust, gyroscopic moments and infrared tests. Rotor integrity tests including LCF and containment tests of components are also often undertaken in Navy spin pits.

Endurance tests are usually conducted in sea level facilities and may utilize high pressure facility air conditioned to a desired temperature. Endurance tests at altitude have been considered by the Navy, but the use of electricity to operate plant equipment makes this type of test costly (the savings in fuel and its cost does help offset the high electricity cost) and lengthy endurance tests reduce the availability of precious altitude test facilities. At this time it has not been proven that testing at altitude for durability verification purposes will provide much more information than testing at sea level.

Certification Procedure for Military Engines in Germany

Fritz Biel

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SUMMARY

It will be shown how in the Federal Republic of Germany Qualification and Certification of military used aeronautical equipment, here engines, is in agreement with the airtransportation legislation of this country on the one hand and how it is bound in the development and usage of a weapon system on the other hand.

The activities in the course of development and during usage will be presented with special emphasis on the aspect that BWB-ML* is only occupied with the subject of airworthiness free from the need for the pursuance of schedule and financial matters. Differences to the procedures and the organization of other countries, if known, will be shown.

INTRODUCTION

In popular-science encyclopaedias AIRWORTHY is defined as FIT to FLY. However thosewho are engaged in the aeronautical field have to consider other, i.e. official definitions as for example the definition of the British Standard Institution (B.S. 185, Section 1)

> AIRWORTHY: Complying with the regulations prescribed by the competent authority certifying the fitness for flight of an aircraft.

This definition means that during development and manufacture of aeronautical equipment, in this case an engine, not only technological and scientific fundamentals but also legal and administrative aspects have to be considered.

Those legal and administrative aspects which we have to consider in the Federal Republic of Germany differ in some areas from those valid in partner countries. This we experienced throughout the development phase and also when approaching entry into service of the two most important projects of the German Air Force, the ALPHA JET (joint production with France) and the TORNADO (joint development and production with Great Britain and Italy) as well as when adopting aircraft originally developed for the civil market, such as VFW 614.

Having learnt this lesson we feel in the interest of both parties, the companies and the certification authorities, knowledge of the regulations and proceedings plus understanding of the restraints put on the officials are most important because this will help to simplify and facilitate cooperation between companies and the authorities within their country and also cooperation between the authorities and companies participating in a joint project.

STATUTORY FUNDAMENTALS

In the Federal Republic of Germany for both military and civil air traffic the Luftverkehrsgesetz (LuftVG) = Air Traffic Law applies uniformly. Thus it has been ensured by law that the public safety will not sensibly be affected neither by military nor by civil aircrafts. The LuftVG is a prevention law, i.e. it requires that before any aeronautical equipment can be released for flight it has to pass an official assessment and get approval from the appropriate authority.

This law is executed in the Federal Republic of Germany by different authorities independent of both the manufacturer and the user. In this context only the complex of qualification and type certification of engines will be followed up.

The Type Certificate will be given:

- for civil-used engines by the Luftfahrtbundesamt (LBA) = Federal Aviation Agency

and

- for military-used engines by the Leiter des Musterprüfwesens für Luftfahrtgerät der Bundeswehr (BWB-ML) = Director of Aeronautical Equipment Qualification for the Federal Armed Forces.

This separation between civil and military engines was considered to be necessary for

*Note:Names of authorities and abreviations are given according to the German designation

various reasons, among others the protection of military secrets. The legislator realized the different treatment by granting an exeptional provision in the LuftVG. Apart from the different administrative responsibility this provision also permits departure from the LuftVG as far as necessary to accomplish the special tasks of the Federal Armed Forces.

INDEPENDENT AIRWORTHINESS AUTHORITIES

In the Federal Republic of Germany the general view is taken that the rights of the general public can best be safeguarded impartially by an independent institution. By this way the different interests of the parties concerned, i.e. the manufacturer, the user, and the general public are to the utmost respected.

Although engaged in the whole life cycle of aeronautical equipment BWB-ML is independent not only of the user, that is the Air Force, but also of the divisions responsible for development and procurement within the Bundesministerium der Verteidigung (BMVg) = Federal Ministry of Defense and the Bundesamt für Wehrtechnik und Beschaffung (BWB) = Federal Office for Military Technology and Procurement. BWB-ML is directly subordinate to the President of the BWB which demonstrates its independence from the organizational point of view as well. Figure 1 shows the integration of BWB-ML into the armament branch, and Figure 2 presents the organizational structure.

ENGINE LIFE CYCLE

The procedure for the introduction of equipment for the Federal Armed Forces is covered by the " General Regulations for the Development and Procurement of Military Materiel." The flowchart contained in these regulations is subdivided in individual phases. At the end of each individual phase an assessment has to be made and a decision how to proceed has to be taken.

The individual phases and their decisions are as follows:

Pre-Conceptual Phase	Military Needs
Conceptual Phase	Operational Requirements
Definition Phase	Operational Economical Requirements
Development Phase	Approval for Introduction into Service
Procurement Phase	Approval of Final Report
In-Service Phase	

The Type Certificate of an engine is one precondition for the decision of the introduction into service. It has been established by internal administrative directive at which point BWB-ML is to be included into the equipment life cycle. The following regulation exists:

- During the Conceptual Phase BWB-ML is notified about major events in a purely informative way, for example BWB-ML is provided with a copy of the Military Needs.
- During all the other phases BWB-ML takes an active hand in the programme.

In Germany we are of the opinion that inclusion of the Airworthiness Authority early in a project is beneficial to it. The parties concerned have to have a clear understanding about all requirements with regard to airworthiness already when defining the engine. Also from the technological and economical point of view early participation is absolutely necessary. Due to reasons of cost and time, today, it is no longer justifiable that an engine manufacturer approaches the Airworthiness Authority for the first time after completion of development requesting qualification. BWB-ML design and verification requirements, e.g. with respect to necessary minimum performance, structural strength, or functional reliability, particularly when using new materials or not yet covered new technologies represent a cost factor which must be known to the manufacturer by all means prior to tendering.

In case of complex equipment, such as an engine, and due to the partly opposite interests of the user, the procurer, and the Airworthiness Authority coordination too is absolutely necessary. Balance can only be achieved if all concerns are duly taken into account. Balance will be disturbed immediately if one of the coordinated factors is altered, e.g. if the user's request increases costs will increase and airworthiness will possibly be affected. This balance can be ideally illustrated by an equilateral triangle containing both the functions and the parties involved (Figure 3).

INTERNAL ADMINISTRATIVE PROCEDURE

Information on the military administrative organization within the Federal Republic of Germany should be concluded by referring to the regulation which generally governs the work of the Airworthiness Authority Representative. It is the Joint Services Regulation ZDv 19/1 " Das Prüf- und Zulassungswesen für Luftfahrtgerät der Bundeswehr " = Inspection, Certification, and Licensing Procedures for Aeronautical Equipment of the Federal Armed

Forces. Each Airworthiness Authority Representative has to comply with the provisions contained therein. Individual chapters of the ZDv 19/1 particularly those covering cooperation with industry are seperately compiled and are normally used by BWB in this form for incorporation into the contracts placed on the manufacturer of aeronautical equipment.

QUALIFICATION

A precondition for issuing a type certificate is the successful conclusion of the qualification. It has to be demonstrated by theoretical and/or practical proof that the engine to be qualified is airworthy, i.e. it complies to the Design and Test Specification. (" complying with the regulations prescribed by the competent authority "). The proof of compliance is to be provided by the manufacturer. The Airworthiness Authority Representative is entitled to supervise the conduct of these proofs. BWB-ML is not equipped with facilities which would allow to perform tests within its own responsibility but can make use of other official establishments like the Erprobungsstellen der Bundeswehr = Federal Armed Forces Test Centres, i.e. in case of aeronautical equipment E 61 Manching.

MODEL SPECIFICATION

The airworthiness requirements are included in the Design and Test Specifications. In the Federal Republic of Germany the civil sector prescribes FAR Part 33 as regulation for the certification of an engine and depending on the engine intallation FAR Part 23, 25, 27 or 29. The military sector issues its own national Airworthiness Regulations, the Luftfahrttauglichkeitsvorschriften (LTV). For the qualification of military engines no regulations have been issued so far. In the meantime BWB-ML uses the U.S. Military Specifications

- MIL-E-5oo7 D and
- MIL-E-8593 A

as the basis for elaborating the Model Specification of the engine to be developed. To a certain extent utilization of the U.S. Military Specifications is due to history. The important weapon systems introduced in the Federal Republic of Germany after World War II, such as F-1o4, have been equipped with engines built and qualified to U.S. Military Specifications. Further utilization of these specifications is preferred not only for practical reasons but also for continuity and treatment of engines to the same basic rules.

It has to be clearly pointed out that the Military Specifications referred to will be only the basis for the regulation which define the requirements with respect to airworthiness of the engine.

It is said as in other countries in the Federal Republic of Germany a Model Specification will be established. The difference is this Specification gets here more attention than the basic regulations like FAR, JAR etc. One can explain this point of view by the special task of BWB-ML namely to make sure that an engine taken into the inventory of the Armed Forces is airworthy and stays airworthy throughout the whole In-Service Phase. This handling enables BWB-ML to take into consideration divers points in more detail compared to some civil or earlier U.S. Military Specifications. Moreover this proceeding is more flexible with respect to international cooperation or joint military and civil development of an engine which forces the adaption of an other specification system. For the inclusion of most modern engine technologies, not yet covered by regulations, this is the only possibility.

Some special features BWB-ML looks after in more detail are:

- the correlation between the engine and the engine installation in the propulsion system of the relevant aircraft
- the planned mission and mission profile
- the maintenance concept required by the user

For better understanding the following more extended comments will be made:

Correlation

In former days (and partly still today for civil applications) engines of specified performance categories were developed separately from the aircraft. The aircraft engineer had to comply with the installation instructions laid down by the engine manufacturer to ensure that the engine functioned in a satisfactory and safe way. This made possible a distinct separation into engine and engine installation both at the manufacturer's level and on the side of the Airworthiness Authorities and has been embodied in the Design and Test Specifications, for instance:

Engine Regulation	Engine Installation Regulation
FAR Part 33	FAR Part 23, 25, 27, 29
MIL-E-5oo7 D, -8593 A	MIL-I-83294
DERD 23oo	AvP 97o

Today the integration of the engine into the propulsion system of the aircraft has advanced so far that a close cooperation between the aircraft manufacturer and the engine

manufacturer is indispensable. This design concept now most commonly used has also to be taken into consideration by the Airworthiness Authority when preparing the qualification programme. This is illustrated by means of an example: The engine of the Tornado Weapon System is supplied with the required amount of fuel at the required pressure from the low pressure fuel system. This pressure is produced by an engine-speed dependent pump which belongs to the low pressure fuel system and thus to the aircraft part for which the design responsibility lies with the aircraft manufacturer. Each change in fuel flow required for a given engine speed will result in a change of fuel pressure at the engine inlet. But there are other factors which increase complexity:

- There is the requirement for utilization of different fuels (F 34, F4o)
- The fuel is used as a coolant
- Compressor Bleed Air is used for fuel tank pressurization or as medium for an ejector pump of a fuel/air cooling system in the heat exchanger.

In the end a great number of parameters are interconnected and each change of one parameter can cause a change of one or more parameters or all of them. This interrelation can only be controlled by incorporation of a set of data as detailed and comprohensive as possible into the Model Specification or a related document such as an Interface Control Document or by defining a computer programme established for that purpose. At BWB-ML the organization of the various branches has been adjusted to the problem outlined above by integration of the Airworthiness Authority Representatives responsible for engine qualification with those responsible for qualification of the low pressure fuel system in the BWB-ML section Propulsion System.

Mission Aspects

It is common practice that airworthiness of an engine will be verified to a general specification. From this it could be concluded that the design will not be based upon the factual requirements of the missions but upon those necessary for satisfying those specification. BWB-ML cooperates closely with the user from the very beginning to take account of reality. For instance, the planned mission profile is taken into consideration when designing the engine and demonstrating compliance. Thus BWB-ML is prepared to adapt the schedule of the endurance run established in the specification to the requirements of the mission, namely to relate it to a mission cycle. This is of special importance for engines with a high power to weight ratio.

Maintenance Aspects

During the In-Service Phase of the engine an essential cost factor is the expenditure for maintenance. It is the intention of each user to keep it as low as possible. In this phase however besides the aspects of operational readiness which falls within the user's responsibility the concerns of airworthiness have to be taken into account by BWB-ML. This is contrary to the procedures in other countries. The maintenance concept, if known, can be taken into account when determining the limits defined by airworthiness. For instance, if it is possible to count the load cycles of the critical parts with a recorder the number of the allowable LCF cycles can be established as life limit; where this is not possible a cycle to hour ratio has to be determined in the course of qualification in order to establish the number of hours as the limit for the life of the critical parts.

Cooperation Aspects

Today engine developments even of relatively small engines are hardly or not at all possible in a solo attempt. Such developments require international cooperation to reduce development riscs and also to open a larger market. The latter will also be achieved by certifying a newly developed engine for both the civil and the military market. BWB-ML is in a position to cooperate in such projects as long as it is possible to meet the requirements of the LuftVG. As an example, reference is made to the MTM 38o engine presently in the planning stage. In this project France and the Federal Republic of Germany are cooperating on the military side; however, in addition, this engine is intended for commercial application too. The set up of the Model Specification is to be such that the JAR Engine (that is basically BCAR Section C) serves as the basis. Where requirements of the Military Specification MIL-E-8593 A are not satisfied by BCAR Section C they will be specified as additional subjects in the Model Specification. In addition, installation requirements as per FAR Part 29 or special regulations concerning crashworthiness (MIL-STD-129o) have to be met.

TYPE CERTIFICATION

In the Federal Republic of Germany the Type Certification of a civil as well as of a military engine is documented by issuing a certificate. This document which is set-up in a way similar to the Type Test Certificate used in Great Britain certifies the Airworthiness of the engine. Furthermore it contains the limitations within which the engine is airworthy. Unlike the above mentioned UK Type Test Certificate this certificate does also specify the operation- and maintenance manuals. In addition it contains the requirements for production acceptance testing of new and reconditioned engines.

In the Federal Republic of Germany BWB-ML certifies airworthiness of an engine at the end of the development phase. This does not mean the conclusion of BWB-ML involvment in the life cycle of an engine but only the achievement of a milestone. During the In-Service Phase responsibility of airworthiness still rests with BWB-ML and hence the Type Certificate will be amended continually according to further evidence plus experience by BWB-ML.

CONCLUSION

The process of the qualification and type certification procedure for military engines and the administrative responsibility is regulated in the Federal Republic of Germany in accordance with the Luftverkehrsgesetz = Air Traffic Law. For special features due to the military aspects an excepional provision has been granted. A rough comparison with the procedures applied in other countries shows that our procedure conforms rather with the civil regulations introduced in these countries. In comparison to this civil practice BWB-ML is not only the Certification Authority but has also the function of an Airworthiness Department of a commercial airline or an aircraft manufacturer.

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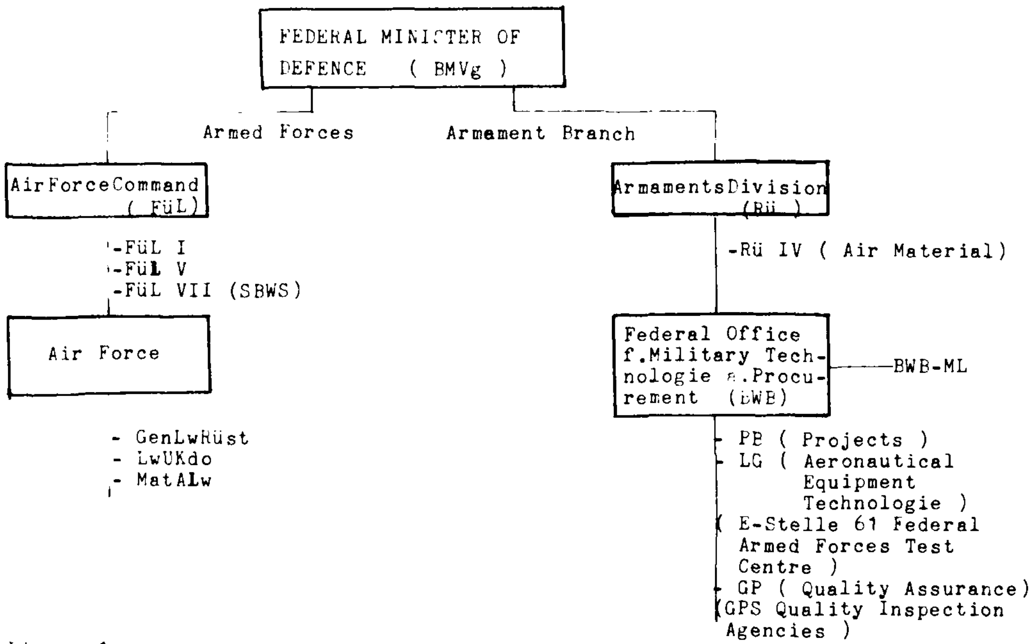


Figure 1

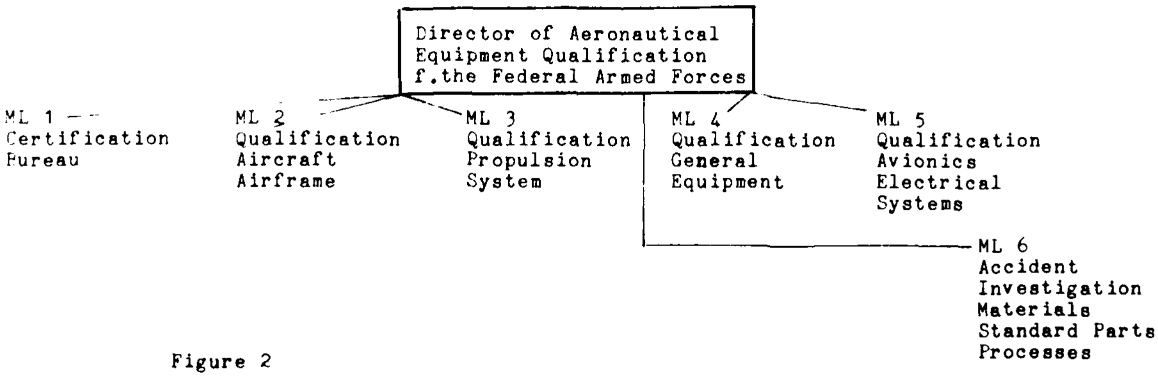


Figure 2

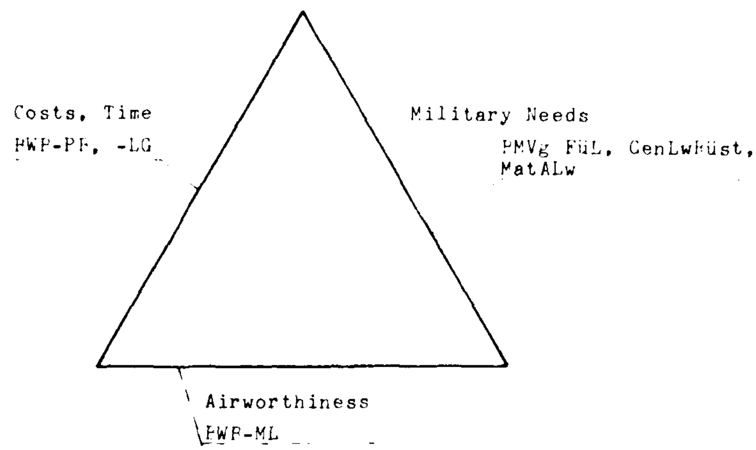
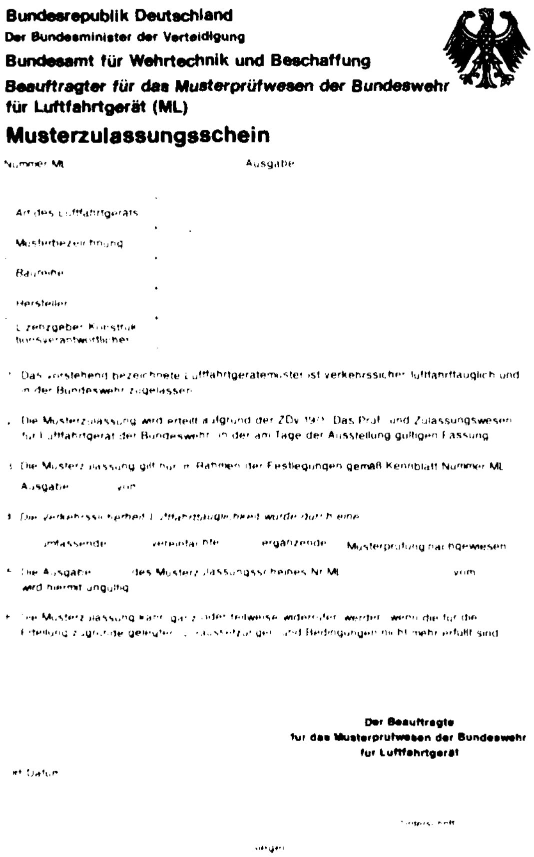


Figure 3



Bundesrepublik Deutschland
Der Bundesminister der Verteidigung
Bundesamt für Wehrtechnik und Beschaffung
Beauftragter für das Musterprüfwesen der Bundeswehr für Luftfahrtgerät (ML)

Musterzulassungsschein

Nummer ML Ausgabe

Art des Luftfahrtgeräts
Musterbezeichnung
Baureihe
Hersteller
Lizenzgeber Konstruktionsverantwortlicher

1 Das vorstehend bezeichnete Luftfahrtgeratemuster ist verkehrssicher lufttahrttauglich und in der Bundeswehr zugelassen

2 Die Musterzulassung wird erteilt aufgrund der ZDv 19/1 Das Prüf- und Zulassungswesen für Luftfahrtgerät der Bundeswehr in der am Tage der Ausstellung gültigen Fassung

3 Die Musterzulassung gilt nur im Rahmen der Festlegungen gemäß Kennblatt Nummer ML Ausgabe vom

4 Die Verkehrssicherheit Lufttahrttauglichkeit wurde durch eine umfassende vereinfachte ergänzende Musterprüfung nachgewiesen

5 Die Ausgabe des Musterzulassungsscheines Nr ML vom wird hiermit ungültig

6 Die Musterzulassung kann ganz oder teilweise widerrufen werden wenn die für die Erteilung zugrunde gelegten Voraussetzungen und Bedingungen nicht mehr erfüllt sind

Der Beauftragte für das Musterprüfwesen der Bundeswehr für Luftfahrtgerät

Ort Datum

Unterschrift

Siegel

Figure 4: Type Test Certificate

Bundesrepublik Deutschland
Der Bundesminister der Verteidigung
Bundesamt für Wehrtechnik und Beschaffung
Beauftragter für das Musterprüfwesen der Bundeswehr für Luftfahrtgerät (ML)

Kennblatt

Nummer ML Ausgabe vom Seite von Seiten

Präambel

Die Musterzulassung gilt nur im Rahmen der in diesem Kennblatt aufgeführten Festlegungen Änderungen dieser Festlegungen bedürfen einer ergänzenden Musterprüfung durch den Beauftragten (ML)
Alle nicht in diesem Kennblatt enthaltenen Festlegungen zum Muster sind den Technischen Vorschriften für Betrieb und Materialerhaltung in der jeweils gültigen Ausgabe zu entnehmen
Die Festlegungen dieses Kennblattes sind verbindlich Bei etwaigen Unterschieden zwischen Kennblatt und Technischen Vorschriften für Betrieb und Materialerhaltung sind letztere entsprechend zu berichtigen bzw zu ergänzen
Besondere Vorfälle die die Verkehrssicherheit Lufttahrttauglichkeit des Musters beeinträchtigen können sind umgehend dem Beauftragten (ML) anzuzeigen
Dieses Kennblatt ist Bestandteil der Musterzulassung Nr ML
Ausgabe vom
Vorhergehende Ausgaben dieses Kennblattes werden hiermit ungültig

I Grundmuster

1 Allgemeines

Art des Luftfahrtgeräts
Musterbezeichnung
Versorgungs Nr
Baureihe
Hersteller Name und Anschrift
Lizenzgeber Konstruktionsverantwortlicher
Betreuungsfirma

Figure 5: Data Sheet, Attachement to the Type Test Certificate (Cover Sheet only)

SPECIFICATION AND REQUIREMENTS RATIONALE FOR MILITARY AND CIVIL HELICOPTER ENGINES.

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SUMMARY

The paper reviews the endurance and supplementary qualification testing required for the granting of Type Approval of military helicopter engines. The rationale of these is given, and variations in different nations' requirements are discussed. A comparison is made with civil certification requirements. Attention is drawn to differences between the requirements for helicopter and fixed-wing aircraft engines. The paper illustrates how service experience has revealed certain deficiencies in the test requirements, and has led to the development of more realistic procedures. The options being considered for the current review of the UK qualification requirements are discussed.

1. INTRODUCTION

There are in the NATO countries several general specifications for military engines and codes of civil airworthiness regulations. All of them deal inter alia with helicopter engines. These sets of requirements are similar in their general philosophy but differ considerably in detail. Whilst some of the differences arise from genuine variations in operational requirements, others are simply results of the separate evolution of the specifications, and a lack of successful effort at standardization. To achieve Type Approval in all the markets in which he hopes to sell his engines, an engine manufacturer would obviously prefer to carry out a single series of tests, rather than duplicate tests to meet small differences in requirements. It is equally in the interests of military authorities, who not only set the requirements but have to pay for them to be met, to avoid unnecessary duplication of testing when purchasing foreign-made engines. The UK military specifications have not been amended for some years, and the time has come for a thorough review. It is therefore opportune to look not only at our own requirements but also at others to see what can be gained from those in force elsewhere, both military and civil. We must examine whether these requirements have adequately met the needs of the past, and what changes are necessary to meet the needs of the future. This paper sets out to show the path we are taking.

2. TYPE APPROVAL TESTING REQUIREMENTS

2.1 General and Individual Engine Specifications. Airworthiness regulations and general military specifications set out the mandatory requirements which normally must be met by a new engine. The civil requirements cover factors affecting the safety and durability of the engine, but the military specifications deal also with requirements enabling the engine to operate satisfactorily in a military role. For a military engine an individual engine specification is prepared which gives the performance requirements. It will also detail the way in which the mandatory testing will be applied, and any exemptions granted, following discussions between the contractor and the authorities concerned. Thus the general specifications are administered with a degree of flexibility sufficient to meet the needs of a particular engine. The main general specifications and airworthiness codes are shown in Table 1.

2.2 Brief Description of the General Specifications and Airworthiness Codes

2.2.1 D Eng RD Specifications. D Eng RD 2100 covers Test Requirements and D Eng RD 2300 covers Design and Construction. They are used for all UK service engines and are controlled by the Directorate of Engine Technology in MOD(PE), formerly the Directorate of Engine Research and Development, from which the specifications took their name. Separate Type Test Schedules and Acceptance Test Schedules are provided for engines of Single-engined Rotorcraft, Multi-Engined Rotorcraft and for coupled engines, as well as for various categories of fixed-wing aircraft engines. D Eng RD 2100 was last comprehensively revised in 1967, but a number of amendments were issued up to 1975 to keep it up to date.

2.2.2 British Civil Airworthiness Requirements Section C (BCAR). BCAR Section C, dealing with engines and propellers, is accepted as a European Standard by the Airworthiness Authorities of Belgium, France, Federal Republic of Germany, Italy, The Netherlands, the United Kingdom and Sweden. The requirements are controlled by the Engine Requirements Co-ordinating Committee of the UK Civil Aviation Authority. All participating nations are represented on the Committee. BCAR Section C covers both Design and Construction, Type Tests and Acceptance Tests. It is reviewed annually, and amendments are made between revisions by means of 'Blue Papers' giving additional or revised requirements. Separate sections are devoted to engines of single and multi-engined Rotorcraft, as well as engines of fixed-wing aircraft. The Type Test schedules for Rotorcraft engines have recently been amended to facilitate cross-validation with the US FAR Part 33. Previously they were identical to those in D Eng RD 2100.

2.2.3 MIL-E-8593A. This is the US general military specification for turbo-prop and turbo-shaft engines. It was prepared by the US Navy, and is similar to MIL-E-5007D, the specification for turbo-jet and turbo-fan engines. It is subscribed to by all 3 US Services. Like the European specifications, it covers both Design and Construction (Requirements) and Preliminary Flight Rating, Qualification and Acceptance Tests (Quality Assurance Provisions). The specification was issued in 1975, superseding a number of earlier specifications dating from 1964. The UK is not aware of an imminent revision to this specification.

2.6 FAR Part 33. Part 33 of the United States Federal Aviation Regulation is entitled 'Airworthiness Standards Aircraft Engines' and covers civil engines. It consists of a general sub part dealing inter alia with engine ratings and limits, a sub-part dealing with general design and construction requirements, sub-parts dealing with design and construction of reciprocating and turbine engines, and with testing of reciprocating and turbine engines. Separate endurance test schedules are provided for helicopter engines having various rating structures.

2.7 The Role of Testing. As the theme of the Symposium is Turbine Engine Testing, this paper concentrates on testing requirements; on how the testing validates the design requirements, rather than how the severity of the design requirement is arrived at. From the earliest stages of an engine's evolution, the manufacturer will carry out extensive testing, starting with component rig tests. Formal testing also starts at an early stage and continues throughout an engine's development and service life. As soon as practicable in its development an engine should be flight tested to evaluate its performance at altitude during flight manoeuvres. A formal test is required to assure the integrity of development engines during this programme. As development approaches completion, a programme of Type Approval tests is undertaken to ensure the fitness of the engine for service use, and to establish the performance limits within which it may be safely operated. To ensure that production engines conform to the Type Approval standard, all are given an Acceptance Test to check their performance and correct functioning and, in the UK, sample engines are subjected to a more searching "Production Quality" Test. On engines in service, checks must be undertaken to ensure that undue performance deterioriation has not occured. Formal test programmes are also required to accept repaired engines for continued use and to validate modifications. Depending on the significance and the extent of the modification, the latter may vary from a short check up to a repetition of much of the Type Approval testing. The structure of these tests is shown in Table 2. The remainder of this paper deals with Type Approval Tests, as these are the most comprehensive tests to which an engine is subjected, and set the standard against which subsequent testing is measured.

3. TYPE APPROVAL TESTS (QUALIFICATION TESTS)

3.1 The Common Elements. The four codes of requirements mentioned in Para 2.1 have 3 features in common.

a. Conditions according to which the satisfactory completion of the tests will lead to the approval of the engine.

b. Endurance Tests.

c. Supplementary Tests.

3.2 Conditions for Type Approval. It is a feature of all the requirements that satisfactory completion of the formal tests is a necessary but not a sufficient condition for approval. A fault which may be serious in service operation may occur only a very few times in the limited hours of development testing, and possibly not at all during approval tests. It is essential therefore that an authority bases its approval on its total knowledge of an engine, not just on the outcome of the formal tests. This knowledge should include all development testing of an engine and possibly service experience of earlier versions of the engine. D Eng RD 2100 therefore states that in addition to meeting the Type Test requirements, an engine must have "a satisfactory background of development experience". BCAR includes like requirements in Section A, which deals with Certification and Approval Procedures. MIL-E-8593A adopts a somewhat different approach. It acknowledges that the formal tests are only spot checks on the performance and integrity of the engine. It therefore states that "notwithstanding the requirements for test verification of individual points of performance or operating characteristics ...., the engine manufacturer/constructor shall continue to be fully responsible for all features, characteristics and performance of the engine throughout the environmental conditions and operating envelope, to the extent required by the applicable contract."

3.3 Relation between Endurance and Supplementary Tests. The most significant stage in the approval testing of an engine is the endurance test. The build standard of the engine which passes this test is the standard for which approval is given for service use. The endurance test is not able to assure certain aspects of engine functioning and integrity, so supplementary tests are required. These tests also must be carried out on engines representative of the production standard. Should problems arise during these tests, modifications to the engine may be necessary. It is therefore desirable for the supplementary tests to be carried out before the endurance test, so that any modifications can be incorporated if possible in the endurance test engine, so avoiding the need for clearance of the modifications in subsequent modification approval tests. In practice this ideal is not always achieved, as modifications take time to develop and procure. Following an unsuccessful supplementary test, a repeat of both the supplementary and endurance tests may be required to ensure the integrity of the modification. The repeated endurance running does not always incur extra expense, as a modification can often be "given a ride" on an engine being endurance tested for some other purpose, e.g. to clear a different type of oil.

3.4 The Endurance Test

3.4.1 Description of the Test. When an engine has been developed to the stage where its performance and functioning are satisfactory, and its integrity has been established by supplementary testing, its durability at a determined level of performance is established by endurance testing. The various codes of regulations all use for this purpose variations on the theme of the 150 hour test, divided into 25 6-hour stages.

Table 3 compares the endurance test schedules for helicopter engines required by the four codes of regulations by illustrating the running hours required at each rating during the 150 hour period. The various instructions for carrying out an endurance test are complex, and it is impossible to represent them all on one diagram.

The total running time at each power rating is only one measure of severity. A further measure is the cyclic usage to which an engine is subjected. This is illustrated in Figures 1 which represent one 6-hour stage of the four schedules for multi-engine helicopters marked with an asterisk in Table 3. The severity of the cycle is influenced by the times taken for power lever movement, which are also defined in the specifications. Normally all power lever movements are rapid.

An additional measure of severity is imposed in both the UK and US military specifications by the use of temperature margins. D Eng RD 2100 requires all running at ratings other than contingency ratings to be at a turbine inlet temperature (at stator outlet) 30K above the operating limit to be cleared by the test. A lower margin, negotiable between the contractor and the authority, is applied to contingency ratings. In practice lower margins have been used until now. All engines tested since the 30K margin was introduced in 1975 have been derivatives of engines whose Type Test requirements were fixed before that date. Likewise MIL-E-8593A requires a margin of at least 30K at all ratings, there being no contingency ratings in normal US practice. A further margin is added to cover system inaccuracy limits.

Returning to Figures 1-4, it is instructive to compare the schedules of the different authorities. All have the same principal elements; the 25 6-hour stages each consisting of long periods at the "normal use ratings", cyclic periods between low and high ratings, and incremental running. The differences between the schedules arise from the varying mix of these kinds of running, which result in the varying total running times at each rating shown in Table 3. The differences between the D Eng RD 2100, BCAR and FAR schedules are comparatively minor, but between these three and MIL-E-8593A the difference is large. This is a direct reflection of the lack of contingency ratings in US Rotorcraft engine practice. The nature of a Type Test is determined to a considerable extent by the rating structure of the engine, and this subject is dealt with in detail in Para 4.1 below. The BCAR schedules were previously identical with those in D Eng RD 2100 but excluded the temperature margin requirement. The new BCAR schedules are very nearly identical to the FAR rules, so that an engine tested to them will qualify for FAA approval. The change was made because standardization with US civil rules was thought more advantageous than with UK military requirements. Technically the change was of little consequence, the FAA test being of similar severity to the D Eng RD 2100 schedule, when the effect of temperature margin is discounted.

The regulations also require a minimum number of starts to be carried out during and/or after the endurance run, as shown in Table 4. BCAR Section C requires in addition that where applicable all normal starts be made with the free power turbine locked to simulate operation of the engine in the helicopter with the rotor system locked. There is a similar requirement in D Eng RD 2100 for running at ground idle or during starts with the free power turbine locked.

Although all four sets of regulations include a basic endurance test of 150 hours, MIL-E-8593A is unique in requiring four such tests for full approval. The four tests are run as two 150 hour segments on each of two engines, each engine using a different type of fuel and oil. (e.g. MIL-L-7808 oil and JP-4 fuel in one engine, MIL-L-23699 oil and JP-5 fuel in the other). The specification gives an option of running only one 150 hour segment on each engine, an option which has certainly been exercised on occasion. D Eng RD 2100 has a normal requirement for a single 150 hour test, but an option for a double 150 hour segment, which has never been used. All specifications require further testing for clearance of additional fuels and oils.

As important as the requirements for running the test are the criteria by which an engine is judged to pass or fail, but these are specified with much less precision. The criterion is usually that the engine must be "satisfactory" and must not show "excessive" wear or "undue" deterioration. MIL-E-8593A at least goes as far as to define "satisfactory". It is the intention of D Eng RD 2100, although not stated explicitly, that an engine which has just completed a 150 hour test should be capable of repeating it. It was the thought that this capability should be demonstrated that led to the option for a double 150 hour test.

3.4.2 Evolution of the Test. In the early days of helicopter turbine engines, the Type Test schedules in D Eng RD 2100 were similar to those for engines of fixed-wing aircraft, which in turn had evolved from piston engine tests. Service problems of unexpectedly low cyclic lives and vibration soon indicated that the traditional test method of running an engine for extended periods at high powers was not adequate. The extended periods also led to practical difficulties in running the test. This led to a revision of the schedule at the beginning of the 1960's to the present pattern of 25 6-hour stages, with stops being permitted between stages. The content of the stages was arranged so that much more cyclic usage was accumulated during the 150 hours, and periods of incremental running were introduced to identify vibration problems. It is a feature of all regulations that the incremental power levels must be adjusted if vibrations are detected so as to run the engine deliberately at the speeds where vibration occurs. More recently still the turbine temperature margin noted above was introduced to give a greater assurance of engine life.

3.4.3 Rationale of the Test. At the present state of evolution the rationale of the Type Test is that it demonstrates that the engine is capable of being operated safely at each of the declared ratings for an adequate period, which will be the initial overhaul life. The life of the engine is of course capable of extension by further engine and component testing and by examination of engines in service. The demonstration is effected by running the engine for a sufficient number of periods at each rating, each at least as long as the time for which the rating is to be approved. Insufficient cyclic work is at present included to demonstrate the safe cyclic life of the engine for the initial period, so supplementary component LCF tests

must also be carried out. For engines which are to be used for short duration sorties, D Eng RD 2100 requires a supplementary test consisting of additional accelerations and decelerations. The total cyclic usage in the 150 hour test and the supplementary test must be at least equal to the predicted usage in the initial overhaul life of the engine. Incremental running is done to identify potential weaknesses at intermediate power levels.

The comparatively slow evolution of the test schedule has been caused in part by the desire to use the Type Test as a yardstick by which to compare an engine with its earlier versions and with previous engines. Partly it has been due to the desire not to fall too far out of line with other authorities and so hinder mutual recognition of approvals.

It has to be admitted that after engines enter service they meet problems which the Type Test failed to reveal. It is obvious that the 150 hour bench test does not reproduce in every respect either the way an engine is operated or the environment it meets when installed in a helicopter. A service engine is given many more power lever movements imposing minor stress cycles, not simulated in the Type Test; it is subjected to a severe vibration environment by the helicopter airframe and rotor system; the inlet air flow is distorted; the engine structure is subject to deflections caused by flight manoeuvres. A further factor is that the Type Test is not necessarily carried out on a "worst" engine as regards resonant frequencies, mechanical tolerances etc. Attempts have been made to overcome these deficiencies, both by trying to simulate more accurately the aircraft environment during the endurance test, and by carrying out further supplementary tests. As an example of the former, the GE T700 was type tested in the United States on a test bed mounted on a vibrating platform to simulate the helicopter environment. The US have also carried out a Flight Qualification Test (Preliminary Flight Rating Test) with an engine built to maximum permitted imbalance. Accelerated Mission Testing and Simulated Mission Endurance Testing have been carried out to try to subject the engine to operation more like that which it will see in service. It would be interesting to know how successful these tests have been in preventing problems from occurring in service.

3.5 Supplementary Tests

3.5.1 The Purpose of the Tests. The title of this section, Supplementary Tests, was chosen as it is the term used in D Eng RD 2100 to describe the tests other than the endurance test required for Type Approval. The word "supplementary" is deceptive. It implies that these tests are in some sense less important than the 150 hour endurance test. This is not so. The supplementary tests are an integral and essential part of the Type Test. They may be classified under 3 broad headings:

a. Demonstration of correct functioning and achievement of performance targets at all points in the flight envelope.

b. Demonstration of integrity and safety margins.

c. Demonstration that the engine can survive the environment and hazards it is likely to meet and will not itself cause unwanted environmental effects.

3.5.2 Description of the Tests. Table 5 shows the principal supplementary tests and indicates which of them are invoked by each of the main codes of regulations. All of these tests are common to both fixed-wing and helicopter engines, but the condition under which the test is run, e.g. the flight envelope, is set to meet the requirements of the particular aeroplane or helicopter. An approving authority would normally grant an exemption where a test is not appropriate for a particular engine. Different terminology sometimes occurs in different specifications for similar requirements. Table 5 does not follow any particular specification but uses descriptions which most clearly indicate the nature of the requirement. A further problem in making comparisons is that one test sometimes covers more than one requirement. For example, over-temperature and over-speed demonstrations must be run consecutively on the same engine to comply with MIL-E-8593A but the other rules allow them to be run separately. Another difficulty is that checks required as a separate test by one specification may need to be carried out during the endurance run by another (e.g. assessment of the effect of air bleed on performance). Certain requirements in FAR Part 33 are found in 'Design and Construction' for which specific tests are laid down in the other Codes; the test requirement is therefore implicit, in that the engine manufacturer must show the authority he has met the design requirement, rather than explicit. Space unfortunately does not permit a complete listing of the requirements for component and accessory tests.

An indication in Table 5 that a topic is dealt with by more than one set of rules does not indicate that they all require identical tests, although in many cases they do. In particular a number of requirements in D Eng RD 2100 and BCAR Section C are identical. Over the years attempts have been made to standardize certain of the supplementary tests in D Eng RD 2100 with those of MIL-E-8593A and MIL-E-5007D, through the Propulsion Systems Working Party of the Air Standardization Co-ordinating Committee (ASCC) on which UK, USA, Canada, Australia and New Zealand are represented. This effort is slowly producing useful results, and a more determined effort on the part of the US and UK, the two major engine building nations in ASCC, would enable real progress to be made. Unfortunately there is no equivalent NATO counterpart for this Working Party as there are for some of the other ASCC groups.

The list of supplementary tests shows the differences in the roles of the civil and military authorities. The civil requirements are limited to those needed to ensure integrity and airworthiness, whereas the military specifications include other tests needed to enable an engine to operate satisfactorily in a military environment, e.g. salt water and sand ingestion tests. FAR Part 33 does not deal with such matters as pollution, noise and smoke, as they do not affect the airworthiness of the engine. They are regulated by other US rules or agencies, but not of course with the interests of the aircraft operator in mind. MIL-E-8593A includes a number of topics not dealt with in D Eng RD 2100, to which the UK will give attention during

[illegible] that, in the [illegible]

[illegible] Design Requirements [illegible] specification [illegible] BAR [illegible] engines.

[illegible] fixed-wing [illegible]

In BCAR [illegible] tests are specified for helicopter [illegible] aircraft engines [illegible] fixed-wing aircraft.

[illegible] At the time of writing [illegible] their General Specification [illegible] part of this review all aspects of testing will be examined, but it is known already that certain supplementary tests will require amendment.

[illegible] Lifing Procedure for Critical Parts. A new procedure has recently been [illegible] for determining, substantiating and extending the low cycle fatigue lives of critical parts (i.e. parts such as discs and shafts, the failure of which would hazard the aircraft). The procedure defines the method of life prediction with more precision than at present, but still leaves the contractor with a degree of flexibility [illegible] a method suited to his design. It is envisaged that the principal means of determining initial safe lives will be by spin-pit testing rather than by testing of the complete engine or by relying on an analytical means alone. The life derived from the test will take into account the scatter in material fatigue properties, and the initial declared life will be half the predicted safe life. Extension of the full safe life will depend on further testing and inspection of ex-service components. [illegible] to be adopted in [illegible] to BCAR Section [illegible] MIL-E-[illegible]A. The US military requirement for low cycle fatigue life prediction is [illegible] twice the required number of cycles, and a complete engine test up to the required number. It is the UK view that component testing well beyond twice the life to be declared is required if adequate allowance is to be made for material scatter.

[illegible] Corrosion Susceptibility Test (Salt Water Ingestion). [illegible] experience has revealed corrosion problems in engines which were not evident [illegible] susceptibility tests at present prescribed by [illegible] is therefore considering adopting the much more severe test in MIL-E-[illegible]A for engines which will operate in severe marine environments, such as those fitted to naval [illegible]. The existing requirements would remain for engines which operate in less severe conditions. A test is to be carried out [illegible] "Gem" engine to the MIL-E-[illegible]A schedule, modified slightly to take account of the different rating structure of the Gem. The MIL-E-[illegible]A test is long and expensive; [illegible] balanced against the savings made by detecting weaknesses sufficiently early to avoid the need to modify engines in service. A comparison of the current [illegible] requirements and those in MIL-E-[illegible]A is given in Table [illegible].

[illegible] Sand Ingestion Test. Both [illegible] and MIL-E-[illegible]A require sand ingestion tests, each specification requiring approximately [illegible] hours of running at maximum continuous power. Whereas MIL-E-859[illegible]A states that there must be not more than [illegible] in power or increase in [illegible], [illegible] merely states that should these limits be exceeded before the end of the test, the test may be terminated. MIL-E-[illegible]A states that "the test will be considered to be satisfactorily completed when the performance criteria .... have been met and teardown inspection reveals no failure or evidence of impending failure". [illegible] requires only that the engine be completely stripped and examined, and a report submitted. There is thus no pass or fail criterion, as there is clearly in the US specification, an omission which requires correction.

[illegible] Air Bleed Contamination. At the most recent [illegible] Propulsion Systems Working Party meeting, the Nations agreed to standardise on the contamination limits given in MIL-E-[illegible]A. This makes no difference to [illegible] as far as many contaminants are concerned, but the limits for [illegible] breakdown products are different. Current specifications are based on the assumption that the contaminants exist in gaseous form. However there is evidence that contaminants in the form of aerosols (tiny droplets) can be particularly irritating; this is a problem which requires further investigation to establish acceptable limits and satisfactory test procedures.

3.6 New requirements in D Eng RD 2100. Certain tests are included in MIL-E-8593A which are not found in D Eng RD 2100. The UK will be considering whether to include some of these requirements, and if so whether the US method of test can be adopted directly in the interests of standardization. New requirements which are likely to be dealt with in D Eng RD 2100 and D Eng RD 2300 are Electro-Magnetic Compatibility (EMC), Infra-Red Radiation, Nuclear Hardening and possibly Exhaust Gas Emissions and Noise.

3.7 Assessment of Test Results. The problem of objective judgement in assessing test results is perhaps more difficult with supplementary tests than with the endurance test. As in the case of the UK Sand Erosion Test noted above, a number of test requirements give no indication of an acceptance criterion, and indeed may be regarded more as "demonstrations for information" rather than pass/fail tests. Nevertheless some common and objective standard of judgement is desirable particularly when one nation needs to validate the approval of an engine granted by another. A useful view of other nations' approaches can be gained from multi-national qualification groups, such as that set up by the UK, Federal Republic of Germany, and Italy to approve the RB 199.

4. OPTIONS FOR THE FUTURE

4.1 Providing for New Rating Structures. Each of the 150 hour endurance test schedules shown in Table 3 assumes a particular rating structure for the engine. As already noted, the MIL-E-8593A schedule diverges from those of the other 3 authorities because it makes no allowance for contingency ratings for engines used in multi-engined helicopters. In this respect at least UK military requirements are unlikely to fall in line with those of the US. The traditional UK rating structure for such engines is nevertheless being called into question. Operational needs in future will demand greater flexibility in the way engines are operated. The use of helicopters for tasks such as sonar dunking will require the repeated and longer use of high ratings, up to take-off power, on certain sorties. The engine operating limits determined by type testing normally allow only for a limited use of each rating per flight. However the advent of time-temperature recorders and low cycle fatigue counters will permit a much more flexible use of power. No longer will it be necessary to limit the usage of each rating in every flight in order to ensure the safe and reliable operation of the engine throughout an overhaul life measured in hours. Instead life could be declared in time-temperature and low cycle fatigue counts, and the service operator would be free to use as much or as little of this life on each flight as he chose. If the time between overhauls is not to be too short, some sort of engine management policy will still be necessary in practice to restrain over-enthusiastic pilots. One role of the Type Test therefore will be to determine the initial life recorder limits up to which an engine may be operated rather than to determine an overhaul life in hours during which an engine may be operated for a given time per flight at each rating.

A further development is the need for emergency or "once only" powers. The contingency ratings used at present in European engines for multi-engined helicopters provide additional power for use when the other engine fails. Maximum Contingency, normally a 2½ minute rating, is used to recover the helicopter from immediate danger, should failure occur during a take-off, landing or hover. Intermediate Contingency may be used for 1 hour (D Eng RD 2100) or an unlimited period (BCAR) in order to return to base. These ratings may be used without restriction, with the proviso that excessive use, for example during training in "engine-out" operation, would require the overhaul life of the engine to be reconsidered. The proposed emergency power would be above the current Maximum Contingency level, and could be used only once or possibly a very few times. The advantage to be gained is that smaller and lighter engines could be used for a given aircraft weight, requiring a higher percentage of Maximum Continuous power to be used during the cruise, providing better SFC and extended range, but at the expense of engine life. As originally proposed, emergency power level was to be available once only; its use would entail the rejection of the engine at the end of the flight. As it would only be used following the failure of the other engine, two engines would need to be changed when the aircraft returned to base. This would be particularly disadvantageous for ship-borne helicopters, as sufficient spare engines would not normally be available. An emergency power level which could be used for, say, 3 times would therefore be more useful. An operational difficulty with this concept is the impracticability of training in its use, as continued use of the emergency power would result in a high rate of engine rejections. It is envisaged that emergency power would be required for a period of approximately 1 minute before a reduction to Maximum Contingency could be made safely. To give an adequate margin of safety, the inclusion in the 150 hour endurance test of 6 1½ minute excursions up to emergency power has been proposed by MOD(PE).

Not only is it true that changes in the rating structure may influence Type Test requirements. It is equally true that the nature of the Type Test can influence the ratings that can be offered. In determining these ratings the engine manufacturer has regard not only for the proposed operational usage of the aircraft, but also for the Type Test requirements that the engine must meet; the more severe the test, the more conservative the ratings. As already seen both the US and UK military authorities require the endurance test to be run at turbine inlet temperatures 30K above the operating limits. If the aim of 50% creep life usage during the test is to be met in both the civil and military cases, obviously the power levels which can be declared from a military test will be those obtainable at a turbine inlet temperature 30K lower than those which can be declared from a civil test. The consequence is that the manufacturer is obliged to offer a conservatively rated engine for military use, but can provide a "hot-rod" version of the same engine for civil use; the reverse of what one considers to be usual. When deciding the temperature margin required in an individual engine specification, careful consideration must be given to the trade-off between power and engine life which is appropriate to the particular application of the engine.

4.2 The Reason for Change. The principal criticism of the current method of Type Approval testing is that it still fails to reveal all the problems that occur in service. Up to a point this is not surprising. By the time Type Approval is granted, experience will have been gained from perhaps 10,000 hours of engine running. Series engines will in the course of time accumulate many hundreds of thousands or, for civil types, millions of service hours. Little wonder then that not every

[illegible] Nevertheless [illegible] this [illegible] cause for complacency, and the [illegible] which latent fault arise early in service experience because [illegible] were not adequately simulated during testing. Two remedies have been adopted [illegible]. Firstly, the 150 hour endurance test has been made more [illegible] of service running, as described in para [illegible]. Secondly, additional cyclic testing [illegible] complex cycles which simulate the sortie pattern [illegible] engine will be operated in service.

[illegible] aircraft sortie pattern can be expensive and time consuming, particularly if the [illegible]. There are obvious advantages therefore in devising an "accelerated" test, [illegible] the time spent at the lower power ratings of Maximum Continuous and below, but retain [illegible], and [illegible] of running at high power. There is much to be said for [illegible] representation of the sortie actually flown. This is particularly true for military engines, [illegible] usage is usually less predictable than of civil engines.

Cyclic testing of the Rolls-Royce "Gem" engine in the Westland "Lynx" helicopter has used both [illegible] cycles and sortie-pattern cycles. The simple cyclic testing revealed a number of problems not [illegible] during the [illegible] hour endurance test, but was not able to predict at what stage in service life [illegible] would become apparent. Sortie pattern testing has used a cycle combining the severest features of the usage to which the engine is subjected by the Royal Navy, the British Army and the French Navy. [illegible] has predicted several problems, after about half the engine hours at which the problem appeared in service. Amongst these problems were failure of the HP nozzle vane support ring, [illegible] of the interstage [illegible] sealing of the centrifugal compressor impeller on its shroud, and the ultimate failure of the HP turbine blades. [illegible] hours of sortie pattern testing and [illegible] cycles have so far been run on the Gem engine and a further extensive programme is planned.

It is accordingly true that sortie-pattern testing has on many occasions given early warning of [illegible] which [illegible] occurred during Type Test. But the 150 hour endurance run has also revealed problems which [illegible] during sortie pattern testing. The fundamental decisions that need to be made are whether to adopt sortie pattern testing as a Type Approval requirement, either in [illegible] the 150 hour test and how, if the 150 hour test is to be retained, its [illegible] improved to make it more realistic of service conditions. There are a number of difficulties and disadvantages in adopting sortie pattern testing as the main Type Approval endurance test. [illegible] full benefit from the test the schedule should follow exactly the cyclic pattern which the engine will experience in service. In order to evaluate properly design changes which are [illegible] during development, endurance running prior to the Type Test should preferably be [illegible] the Type Test schedule. Experience from trials of the UK Engine Usage Monitoring System has [illegible] differences between the actual usage and the usage which had been predicted during development. It is [illegible] therefore that a truly realistic sortie pattern, based on [illegible] data, would be available sufficiently early. Another difficulty is that, to be truly representative of service usage, the sortie pattern testing should not be run using artificially high turbine [illegible] temperatures, as currently demanded for military Type Tests. To achieve sufficient confidence [illegible] using a less severe schedule would require a longer test. Sortie pattern [illegible] after formal Type Approval, [illegible] in UK and US, have run for over 1,[illegible] hours. [illegible] would [illegible] be inevitably lengthy for Type Approval.

There are therefore good reasons for sortie pattern testing, essential as it is as part of the [illegible] development programme, not being part of formal Type Approval. If sortie pattern testing [illegible] Type Approval, there is no reason, as noted in paragraph [illegible] above, why the authority [illegible] the results when considering the extent of approval to be granted.

[illegible]

While there is [illegible] criticism of Type Approval procedures, there is a reluctance to find the time and money to [illegible] more severe tests. The US already possesses in MIL-E-[illegible]A a specification [illegible]. But these requirements are often trimmed [illegible] to meet [illegible], as we understand, to meet the [illegible]. The ideal balance is struck when the cost of carrying out more stringent testing, [illegible] corrected early, is balanced by the savings in life-cycle costs of production [illegible]. It is this balance which the UK will be seeking in drawing up revised requirements for realistic [illegible] testing.

[illegible] suggested that, in an age when most new engine developments are collaborative, a common [illegible] requirement [illegible] be evolved for military engines, as has already happened in the civil field. [illegible] we would welcome it. But before any such agreement could be reached, each nation would need to form a clear idea of its own future requirements. This paper seeks to give an indication, [illegible] preliminary, of the direction which UK thinking is taking, and we hope it will stimulate discussion [illegible] elsewhere.

ACKNOWLEDGEMENT

The author wishes to acknowledge the co-operation of Rolls-Royce Ltd, Aero Division Leavesden, in the preparation of this paper.

TABLE 1

AIRWORTHINESS REQUIREMENTS

UK MILITARY	D Eng RD 2100	General Specification for Testing of Gas Turbine Aero Engines
	D Eng RD 2300	General Specification for Design and Construction of Gas Turbine Aero Engines and Jet Pipes
EUROPEAN CIVIL	BCAR Section C	British Civil Airworthiness Requirements Section C: Engines and Propellers (European Participation)
US MILITARY	MIL-E-8593A	General Specification for Turboshaft and Turboprop Aircraft Engines
US CIVIL	FAR Part 33	Airworthiness Standards, Aircraft Engines

TABLE 2

STAGES OF TESTING

DEVELOPMENT	Flight Qualification Test (Preliminary Flight Rating Test)
	Type Approval Test (Qualification Test)
PRODUCTION	Production Acceptance Test
	Production Quality Test
MODIFICATION	Modification Approval Test
SERVICE USE	Power Performance Index (Deterioration) Check
REPAIR	Overhauled Engine Acceptance

TABLE 3

ENDURANCE TEST SCHEDULES

		Max Contingency	Inter Contingency	Take-off (5 min)	Max 30 min	Max 1 hour	Max Continuous	Incremental	Idle
D Eng RD 2100 **	Single Engine			10h 50m		37h 30m	35h 25m	54h 10m	12h 5m
	Multi-Engine *	1h 27½m	26h 2½m	20h 50m			35h 25m	54h 10m	12h 5m
BCAR	Multi-Engine Unrestricted Inter-Contingency *	2h 5m	25h 0m	16h 40m			20h 0m	62h 30m	23h 45m
	Multi-Engine 30 min Inter Contingency	2h 5m		16h 40m	12h 30m		32h 30m	62h 30m	23h 45m
	Single/Multi Engine No Contingency			18h 45m			45h 0m	62h 30m	23h 45m
FAR Part 33	Multi-Engine with 2½m and 30m ratings *	2h 5m		11h 40m	12h 30m		50h 0m	59h 0m	23h 45m
	Single/Multi Engine No Contingency			18h 45m			45h 0m	62h 30m	23h 45m
MIL-E-8593A**	*			30h 50m	44h 10m		15h 0m	40h 0m	20h 0m

* See also Figs 1-4.
**Additional requirements for coupled engines also included.
All schedules total 150h 0m.

TABLE 4

STARTS

	Starts during 150 hour Endurance Test	Hot Starts	False Starts	Additional Hot or Cold Starts	Total
D Eng RD 2100	25	10	10	55	100
BCAR Section C	25	10	10	55	100
FAR Part 33	25	10	10	55	100
MIL-E-8593A	150	10	10	150*	320

*NOTE: Each additional start to be followed by slam acceleration to Maximum power, remain at Maximum for 30 sec, followed by immediate shutdown. Time between starts for 18 of these additional starts to increase incrementally by 5 mins, ie 5, 10, 15 ...... 90 minutes.

TABLE 5

SUPPLEMENTARY TESTS

5.1 PERFORMANCE

	D Eng RD 2100	MIL-E-8593A	BCAR	FAR Part 33
Low Temperature Starting	X	X	X	X
High Temperature Starting	X	X		X
Altitude Performance and Functioning		X		X
High Temperature Performance and Functioning	X	X		X
Altitude Relight Envelope		X	X	
Acceleration Response Times	X	X		X
Surge Tests (Hot Reslams)	X			
Bleed Air - Effect on Performance		X	X	X
Power Offtakes - Effect on Performance		X		X
Fuel Pressure Tests	X	X	X	
Fuel Temperature Tests	X	X	X	
Alternate Fuels	X	X		
Emergency Fuels	X	X		
Inlet Pressure Recovery and Distortion - Effect on Performance		X		
Altitude Windmilling Test		X		
Starting Torque Test		X		

5. MECHANICAL INTEGRITY

X Normally Mandatory   0 Optional

	D Eng RD 2100	MIL-E-8593A	BCAR	FAR Part 33
Low Cycle Fatigue of Critical Parts	X	X	X	X
Other Fatigue Tests	X	X	X	
Disc Crack Propagation and Burst Tests	0	X	0	0
Vibration Survey	X	X	X	X
Rotor Integrity (Overspeed following failure)	X	X	X	X
Operation at Max Overspeed Limit	X		X	
Overtorque Test	X		X	
Overtemperature Test	X	X	X	X
Blade Containment	X	X	X	X
Acceleration Cycles	0		0	0
Sortie Pattern Tests	0			X
Governor Checks	X		X	
Top Temperature Limiter Check	X		X	
Fuel Contamination	X	X	X	X
Windmilling without Oil Supply	X	X	X	X
Accessory Drive Tests	X	X		
Free Turbine Overspeed Trip Check	X		X	
Component Pressure Tests	X	X	X	
Engine Carcase Loading - Strength	X	X	X	X
- Deflection	X	X	X	X
Gyroscopic Loads	X	X	X	
Engine Mounting Point Test				X
Engine Slinging Point Test	X			
Simulated Foreign Object Damage Test		X	X	X
Engine Heat Rejection and Oil Cooling		X		
Engine Operating Attitude Test			X	
Effect of Power Turbine Shaft Failure			X	
Failure Indicating System Test			X	
Continuous Ignition Test			X	
Exhaust Gas Overtemperature			X	
Cooling Air Supply Failure			X	
Maintainability Demonstration	X	X		

[illegible]. ENVIRONMENTAL

	I Eng BI [illegible]	MIL-E-8593A	BCAR	FAR Part 33
Multiple Small Bird Ingestion	X	X	.	X
Multiple Medium Bird Ingestion	X	X	X	X
Single Large Bird Ingestion	X	X	X	X
Operation in Icing Conditions	X	X	X	X
Hailstone Ingestion	X	X	X	X
Atmospheric Water (Rain) Ingestion	X	X	X	X
Cabin Air Contamination	X	X	X	
Corrosion Susceptibility (Salt Water Ingestion)	X	X		
Sand Ingestion	X	X		
Smoke Test	X	X		
Armament Gas Ingestion		X		
Noise Survey		X		
Exhaust Emissions		X		
Nuclear Hardening		X		
Radar Cross-Section		X		
Infra-Red Radiation		X		
Electromagnetic Interference and Compatibility		X		

TABLE 6

CORROSION SUSCEPTIBILITY TESTS

UK

SUSCEPTIBILITY TO CORROSION

1. Motor engine; spray salt solution into engine intake and over exterior.
2. Plug openings and leave to stand for 1 week.
3. Repeat 1.
4. Repeat 2.
5. Strip and examine

TOTAL TIME [illegible] WEEKS (67[illegible] HOURS)

SALT WATER EFFECT

1. Performance calibration.
2. [illegible] Half-hour cycles.

MINS	RATING	SALT WATER INJECTION
5	Maximum	ON
20	Maximum Continuous	ON
5	Ground Idle	OFF
-	Two Accelerations from Idle to Max Rating	OFF

3. Performance calibration before and after washing.
4. Strip and Examine.

NOTE: Compressor washing to be carried out during test if performance falls to a pre-determined level.

ENGINE RUNNING TIME: 10 hours.

US

CORROSION SUSCEPTIBILITY TEST

25 x 48- HOUR CYCLES AS FOLLOWS:

Time (hours)	Engine Operating	Salt Water Injection	Engine Air	
			Temp	R.Humidity
3	Operating (Note 1)	ON	$10^{\circ}C$	73% Min
2	Not Operating	OFF	Atmospheric	Atmospheric
7	Not Operating	ON	$10^{\circ}C$	93% Min
12	Not Operating	OFF	$43 \pm 5^{\circ}C$	90% Min
24	Repeat first 12 hours but with engine operation as in Note 2.			

Note 1
4 x 10 m cycles of [illegible] m Maximum
5 m Idle
110 m Max Continuous
30 m Intermediate

Note 2
10 m Max Continuous
6 x 5 m cycles of $2\frac{1}{2}$ m Idle
$2\frac{1}{2}$ m Intermediate
130 m 90% Max Continuous
10 m Max Continuous

TOTAL TIME: 1200 HOURS, INCLUDING 150 HOURS OF ENGINE RUNNING

FIG. 1

D Eng RD 2100 Schedule G

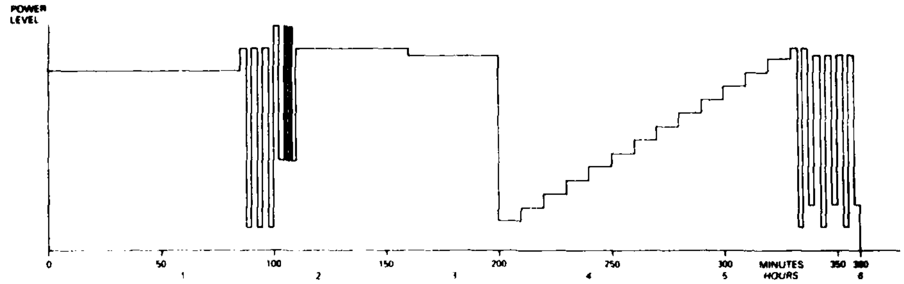


FIG. 2

BCAR C4 - 6 + BLUE PAPER No. 683

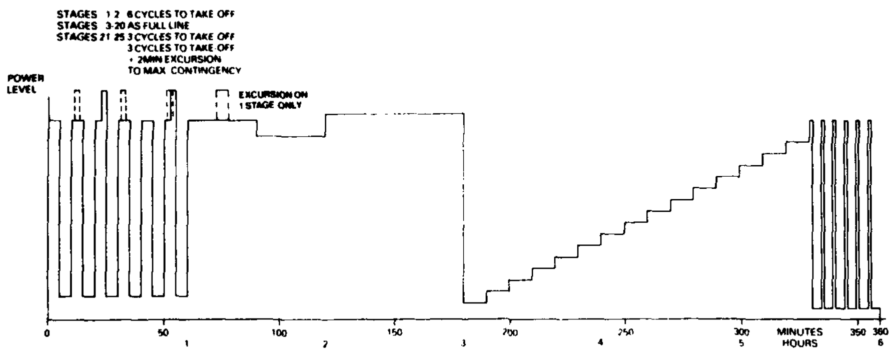


FIG. 3

FAR Part 33

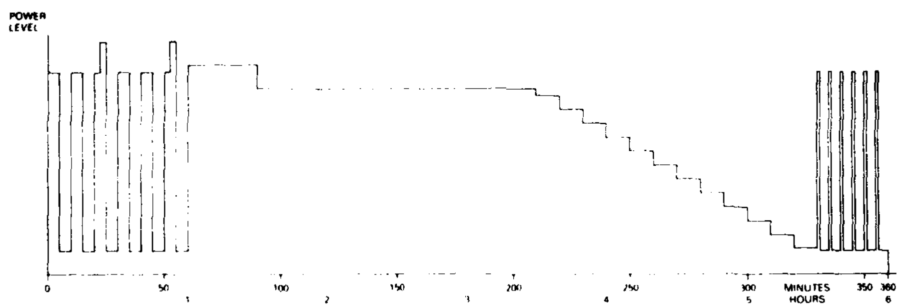
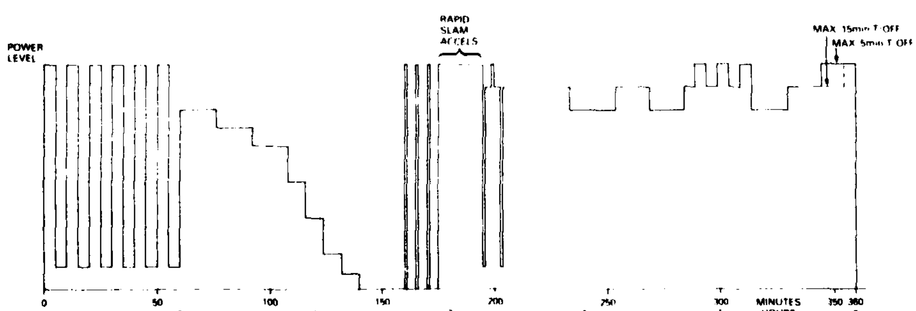


FIG. 4

MIL - E - 8593A



DISCUSSION

V.Fiorini, R.A.I., It

First, the author is to be congratulated for his comprehensive review.

My question refers to the present engine acceleration rate requirement (both FAR33 and JARE refer to 5 seconds time), in the light of recent multi-service helicopter applications which have recently employed a collective bias actuator to increase the acceleration rate of power response of the engine by some artificial means, what is the UK position with respect to future engine requirements in this area?

Author's Reply

There is at present no requirement for acceleration response time in the UK general specifications DEngRD2100 and 2300. This is a subject which is usually dealt with in the individual engine specifications. For helicopter engines I would not consider an acceleration time of 5 seconds to be sufficiently fast, and I would expect to see a figure of 3.5 seconds in an individual engine specification.

Alain Deveaux, Service Technique des Programmes Aéronautiques, Fr

You indicated that for the determination of the longevity of critical parts you would not be satisfied with an analytic study, but would demand practical tests. Today we possess a sizeable amount of knowledge concerning crack propagation, and the use of computers enhances the precision with which we will be able to grasp phenomena. What reasons do you have to adopt the policy you propose, and what kind of tests would you include in the regulations?

Author's Reply

The lifing procedure which we have recently drawn up anticipates that safe cyclic lives will be derived mainly from spin pit testing. We do not consider that the development of computational techniques has yet reached a stage where we can rely on them alone for establishing lives. There is in the UK a programme for developing these techniques, which includes work on fracture mechanics and crack propagation. This programme is entitled "Life and Methods Programme" (LAMP). When this programme is completed, we shall reconsider our policy concerning low cycle fatigue lifing methods.

BOEING COMMERCIAL AIRPLANE COMPANY ENGINE PROCUREMENT PRACTICES

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ABSTRACT

For the past decade Boeing has had a unique position among the world's commercial airframe companies in offering major portions of our product line with a variety of engines supplied by all three large commercial manufacturers. An example is the 747 airplane with 15 engines at 12 thrust ratings in 6 nacelles. Our new airplanes, the 757 and 767, are also being offered with several engines from the same manufacturers. Boeing has developed a framework of interrelated documents and an internal evaluation methodology to ensure consistency and control of these multiple engine offerings. This paper briefly defines the documentary framework currently used and provides, in some detail, our engine performance evaluation and development program monitoring methods.

BACKGROUND

The introduction of wide body aircraft with high bypass ratio engines into commercial service in the early 1970's was accompanied by numerous performance, operational, and maintenance problems. The initial engine installation on the 747 suffered from several problems, for a time even impeding scheduled airplane deliveries (fig. 1). The entire airplane program was delayed by an airplane subsystem over which Boeing was able to exert little control, nor did we initially possess sufficient technical understanding of the causes of these engine problems. The most important job Boeing and the engine manufacturer had was to cure the immediate engine and airframe deficiencies affecting airplane deliveries and performance guarantees. From a vantage point of 10 years, we now believe both Pratt & Whitney and Boeing did a pretty good job of developing solutions to these initial problems.

Shortly after getting the program on a more even keel, we began exploring improvement of 747 program offerings by installing alternative engines. Optional engines were not new to Boeing airplanes; the 707-420 was a variation of the 707-320, equipped with the Rolls-Royce RCO-12 Conway rather than the standard Pratt & Whitney engines. However, prior to 1973, no substantial engineering and marketing campaign had been directed toward broadening the market of one airplane by routinely offering engines built by a variety of manufacturers. Accordingly, Pratt & Whitney, General Electric, and Rolls-Royce were encouraged to offer improved derivatives of the initial versions of their large, high-bypass-ratio engines for installation on improved versions of the 747 airplane. The manufacturers were receptive, and figure 2 shows the 18 engine configurations and ratings certified for the 747 during the past 10 years. Those familiar with the requirements of Parts 25, 33, and 36 of the Federal Aviation Regulations will recognize this list as a formidable accomplishment, involving more than 2000 test hours on 115 different 747 airplanes, including the initial certification.

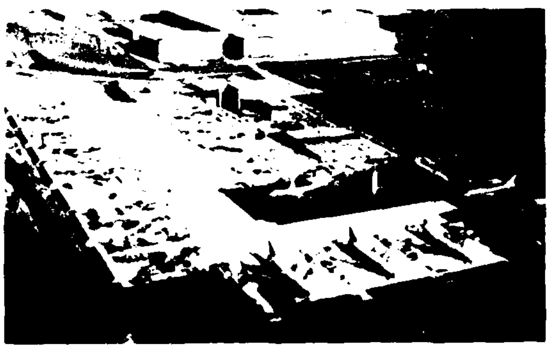

Figure 1. 747 Flight Line, March 1970

ENGINE MODEL	FIRST DELIVERY	ENGINE MODEL	FIRST DELIVERY
• JT9D 3	DECEMBER 1969	• CF6 50E1	JUNE 1978
• JT9D 3A	MARCH 1970	• RB211 524B2	NOVEMBER 1978
• JT9D 7	OCTOBER 1971	• CF6 45A	DECEMBER 1978
• JT9D 7A	APRIL 1973	• JT9D 7J	FEBRUARY 1979
• CF6 50E	OCTOBER 1974	• RB211 524B2SP	JUNE 1979
• JT9D 7F	DECEMBER 1975	• CF6 50E2	JULY 1979
• JT9D 7ASP	JANUARY 1976	• CF6 45A2	SEPTEMBER 1979
• JT9D 70A	APRIL 1976	• JT9D 7Q	OCTOBER 1979
• RB211 524B	JUNE 1977	• RB211 524C2	JUNE 1980

Figure 2. 747 Program Certificated Engines

Obviously, the support of airworthiness certification agencies from many countries, in addition to the United States FAA, was essential in producing this record and is appreciated. Today, current and prospective 747 customers are offered the range of engine installations (fig. 3), including multiple ratings from all three large engine manufacturers.

The availability of alternative engines for the 747 has fostered a healthy, competitive atmosphere among the world's engine manufacturers. The bargaining position of customer airlines has been enhanced in obtaining the lowest possible acquisition and maintenance costs. And, the Boeing Commercial Airplane Company is now able to offer a wide range of airplane/engine combinations with greater market penetration.

INTRODUCTION

As solutions to the early engine problems became apparent, more advanced ratings, with their inherent risks, were offered. Our engineering staff continued to expand our substantial engine data library, and continued developing more comprehensive modeling programs. A series of administrative and technical procedures evolved that today allow for a detailed, technical understanding of a proposed engine and its developmental program risks. Using this as a basis, we are able to establish realistic aircraft performance guarantees and delivery schedules. This applies not only to production airplanes, but also to our new airplane programs, the 757 and 767.

Requests for new or derivative engine proposals may originate from any quarter. An airline may acquire a new, long range route authority that dictates an engine with higher takeoff thrust to carry the larger fuel loads, or an engine of lower fuel consumption to increase range. An engine manufacturer supported by in-house or government research funds may develop higher efficiency components or improved turbine cooling schemes and be anxious to market these fuel saving features. Our own marketing studies might indicate that a higher gross weight airplane requiring higher thrust engines would be well received. Whatever the incentive, the potential business opportunity would be examined by Boeing, followed by discussions with the various engine manufacturers. Both would explore the technical and economic features of the engine configuration and rating under consideration, including the estimated development timetable. If these general discussions appear to be fruitful and a market for the improved airplane seems probable, then detailed business and technical discussions can be undertaken. Comprehensive negotiations may result in a series of business and technical agreements between Boeing and the engine manufacturer.

The business discussions with the engine manufacturer and the coordination of all activities among the Boeing technical, marketing, financial, and other groups are handled by the Engine Management Group. The resulting business agreement includes contract terms and conditions, detailed engine specifications, Boeing's program support plan, and the customer airline's product support plan (fig. 4).

AIRCRAFT MODEL	ENGINE MODEL
747 100B	• JF9D 7A/ 7F / 7J • CF6-45A2/ 45B2/ 50E / 50E 1/ 50E2 • RB211 524B2/C2
747 200B	• JT9D 7A/ 7F / 7J/ 7Q • CF6-50E/ 50E1/ 50E2 • RB211 524B2/C2/D4
747 200F/G	• JT9D 7A/ 7F/ 7J/ 7Q • CF6-50E/ 50E1/ 50E2 • RB211 524B2/C2/D4
747SP	• JT9D 7A/ 7F/ 7J • CF6-45A2/ 45B2 • RB211 524B2/C2

Figure 3. 747 Program Engines Offered

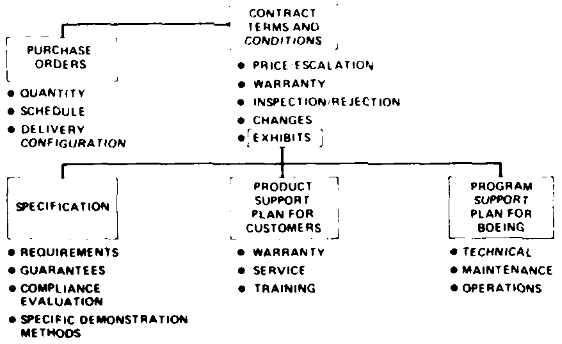

Figure 4. Engine Business Agreement

The detailed engine specification is the technical keystone of the business agreement. The guaranteed thrust, fuel consumption, noise, stability, weight, starting, vibration, and installed operating margin characteristics are defined. Also defined are the special procedures for all bench, ground, and flight tests required to demonstrate compliance with the guarantees. Additional requirements for the engine, such as bleed air or electrical power, airplane interface definitions, supporting test documentation and analyses, and acceptance tests and limits, are similarly spelled out.

TECHNICAL AUDIT

The technical evaluation of any proposed engine offering attempts to answer the following questions:

- Are the thrust ratings, engine configuration, and weight appropriate for the intended application?
- Will the planned fuel consumption be achieved, and will the in-service deterioration rate be acceptable to the airline customer?
- Is the engine properly sized and designed to allow for later growth?
- Will the engine comply with applicable government regulations, including noise and emission regulations?
- Will the stability and control characteristics be satisfactory?
- Will the engine possess adequate temperature and rpm margins?
- Have sufficient engine manufacturer resources been devoted to the development program in order to minimize the possibility of unforeseen performance, durability, or delivery problems?

Throughout our long airplane development history, we have maintained and continued expansion of our own engine development files, engine and component analysis methods, and performance trend prediction computer programs. These engine audit tools have been derived from in-house studies, industry and Government published documents, and continuing discussions with the engine manufacturers. Very stringent internal security procedures are used to safeguard the proprietary data of the individual engine manufacturers.

The evaluation process begins with a very detailed Boeing request for data that substantiates the engine characteristics (fig. 5). Thrust specific fuel consumption (TSFC) is probably the category that receives the most attention during the audit. The engine manufacturer's guaranteed TSFC is usually based on a thermodynamic model of the engine cycle using component analyses or test data, engine ground tests, or flight tests. As shown in figure 6, we independently assess the basic thermodynamic cycle, component efficiencies, required turbine cooling flows, and installation losses in order to estimate the initial TSFC performance, as well as any projected in-service TSFC deterioration. The thermodynamic cycle defined by the engine manufacturer is first duplicated using our own cycle analysis computer programs. We then examine the expected efficiency of each component and compare available test data and the measured trends observed in similar components. If the claimed efficiency is considered optimistic, based on state-of-the-art considerations and the development program identified, the efficiency is degraded to a more realistic level. Similarly, if the turbine cooling flows result in excessively high turbine metal temperatures, the cooling flows are increased. If the inlet and exhaust system installation is provided by the manufacturer, inlet recovery characteristics and nozzle coefficients are also scrutinized and adjusted, if necessary. The short-term and long-term TSFC deterioration, due to seal wear, airfoil erosion, and thermal distortion, is estimated from NASA analyses and the individual engine's characteristics. Using these techniques, we then predict the probable initial and in-service TSFC levels of the proposed engine.

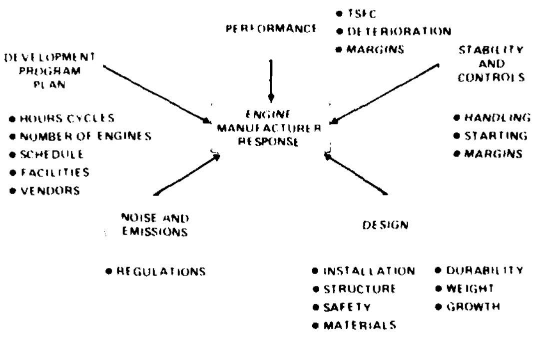

Figure 5. Technical Audit Data Request

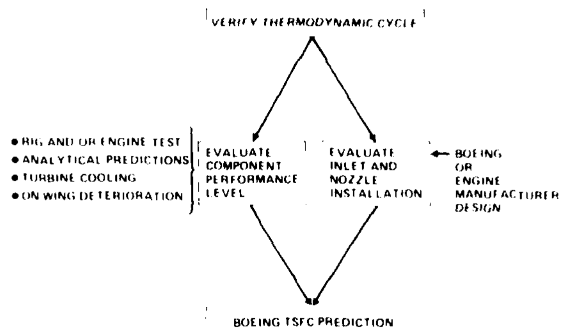

Figure 6. Fuel Consumption Audit

Engine stability is examined to ensure adequate component surge margin, and reliable starting and handling characteristics (fig. 7). Component surge margins may be reduced by manufacturing tolerances, deterioration, flow distortion, variable geometry positioning errors, thermal transients, Reynolds number, and transient variations in the operating and surge lines. During the audit, considerable emphasis is placed on identifying critical flight conditions where surge margin is calculated to be minimal.

The engine's overall structural mechanical design is reviewed (fig. 8) to ensure that the proposed installation is simple, accessible, and can be easily maintained. The static and dynamic structural characteristics are examined, along with the materials used. We also examine the safeguards that are provided against the effects of system failures such as fires, foreign object damage, or blade failure. The engine manufacturer's in-service track record for similar installations is carefully reviewed. Component durability is scrutinized intensively because of its direct effect on fuel costs, maintenance costs, engine reliability, engine safety, and thrust growth capability for later versions of the engine. Component lives are estimated from the engine cycle and design, airplane mission profile forecasts, material properties, empirical data, and our own parts life analyses. These studies are carried out in close coordination with the TSFC analysis previously described. If the component efficiencies are degraded and the turbine cooling flows are increased substantially, the engine's thermodynamic cycle may be sufficiently altered to increase the combustor and turbine temperatures and the shaft rpm required to produce the desired thrust levels. As a result, the basic engine design may be limited by vibrational, stress, or material property considerations and may be unable to achieve the intended thrust with adequate temperature and rpm margins. This could necessitate certification at higher turbine temperatures and rpm than the manufacturer had anticipated or, in the worst case, could require engine redesign and possible delay to the airplane program.

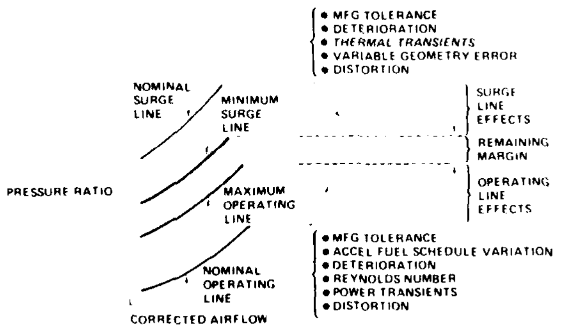

Figure 7. Component Stability Margin

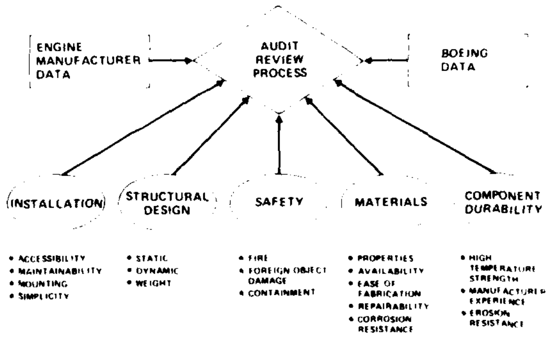

Figure 8. Structural/Mechanical Design Review

Noise and emission characteristics are also examined and compared to current and expected future regulations to ensure conformity to applicable statutes.

Finally, the engine manufacturer's intended development program is examined for the number of test engines committed to the program, the calendar time allotted to the development schedule, the number of expected test hours and cycles, and several other critical characteristics. (A greatly simplified example of a new engine program development schedule is shown in figure 9.) Derivative

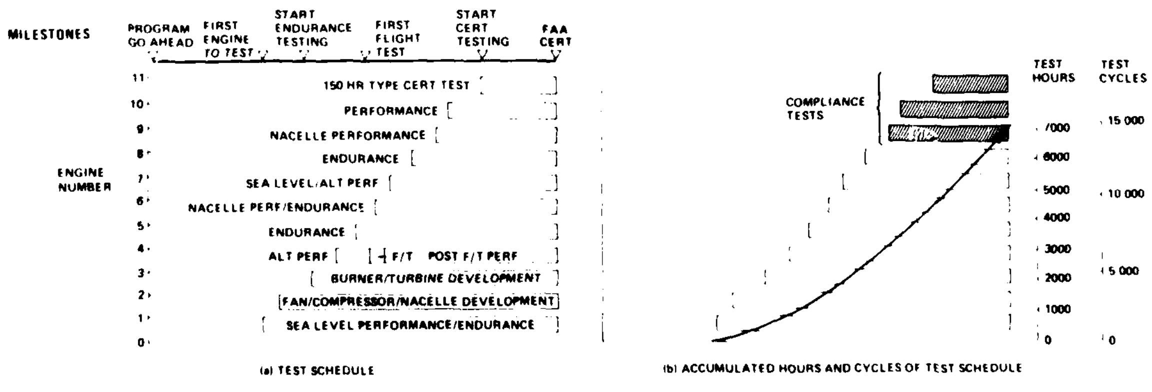

Figure 9. New Engine Development Plan

engine development programs tend to be shorter with fewer dedicated test engines because of the program development background of the parent engine. Possible conflicts in facility utilization with other unrelated development programs are also considered, along with a general assessment of consistency between the stated technological goals envisioned by the manufacturer and the resources allocated toward achieving these goals.

RISKS

From these examinations of expected performance, stability, durability, mechanical design, noise, emissions, and development schedule objectives, we can estimate the probable engine program risk to Boeing and the customer airline. As figure 10 illustrates, our technical and administrative conclusions are communicated back to the engine manufacturer, who is then given the opportunity to respond with appropriate data, analyses, or rescheduling alternatives. It is not our intent to directly impose design alternatives or scheduling changes on the manufacturers, but rather to identify those areas of concern which adversely affect our confidence in the proposed engine.

This technical evaluation, or audit, represents only the beginning of an ongoing dialogue with the engine manufacturer that continues throughout the development, flight test, and certification of the airplane/engine combination. The engine and airframe programs proceed in parallel. Following extensive development testing, that may include both ground and flight testing, the engine is certified by the FAA. Engine certification does not represent a performance endorsement by the FAA, but the certification testing does establish minimum standards for engine durability, foreign object ingestion, transient response, and other categories relating to flight safety. The Boeing-owned 747 is routinely used as the flight test vehicle for new or derivative engines for the 747 airplane program. It will also play a major role in the flight testing of the engines for the 757 and 767 airplane programs. The initial production engines and nacelles that are used in the airplane flight test program are thoroughly ground tested and calibrated prior to installation on the airplane. Compliance will now be demonstrated with the engine installation requirements of FAR 25 and, hopefully, with the performance guarantees of the engine specification. A typical overall engine/airframe development program is illustrated in figure 11.

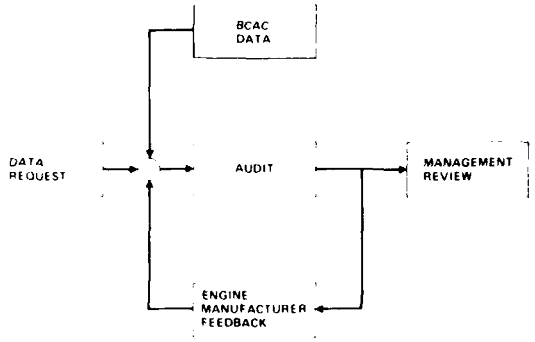

Figure 10. Risk Assessment

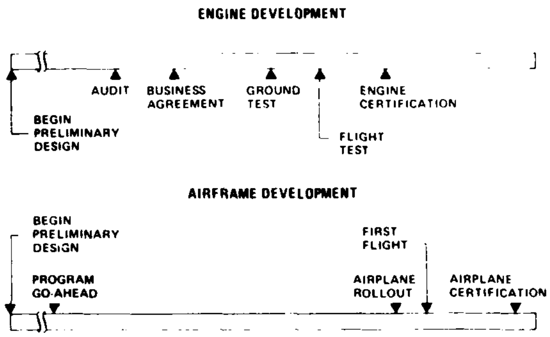

Figure 11. Program Schedule

The importance of continuous, close cooperation between the engine and airframe manufacturers cannot be overemphasized. The impact of engine program delays on airplane production programs, as well as on airline training and scheduling activities, can be enormous. Through the continuous technical audit process which is being vigorously followed on our new airplane programs, the 757 and 767, figure 12, it is our intent to provide our customers with airplanes delivered on schedule, with reliable engines of predictable performance, and to minimize introductory engine service problems.

Figure 12. 757/767 Programs

CONCLUSIONS

The early history of commercial wide-body airplanes was plagued with engine performance, stability, and durability problems. The lessons learned from resolving these introductory problems enabled the Boeing Commercial Airplane Company to adopt administrative and technical procedures which have

- Encouraged a competitive atmosphere among the world's engine manufacturers and thus minimized airlines' acquisition, fuel, and maintenance costs
- Reduced the likelihood of unforeseen development and introductory problems through our audit and monitoring procedures
- Expanded Boeing penetration of the commercial airplane market and allowed Boeing to better tailor products to specific customer requirements

Our in-house engine review procedures will continue to be refined as additional analysis and development data become available in our continuing effort to assist the airlines of the world in providing the traveling public with low cost, dependable, and safe air transportation.

DISCUSSION

M.Mihail, Bureau Veritas, Fr

Mr Nordstrom's lecture was excellent and very timely. I would like to ask him three questions:

(1) How does Boeing see the problem of the incidence of the decay of the power of engines and of the length of flights on the guarantee of performances, particularly the guarantees concerning the power of the aircraft and its specific level of fuel consumption? Does Boeing see it as a problem that concerns the manufacturer or rather the customer?

(2) Do you think that the position of the engine in the aircraft modifies the vibration, and that the vibration of a positioned engine is different? Do you actually define, or do you intend to define in the future maximal values of vibration for positioned engines in function of their position in the aircraft?

(3) In one of your slides "Structural Mechanical Design Review", you speak of consultations with the engine manufacturers and the customers to determine the design review. Do you have a meeting with the engine manufacturer before the meeting with the customers, the airline people, or is it all in one package?

Author's Reply

(1) Deterioration is normally broken down into two aspects. Short term deterioration of the engine is basically wearing in of seals and other rotating parts and usually results in a small increase of TSFC and engine temperatures. This deterioration which may occur prior to delivery of the airplane to the airline customer is normally considered under the airplane performance guarantees. Longer term deterioration of the engine is caused by erosion and/or corrosion of air foils, thermal distortion of air foils and other causes acting over or occurring during long term operation. Normally, the engine manufacturer deals directly with the airline operator in this regard.

The engine thrust setting parameter is chosen such that the desired engine thrust is obtained regardless of engine deterioration, so that the full thrust rating selected is always available to the operator. When the engine is not capable of delivering rated thrust within the operating RPM or temperature limits, an engine overhaul is required. Evidence exists indicating that long term deterioration is primarily sensitive to engine cycles, rather than hours of operation.

(2) With respect to engine vibration, engine position vibration effects in general occur in the range below 10 Hz, while vibration effects of interest to the engine usually exceed 10 Hz. Because of this separation, the variation of engine position normally does not affect engine vibration limits. We have no current plans to define engine vibration limits as a function of engine position.

(3) We have continuous ongoing discussions with potential airline customers; we have continuous ongoing discussions with all potential engine manufacturers; and we have continuous ongoing discussions with the regulatory authorities. The final configuration we specify evolves from all these discussions.

E.E.Abell, USAF, Aeronautical Systems Div., US

How much growth do you normally expect from a new engine when you are considering offering it on a new aircraft?

Author's Reply

We look for a minimum growth of 10% in the same frame size, that is, without changing the fan diameter. We also examine the possibility of generating a family of engines using the same core by downsizing or increasing the L.P. system.

PROPULSION SYSTEM TESTING REQUIREMENTS FOR A COMMERCIAL TRANSPORT

by

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ABSTRACT

This paper describes development and demonstration testing requirements for the next family of transport engines. It includes propulsion system performance and functional testing with emphasis on performance. Rationale is provided for the major tests required.

The testing requirements identified herein reflect experience gained in the L-1011 program. The paper stresses the need for integrated, closely coordinated test programs involving the airframe and engine companies. The test concepts, objectives, and definition of what constitutes performance demonstration are agreed before the airplane program go-ahead. Responsibilities for the various tests are assigned to the engine and airframe companies as appropriate. The requirement for correlating ground and flight propulsion performance results is discussed.

Finally, examples of propulsion testing during the L-1011 development program are described.

INTRODUCTION

In the development of early commercial transports, the tendency was for the airframe manufacturer to treat the engine as somewhat of a separate entity and adapt it to his aircraft. In this current age of advanced technology, high performance aircraft and propulsion systems, a more sophisticated approach is required. The engine can no longer be considered a separate entity because the aerodynamic and functional relationships between engine and airframe are much more critical than in the early transports.

Accordingly, in current and future transport aircraft it is important to use an integrated systems approach in developing the aircraft-propulsion system combination. This is necessary in order to fully exploit the performance potential and ensure mechanical systems functional compatibility. The aerodynamic and mechanical sophistication of modern transport aircraft has attained a level that requires the engine and associated systems be integrated carefully with the pod and the pod, in turn, be integrated carefully with the aircraft and associated systems.

This integrated systems approach requires highly coordinated design, development, and testing programs involving both the engine company and the airframe company. In some cases the engine company is given responsibility for developing the complete propulsion system. In other cases, the responsibilities are shared for portions of the propulsion system other than the basic engine. In either case a high degree of coordination is required.

There are three principal cases for consideration in propulsion system development and testing, namely, (1) new airplane and new engine model undergoing development at the same time, (2) new airplane which will use an existing engine (requires adaptation of engine and associated systems to airplane and associated systems), and (3) existing airplane which will use a new engine. Case (1) is the subject of this paper because it (a) offers the potentially greatest opportunities in the development of a commercial transport and (b) represents the greatest challenge to the airframe-engine company team, particularly when the airframe and engine both incorporate advances in technology. Case (1) also requires the most innovative effort in planning of the development and testing programs.

A simplified plan for a total propulsion system development program is illustrated in Figure 1. This program includes engine, nacelle pod, and propulsion subsystems development, as well as certification of the propulsion system in the aircraft. The major propulsion system development programs are listed below.

- FULL SCALE ENGINE DEVELOPMENT
- ENGINE PERFORMANCE
- PROPULSION SYSTEM-AIRFRAME AERODYNAMIC INTEGRATION
  - NACELLE DEVELOPMENT (AERODYNAMIC CONTOURS)
  - INLETS
  - EXHAUST/AFTBODY/NOZZLES
  - INLET/ENGINE DISTORTION COMPATIBILITY
  - THRUST REVERSER
  - THRUST REVERSER/AIRFRAME AERODYNAMIC INTERACTIONS
  - FOREIGN OBJECT AND HOT GAS INGESTION
- FUEL CONTROL SYSTEM
- STRUCTURAL
- ICING, FIRE PREVENTION, VENTILATION & DRAINAGE
- ENGINE/POD SEA LEVEL ENDURANCE TEST

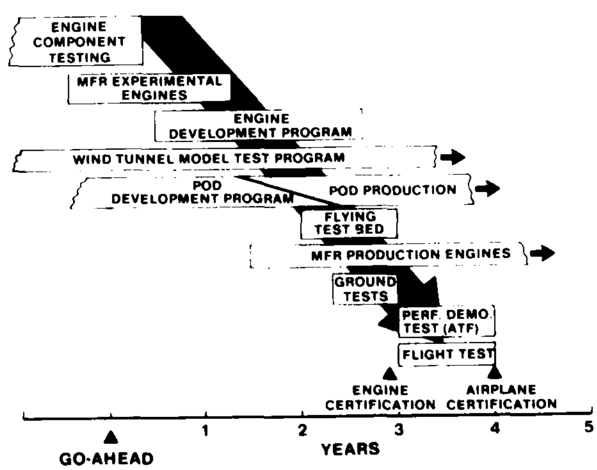


Figure 1. Simplified Propulsion System Development Program

- NOISE
- FLYING TEST BED
- FLIGHT TESTS

In light of the very large magnitude of the total propulsion system development program (Figure 1), this paper will concentrate on engine performance and propulsion system-airframe aerodynamic integration (including performance oriented and functionally oriented aerodynamic integration). The portion of the flying test bed and flight test programs related to these two subjects will also be discussed. The topics for discussion are enclosed in the boxes in the above list. The various test programs required for the development and demonstration of performance and propulsion-airframe aerodynamic integration are identified in Figure 2.

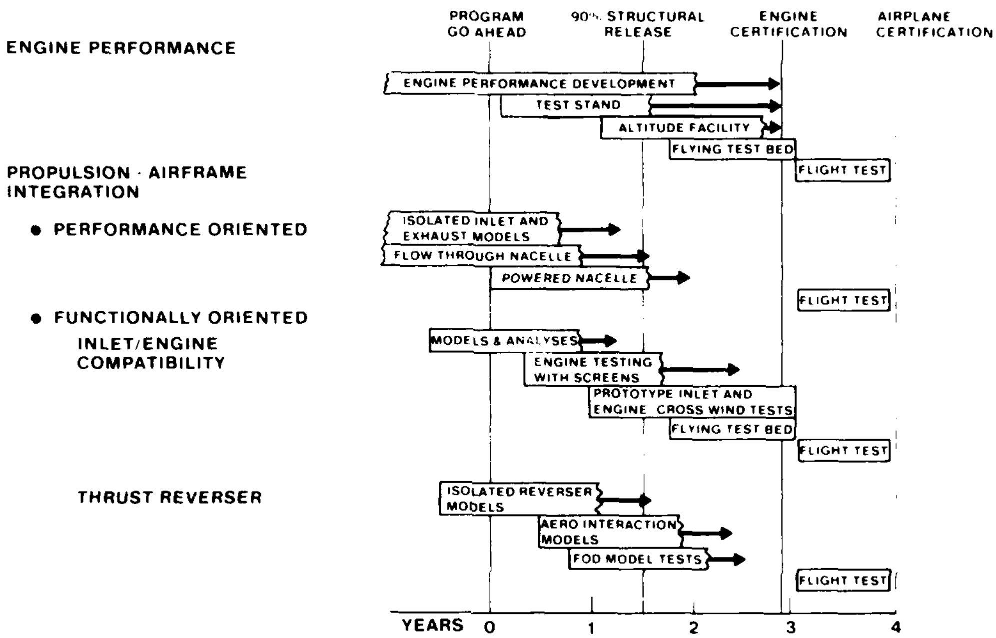


Figure 2. Performance and Propulsion-Airframe Aerodynamic Integration Development and Demonstration Testing

The purpose of the present paper is to (1) describe propulsion system testing requirements and associated rationale for the next generation of commercial transports and (2) show examples of propulsion system testing of a recently developed commercial transport.

While the discussion is centered around development and testing for turbofan propulsion systems, the general approach and philosophy would be similar for propfan (turboprop) systems.

ENGINE PERFORMANCE TEST REQUIREMENTS

This section presents a broad description of engine performance test requirements and associated rationale for the next generation of transport engines. The basic test concepts, objectives, and definition of what constitutes performance demonstration are agreed to before airplane go-ahead. As part of the evolution of required testing, responsibilities of engine company and airframer are also briefly described.

Categories of Performance

Two broad, general categories of performance exist for a commercial transport propulsion system, namely, operation at (1) rated engine thrust conditions (such as takeoff, maximum continuous, max climb, etc.) and (2) part power cruise (including high altitude cruise and low altitude holding). Accurate determination of thrust is required in both categories — in the first to ensure adequate aircraft takeoff, climb, and engine out performance and in the second to ensure that predicted engine cruise specific fuel consumption (SFC) is achieved. Overall aircraft performance related to the second category is determined by measuring the specific air range (ratio of true airspeed to engine fuel flow rate expressed in nautical miles per pound).

Accordingly, accurate determination of installed engine thrust is of paramount importance in the development and certification of a commercial transport. This is the genesis of, and primary requirement for, the testing described in this section. The thrust determination methodology is based on Reference 1. While accurate fuel flow measurement is also required, experience indicates this to be much less challenging than thrust determination.

Sources of Test Data — Facilities

In view of the critical importance and challenge of determining thrust throughout the airplane operating regimes, four sources of test data are required, namely, (1) sea level test stand, (2) altitude test facility (ATF), (3) wind tunnel test, and (4) flight test. Part of the agreement as to what constitutes performance demonstration involves "closing the loop" between these four data sources, as illustrated in Figure 3. The following represents a brief description of proposed procedures in using these facilities in the development of a

thrust map for the next family of engines in an advanced commercial transport. Due to the broad scope of this paper, only limited examples of data from prior tests are presented. The application of the thrust maps in determining the engine and airframe contributions to specific air range is also discussed.

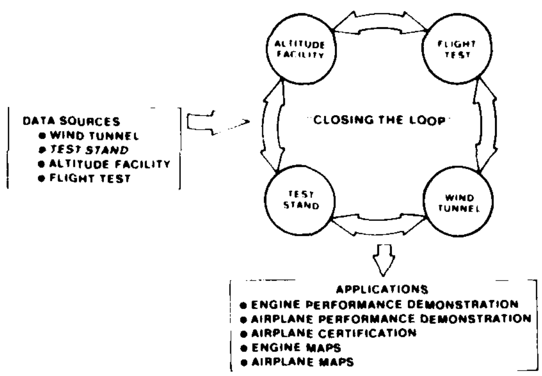


Figure 3. Engine Performance

Thrust Map Derivation Procedures

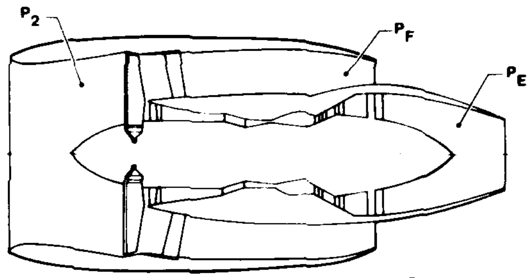
Determination of net thrust in flight requires that the engines be carefully calibrated against some quantity which can be measured accurately in flight. The ratio of area-weighted total pressures in the engine exhaust to the total pressure at the front face of the fan can be shown to be a dependable indication of net thrust. The only additional knowledge required is flight Mach number and altitude. The area weighted total pressure ratio approach is referred to as an "Integrated Engine Pressure Ratio" (IEPR) thrust indicating system. The engine net thrust is expressed by the relationship

$$F_N = \delta f(\text{IEPR}, M_0)$$

where δ = static pressure at altitude normalized by sea level pressure

M_0 = flight Mach number

This system is illustrated in Figure 4.



$$P_{\text{INTEGRATED}} \quad \frac{(P_F \times A_F) + (P_E \times A_E)}{A_F + A_E}$$

$$\text{IEPR} = \frac{P_{\text{INTEGRATED}}}{P_2}$$

Figure 4. Thrust Indicating System

This thrust relationship is initially determined by careful testing of the full-scale, flight qualified engine on sea level static test stands, such as shown in Figure 5, and in an altitude facility capable of simulating flight conditions (i.e., Mach number and altitude by providing appropriate ram pressure and temperature at the engine face and static pressure at the nozzle exhaust). The resultant thrust map is, of course, adjusted for differences in installation effects (i.e., inlet recovery, bleed, and power offtakes) between the ground test facilities and the actual airplane. The engine company is responsible for conducting these ground test thrust calibrations. The tests, however, also are observed and analyzed by airframe company engineering representatives.

As part of the aircraft flight test program, the ground test thrust calibrations described above are independently checked by a direct calibration of the thrust map between Mach numbers of 0 and 0.25. This can be carried out on the runway by the accurate measurement of distance as a function of time from which aircraft acceleration can be determined. An onboard instrumentation package, known as the Space Tracking Airborne Receiving System (STARS), provides this capability. Acceleration tests at various thrust levels and coasting deceleration tests at idle thrust are conducted in both directions along the runway and at various gross weights. This enables the extraneous effects of airframe drag, rolling friction, runway slope, and wind to be extracted. Although runway slopes and winds are known and accounted for in the data reduction procedure, the bidirectional test technique serves to remove any uncertainty in the accounting procedure. The thrust obtained from these STARS tests are then compared with the sea level test stand and altitude facility results.

Following the STARS testing, a substantial amount of takeoff performance testing is conducted during which all engine parameters are carefully recorded including IEPR, turbine gas temperature, and engine shaft speeds. The purpose of this testing is three-fold, namely, to (1) check engine and airplane takeoff and climb out performance using the derived low speed thrust map, (2) establish a data bank of engine parameters which will be used to satisfy FAA certification requirements, and (3) establish engine operating maps.

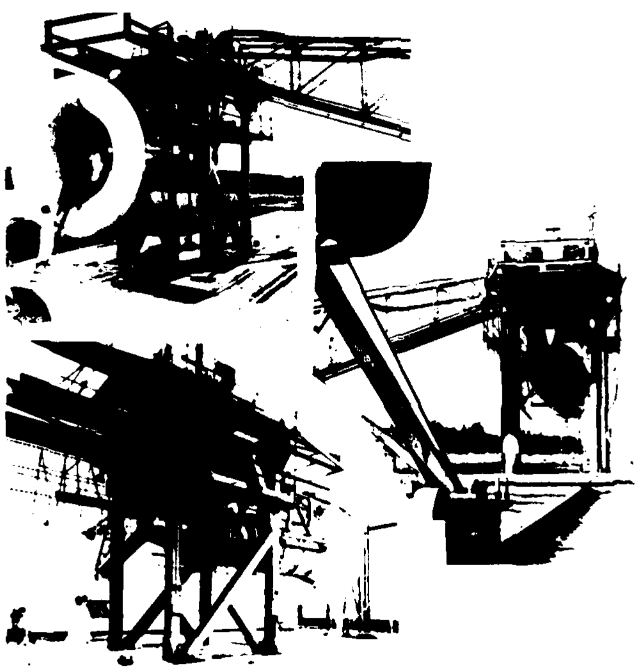


Figure 5. Open Air Static Test Stand

At Mach numbers greater than 0.25 other methods are required to check the thrust levels determined from the altitude test facility. Two such methods, using the airplane as a dynamometer to determine aircraft drag, are (1) idle descent tests at low thrust levels and (2) constant altitude airplane deceleration tests using an inertial system. This testing is conducted over a range of altitudes and Mach numbers between approximately 25,000 and 35,000 feet and 0.7 to 0.85, respectively.

In determining in-flight thrust using method (1) (idle descent), data are taken during constant Mach number descents conducted at low thrust levels, but not so low that an unknown spillage drag increment can appear and cloud the test results. Aircraft drag is determined from the rate of descent, aircraft weight, and the engine thrust. The low thrust levels used in the determination of aircraft drag are obtained from the altitude test facility. The aircraft is then flown at the same Mach number in level flight and the IEPR required to fly is recorded. Since thrust is equal to drag for this level flight condition and drag is known from the idle descent test, a thrust/IEPR point on the thrust map is obtained.

The advantage of low thrust descent testing is that the great majority of the required thrust is provided by the flight path component of the force of gravity, while the thrust contribution of the engines is small. If the relative error of the low-thrust calibration obtained in the altitude facility is at all reasonable, then the thrust used in the idle descent testing will have only a small effect on the accuracy of the aircraft drag. Several such descents at different Mach numbers will serve as anchor points for the flight calibration of thrust.

Examples of drag polars derived from L-1011 idle descent flight tests compared with predictions based on wind tunnel data (adjusted for Reynolds number) are shown in Figure 6. The agreement between the two sets of data is excellent, confirming the aircraft drag and the engine thrust map.

The second method, used as an independent check of the altitude test facility (ATF), consists of a flight test involving airplane deceleration at constant altitude with low thrust levels. An inertial system is used to measure the aircraft deceleration. This method is currently under investigation.

In the speed regime between Mach 0.25 and 0.7, the airplane can be flown in level flight, in climbs, and in descents, so that a wide range of thrust levels can be made available at several different stabilized 1-g flight conditions of Mach number and altitude. Since at any specific flight condition the drag of the airplane is independent of flight path angle, a good check on the incremental consistencies of the thrust map at several Mach numbers is possible. The difference in thrust required for climb or descent can be calculated with respect to level flight and compared with the predictions of the thrust map. A further check on the incremental consistency of the thrust map can be carried out in level flight for several conditions by varying the split in thrust between the wing engines and the center engine in a three-engine aircraft. For each split the equality of total thrust is ensured by maintaining equality of both speed and altitude.

In summary, the thrust map is determined by the combination of low speed, high speed, and intermediate speed and altitude flight testing and is then compared with data obtained in the wind tunnel, the altitude test facility, and the sea level test stand.

Summarizing responsibilities, the engine company is responsible for the conduct and analyses of sea level test stand and altitude facility tests. The airframe company will be a highly interested, active participant. The airframe company is responsible for the conduct and analyses of flight testing with the engine company as a highly interested, active participant. Thus, "closing the loop" between the four data sources (sea level test stand, altitude facility, wind tunnel test and flight test) is, obviously, a joint, coordinated effort between the engine company and the airframe company with shared responsibility.

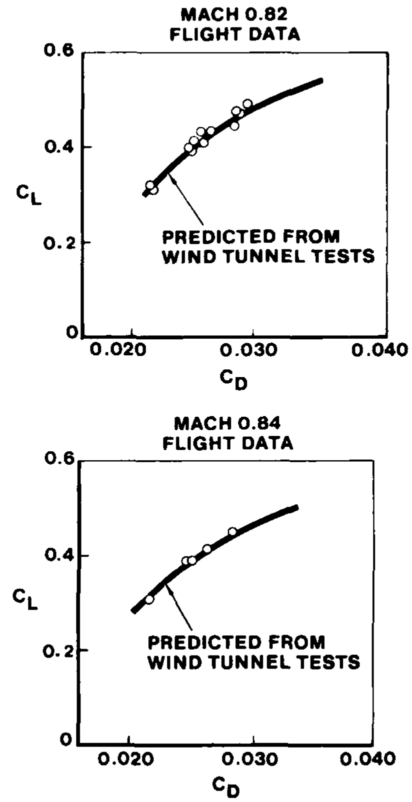


Figure 6. – Comparison of Wind Tunnel and Flight Data

Specific Range and Thrust Map Application

The final products of all of the efforts described above are sets of thrust maps, fuel flow maps, and airplane performance charts covering the operating regime of the aircraft and propulsion system. These maps are agreed by the airframe and engine companies and are used in (1) certifying the airplane (2) demonstrating that the airplane meets its performance commitments and (3) demonstrating that the engine has met its performance commitments.

Having the thrust map, IEPR required to fly, and the measured fuel flow, it is possible to determine the engine and airframe contributions to specific air range (SAR).

$$SAR = \frac{V_O}{W_f} = \frac{V_O}{W_G}\left(\frac{L}{D}\right)_{AIRFRAME}\left(\frac{1}{SFC}\right)_{ENGINE}$$

where
- V_O = aircraft true airspeed
- W_f = engine fuel flow rate
- W_G = aircraft gross weight
- L = aircraft lift
- D = aircraft drag
- SFC = specific fuel consumption, W_f/F_N
- F_N = engine net thrust

The airframe component of the specific air range is the lift-to-drag ratio. For stabilized 1-g flight conditions, the lift is equal to the aircraft weight and the drag is equal to the engine net thrust. The aircraft weight is determined by subtracting the time integrated measured fuel flow rate from the aircraft weight measured prior to takeoff using accurately calibrated scales. The drag at the IEPR required to fly is determined from the engine thrust map.

The engine component of the specific air range is the specific fuel consumption, which is the ratio of fuel flow rate to net thrust. The fuel flow rate is measured during flight, and the net thrust is again determined from the engine thrust map.

In the event of any performance deviation from the predicted level of airplane performance, a knowledge of the contributing elements enables determination of the source of the deviation. This means, of course, that comparisons between predicted and achieved engine and airframe performance can be made at this point in time. Application of the thrust map in this manner is possible since the loop between sea level test stand, altitude test facility, wind tunnel test, and flight test performance has been closed.

In addition to allowing the separation of the airframe and engine contributions to specific air range, the thrust map also enables the development of EPR setting charts. These charts are used by the flight crew to determine the level of EPR that must be set to ensure the proper rated thrust level during aircraft takeoff and climb. Also, maximum cruise setting charts are used to establish the aircraft initial cruise altitude when appropriate. The takeoff EPR setting chart is a function of altitude and ambient temperature, while the climb and maximum cruise charts are functions of Mach number, altitude, and temperature. In all cases, the setting chart is determined from the thrust map and the required rated thrust. High accuracy of the thrust map is required in order that application of the setting charts results in adequate levels of thrust without exceeding engine temperature and shaft speed limits.

Fuel flow, turbine gas temperature, and shaft speed maps are also derived from the flight test data. These maps are important for monitoring engine performance and integrity and are frequently used in consistency checks of engine performance.

PROPULSION SYSTEM – AIRFRAME AERODYNAMIC INTEGRATION TEST REQUIREMENTS

This section presents the requirements for development testing and demonstration of the aerodynamics of the overall propulsion system integrated with the airframe. This is a highly challenging area of developing technology which encompasses nacelle, inlet, exhaust system, and pylon and their mutual interactions as well as their interactions with the wing. This area requires extremely closely coordinated efforts between the engine company and the airframer because there are many subtle trades between the propulsion system internal flow and the external flow around the nacelle, particularly when the aerodynamic interactions are properly considered.

One of the reasons aerodynamic integration of the propulsion system with the airframe is so challenging relates to the rapidly changing flow directions and Mach numbers in the region of the nacelle as illustrated in Figure 7. In this simplified illustration of a long nacelle in a conventional location under the wing, the wing flow field is changing rapidly from an upwash ahead of the wing to a downwash downstream of the leading edge. In the ideal case, to achieve minimum drag, the nacelle contours would be shaped to follow the wing flow field streamlines in cruise. The resulting nacelle with a highly curved centerline is incompatible with engine geometry (straight centerline) and other considerations such as optimal inclination of the gross thrust vector. Accordingly, compromises are involved in arriving at a propulsion system properly integrated with the airframe. The scope of the nacelle design challenge is further enlarged by the variety of candidate wing/nacelle configurations involving both aerodynamic and mechanical trades, some of which are illustrated in Figure 8.

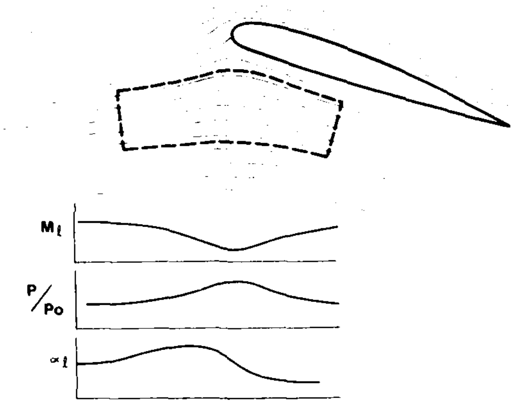


Figure 7. – Ideal Nacelle Aerodynamic Contour

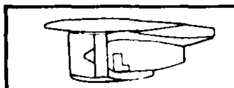
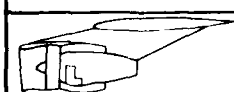
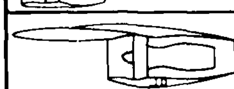
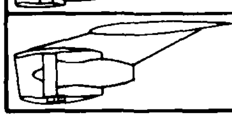
	NACELLE MOUNTING	FAN DUCT LENGTH	EXHAUST / GEAR BOX
	MID CHORD	3/4	BARE ENGINE / CORE MOUNTED GEAR BOX
	PYLON	3/4	BARE ENGINE / CORE MOUNTED GEAR BOX
	AFT WING	LONG (MIXED FLOW)	INTEGRAL FAN DUCT / FAN MOUNTED GEAR BOX
	PYLON	LONG (MIXED FLOW)	INTEGRAL FAN DUCT / FAN MOUNTED GEAR BOX
	PYLON	SHORT	BARE ENGINE / CORE MOUNTED GEAR BOX
	PYLON	3/4	INTEGRAL FAN DUCT / FAN MOUNTED GEAR BOX

Figure 8. – Candidate Wing Nacelle Configurations

The development of a propulsion system aerodynamically integrated with the airframe has to date consisted of a combination of analysis and testing with heavy dependence on the latter. While efforts are underway which may reduce somewhat future dependence on testing, the emphasis in this discussion is on testing.

Categories of Development Testing

In developing a new commercial transport aircraft, the nacelle aerodynamic development will proceed in parallel with the engine development. The airframer will, of course, obtain from the engine company the pertinent engine dimensions, airflows, pressure ratios, and other engine parameters that affect nacelle design.

The airframe and engine companies will proceed with somewhat parallel efforts in the areas of nacelle aerodynamic lines, inlets, and exhaust systems (including thrust reverser). With a new airplane and new engine combination, a certain amount of duplication is desirable in order to achieve the best design compromises and to validate performance trends in the limited time available. The propulsion system components to be examined are defined and identified in Figure 9. Initially, testing will be performed by each company on isolated components and integration will occur as an iterative process as the components come together. The process is necessarily iterative because (1) the new airplane and the new propulsion system are being developed simultaneously and (2) the aerodynamic flows and related integration processes are complex.

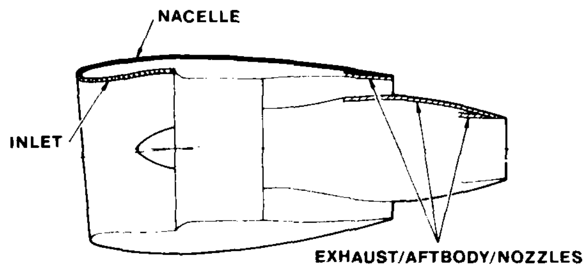


Figure 9. Major Aerodynamic Contours Affecting Performance

Two categories of aerodynamic development testing are defined for the purpose of this discussion, namely, (1) performance-oriented nacelle aerodynamic contour development, including: (a) nacelle, (b) inlet, and (c) exhaust/aftbody/nozzles and (2) functionally-oriented aerodynamic development, including: (a) inlet/engine compatibility, (b) thrust reverser effectiveness, (c) thrust reverser airframe aerodynamic interactions, and (d) foreign object and hot gas ingestion. Development of components in these two areas are mutually interdependent, and in some cases present conflicting configuration requirements. Accordingly, analyses and testing in the two areas must proceed in parallel. Within the scope of this paper, it is possible to cover only the main, principal elements involved in aerodynamic development of the propulsion system. It is also possible, because of scope, to provide only limited examples of test results, representing a very small percentage of total testing of an actual development program.

Responsibilities between airframe and engine companies for development and demonstration testing of a new propulsion system are as proposed in Figure 10. For some components, the assignment of prime responsibility is obvious, such as inlet and nacelle contour development to the airframer. In other cases, the assignment is less obvious, such as the exhaust/aftbody, which generally experiences significant aerodynamic interactions with the wing flow field and the pylon. In such cases, parallel, cooperative, coordinated efforts are pursued jointly by both engine company and airframer to arrive at a mutually agreeable, optimal configuration. This type of coordinated effort is essential in any new airplane, new propulsion system program.

	RESPONSIBILITY	
	ENGINE CO.	AIRFRAME CO
• PERFORMANCE ORIENTED NACELLE AERO CONTOUR DEVELOPMENT		
1 NACELLE	2	1
2 INLET	2	1
3 EXHAUST/AFTBODY/NOZZLES	1	2
• FUNCTIONALLY ORIENTED AERODYNAMIC DEVELOPMENT		
1 INLET/ENGINE COMPATIBILITY	1	2
2 THRUST REVERSER EFFECTIVENESS	1	2
3 THRUST REVERSER/AIRFRAME AERODYNAMIC INTERACTIONS	2	1
4 FOREIGN OBJECT AND HOT GAS INGESTION	1	2

1 PRIMARY RESPONSIBILITY
2 SECONDARY RESPONSIBILITY

Figure 10. – Propulsion System – Airframe Aerodynamic Integration – Categories of Development Testing

Performance-Oriented Nacelle Aerodynamic Development

Figure 2 shows the major elements of performance oriented nacelle aerodynamic testing for an example development program. The testing proceeds chronologically along the following general lines: scale model tests of isolated inlet and isolated exhaust systems, tests of a flow-through nacelle on an aircraft model, tests of a powered nacelle on an aircraft model, sea level and simulated altitude static engine exhaust system tests, engine flying test bed flight tests, and production aircraft flight tests.

In a new airplane-new engine program, wherein development is occurring simultaneously, schedule is the largest single challenge. It is, therefore, of paramount importance that development testing of major components such as nacelle, inlet, and exhaust be underway at the time of airplane program go-ahead. This will generally be the case inasmuch as preliminary design, analyses, and supporting tests will have been in progress in anticipation of program go-ahead. Program go-ahead will initiate an acceleration in the configuration development testing in order to meet required schedules for prototype and production hardware.

Isolated inlet and exhaust model tests are conducted by the airframer and engine company, respectively, early in the program. Both isolated low speed and high speed tests are performed to determine inlet total pressure recovery and drag. A typical inlet wind tunnel model installation is shown in Figure 11.

Isolated exhaust model tests are conducted using a blown model where externally-supplied high pressure air is used to simulate the exhaust flow. A typical exhaust wind tunnel installation is shown in Figure 12. Figure 13 illustrates an example test result of the effect of external flow on fan nozzle flow coefficient obtained from an early model test.

Another example of an early isolated exhaust/aftbody test is shown in Figure 14, wherein shadowgraph flow visualization illustrates the sensitivity of fan external flow field to a modest modification in exhaust/aftbody contour. An associated 3/4 percent improvement in engine gross thrust was realized, which yielded a 2-1/4 percent improvement in net thrust in this example.

The results of isolated component tests and concurrent preliminary design studies form the basis for a preliminary baseline nacelle configuration. Initial tests of this early baseline nacelle configuration on a scale model aircraft are with a flow-through nacelle. The flow-through nacelle, of course, does not simulate the exhaust streamtube of the engine. It does, however, provide a cost effective means for

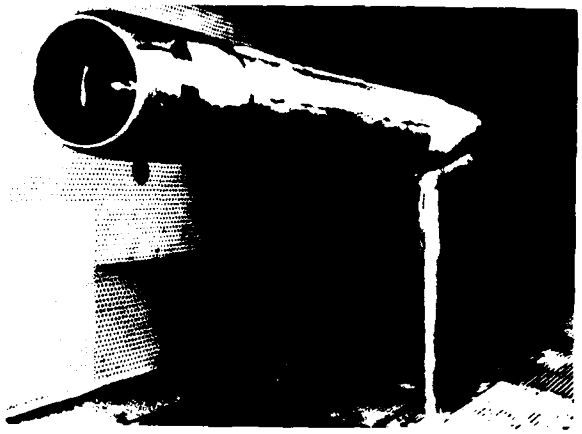


Figure 11. - Nacelle Inlet Model Tests

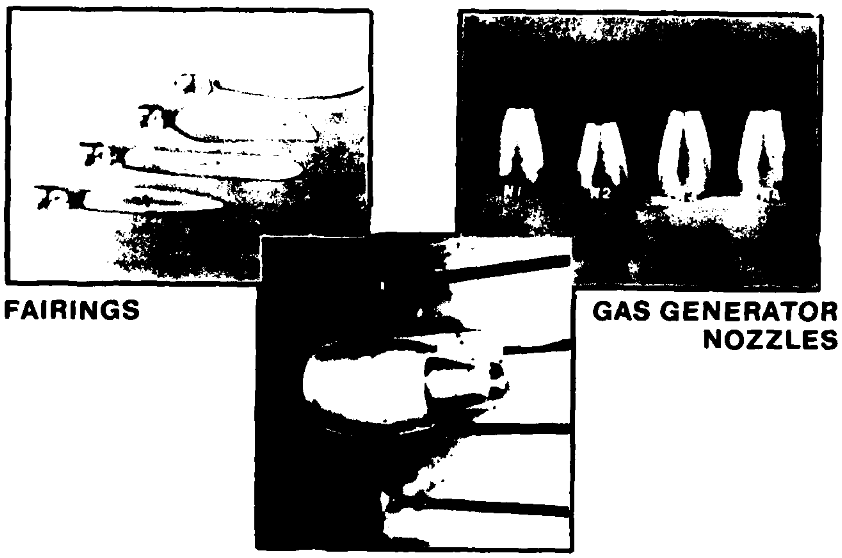


Figure 12. Engine Primary Aftbody Model Tests

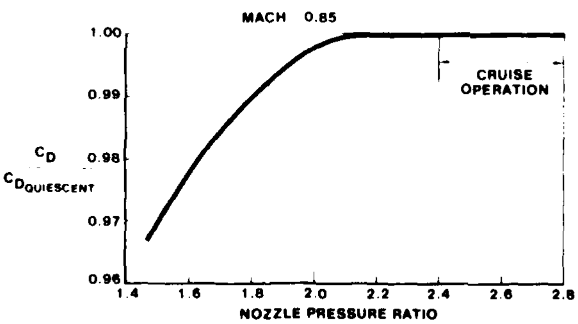


Figure 13. External Flow Effect on Nozzle Discharge Coefficient

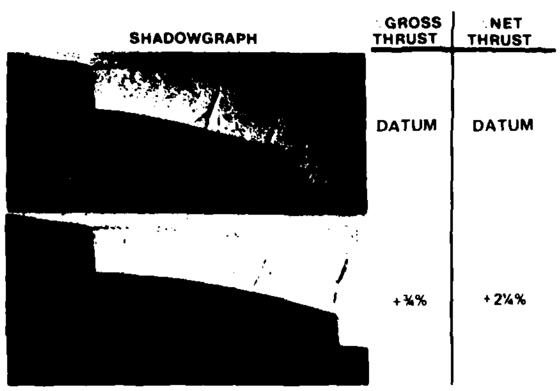


Figure 14. Nozzle Aftbody Contour Development

early studies of nacelle wing flow field interactions and thereby for preliminary nacelle aerodynamic studies by perturbations around a baseline configuration. The configuration perturbations chosen for testing are influenced by parallel analyses of trades between external and internal propulsion system performance and weight.

Having established a preliminary baseline nacelle and pylon configuration on the basis of isolated inlet, exhaust, and aircraft/nacelle flow-through models, powered nacelle testing is initiated. A powered nacelle model is illustrated in Figure 15.

Powered nacelle model development testing is a particularly important part of the nacelle aerodynamic development in that it provides exhaust flow pressures and streamtubes representative of the full-scale application in the critically important region of exhaust/pylon wing flow interactions, while introducing only a modest compromise in inlet streamtube area as illustrated in Figure 16. Experience has shown that small changes in exhaust/aftbody/pylon configuration can affect airplane performance by several percent. Accordingly, powered nacelle testing is exercised early in the airplane development program. This consists of testing a baseline (based on the preliminary isolated components and flow-through nacelle tests described above) and a number of configuration perturbations to optimize performance. Examples of powered nacelle testing are illustrated in Figure 17. For configurations wherein the aerodynamic interference can be shown to be small, blown nacelle testing could be substituted for a portion of the powered nacelle testing.

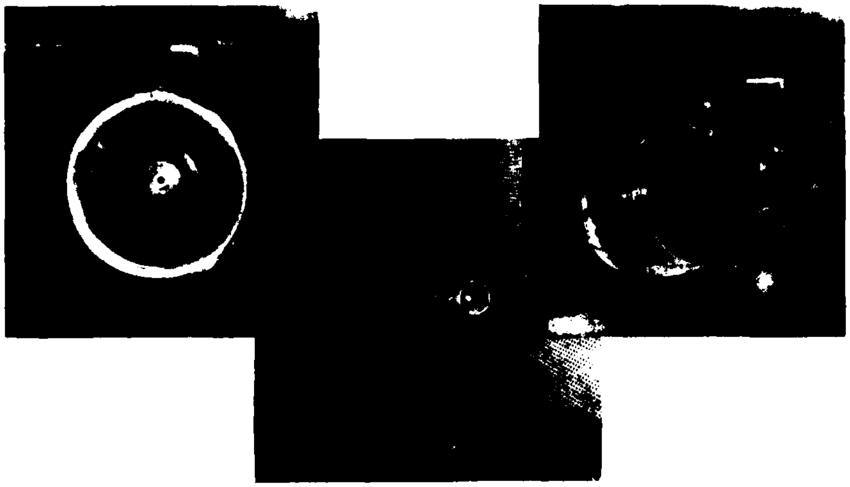


Figure 15. – Propulsion System – Airframe Interaction Model Tests (Turbine Powered Simulator Testing Technique)

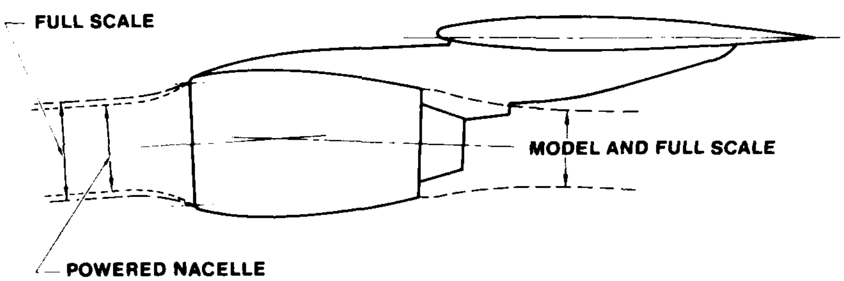


Figure 16. Powered Nacelle Simulation Inlet and Exhaust Flow

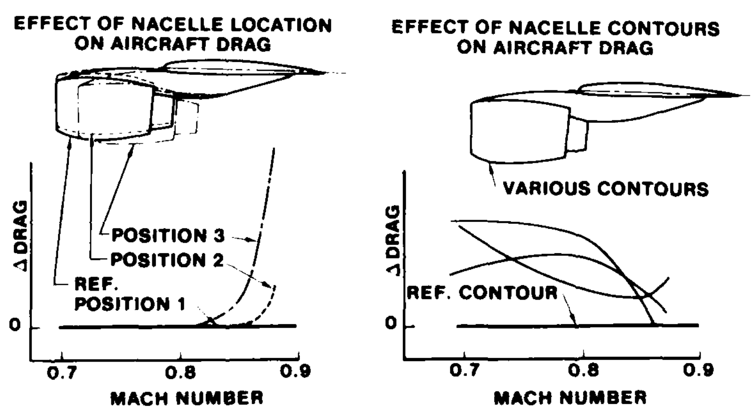


Figure 17. Effect of Configuration Perturbations Cruise Powered Nacelle

The importance of powered nacelle testing is further underscored by the necessity of selecting the production exhaust/aftbody/pylon configuration based largely on these tests. The selection may be made on the basis of carefully measured force increments (as contrasted to absolutes) between configurations which are confirmed in trend by analyses of pressure distributions in the wing/nacelle/exhaust flow regions. In the final analysis all sources of data will be examined and correlated including isolated inlet, isolated exhaust, flow through nacelle, and powered nacelle test results.

While the flying test bed is primarily for functionally oriented engine and pod systems testing and only approximates the flow field of the actual aircraft, it nevertheless provides timely data on some aspects of aerodynamic integration. In particular, it provides the first engine-nozzle matching data for full scale wind-on conditions. Prior small scale model tests include wind-on operation but full scale ground and altitude facility tests of the engine with exhaust nozzle are, of course, wind-off. Similarly, the flying test bed provides the first full-scale in-flight inlet pressure recovery data. A photograph of the VC 10 flying test bed used in the L-1011-RB 211 development program is shown in Figure 18.

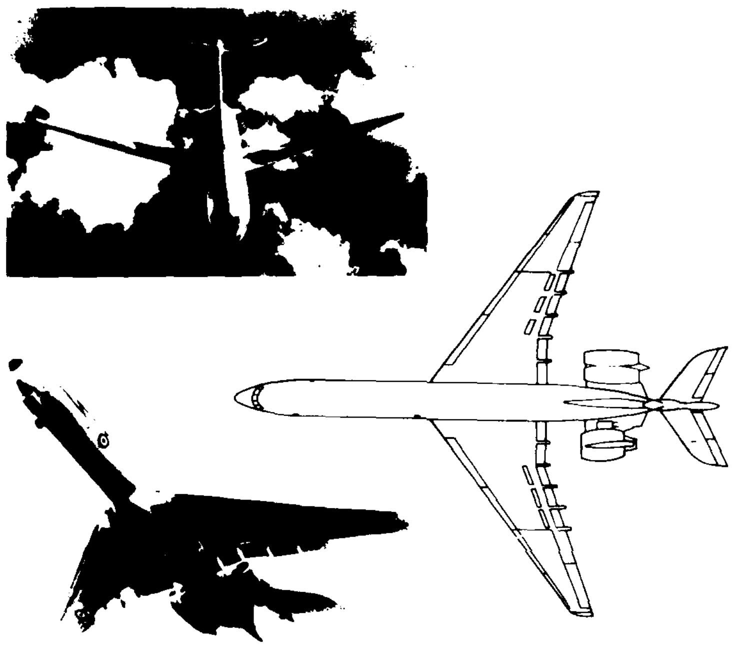


Figure 18. - Flying Test Bed

In order to properly understand the propulsion system airframe aerodynamic interactions, static pressure surveys will be made during the production aircraft flight test program on the nacelle, pylon, engine aftbody and wing lower surface in the flow channel area bounded by wing and nacelle. These surveys are of principal interest in the altitude cruise condition. The pressure measurements obtained are normalized and compared with scale model powered exhaust wind tunnel data as illustrated in Figure 19. The purpose of these comparisons is to determine whether any significant Reynolds number effects exist relative to the scale model data.

The end result of propulsion system airframe aerodynamic integration development is the demonstration of net performance on the production aircraft in flight test. Examples of results of this nature are illustrated in terms of improvements in specific air range for various exhaust aftbodies in Figure 20.

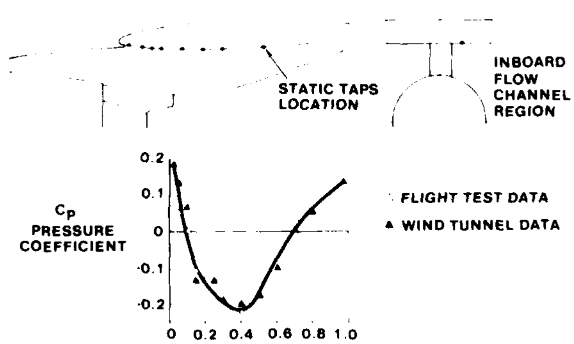


Figure 19. Flow Channel Pressure Survey Cruise

Functionally Oriented Aerodynamic Development

Inlet-Engine Compatibility. Inlet-engine compatibility is one of the schedule critical development areas in any new airplane - new engine program. Compatibility testing consists of three time schedule oriented phases (illustrated in Figure 21), namely, (1) wind tunnel model inlet distortion and calculated/estimated compatibility, (2) screen generated distortion testing with engine and (3) inlet-engine compatibility testing with actual inlet and engine. The compatibility evaluations (Figure 21), become progressively more realistic with each successive phase of the testing program.

The Phase 1 effort consists of a relatively crude look at inlet-engine compatibility. It is based on scale model inlet test data and the engine company estimate of fan and compressor distortion tolerances. The inlet may not be the final configuration, and the distortion tolerance estimates will be based on rig or research tests which only approximate the final fan and compressor configurations. Accordingly, Phase 1, comparing inlet distortion indices with fan/compressor indices, represents a preliminary but necessary evaluation of compatibility. This exercise is repeated as the inlet and fan/compressor configuration development proceeds.

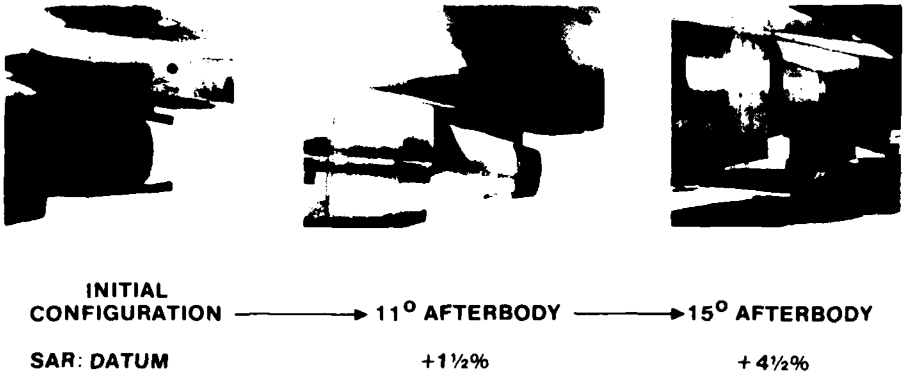


Figure 20. Engine Primary Aftbody Development

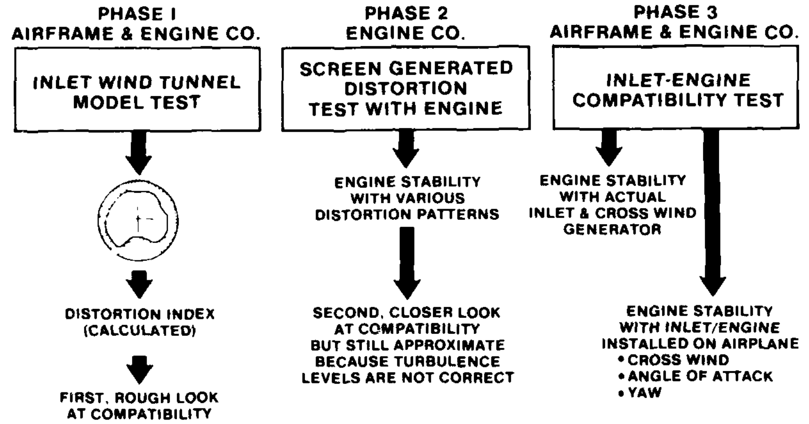


Figure 21. Inlet-Engine Distortion Compatibility Test Requirements

The Phase 2 effort consists of full scale distortion testing of a development (experimental) engine. The distortion is generated by various screen arrangements in the duct ahead of the engine. Fan and compressor flow stability and performance are determined for a wide range of inlet flow distortion patterns, including the inlet flow distortion patterns supplied by the airframe company based on scale model inlet tests. These patterns simulate cross-wind, takeoff, aircraft stall, and cruise conditions. The engine company may elect to test certain screen distortion patterns more severe than any supplied by the airframe company in order to obtain the stability limits of fan and compressor.

Although Phase 2 is a very important and necessary element of the inlet-engine compatibility development program, it still represents an approximate simulation of the actual flow because the screen generated distortion does not provide the turbulence levels that will be encountered by the engine with the actual inlet. The levels of flow turbulence associated with actual inlet pressure distortion can have significant effects on fan and compressor stability. It is, therefore, of crucial importance to initiate Phase 3 testing at the earliest possible time. Phase 3 consists of two major categories: (1) ground testing an engine with the full scale inlet using crosswind generators and (2) flight testing the engine with the production inlet in (a) the flying test bed and (b) the actual flight test aircraft.

The ground testing consists of operating an engine over a wide range of steady-state power settings and wind directions including crosswinds, headwinds, and tailwinds. Throttle bursts and retards are accomplished at key crosswind and quartering wind conditions. The simulated crosswind velocities range from approximately 20 to 60 miles per hour. Stable (surge free) engine operation during this testing provides a high confidence level in inlet-engine compatibility. The ground testing part of the Phase 3 inlet-engine compatibility testing can be accomplished at a relatively early date by using a development engine along with a development inlet. Both the engine and inlet approximate the production configurations. If any differences in configuration occur between this test and the production inlet and engine, the test would be repeated.

In the flying test bed part of Phase 3, the pod is placed in a flow environment (local Mach number, local flow angle, etc.) which simulates, insofar as practical, the flow environment of the production aircraft. Accordingly, during the flying test bed phase it is possible to approximate extreme operating conditions such as aircraft stall (high local angles of attack), large aircraft yaw angles, etc., which will be of the same order of magnitude as those of the production aircraft. Again, a high degree of confidence in inlet-engine compatibility will emerge as a result of this testing.

The definitive test of inlet-engine compatibility consists of demonstrating engine flow stability with the production inlet and engine on the production flight test aircraft. As part of this program, the inlet is usually fitted with a rake consisting of a large number of pitot tubes (60 to 80) just ahead of the engine face.

The ultimate test consists of demonstrating the capability of the engine to operate in a stable manner during extreme operating conditions, such as aircraft stall, maximum operational yaw, and high crosswinds and quartering winds. This demonstration is, of course, accomplished during the flight test certification program. As a part of the inlet-engine compatibility testing, a few key rake probes will be instrumented with fast response transducers to determine both externally and internally generated dynamic distortion. The inlet distortion rake data will be examined in terms of (1) the basic flow pattern characteristics and (2) distortion indices which will be compared with the engine allowable indices.

Two examples of inlet-engine compatibility test results are illustrated in Figure 22, namely, (1) an engine face total pressure recovery pattern at aircraft rotation and (2) in-flight inlet distortion index expressed as a percentage of the engine allowable limit. Both the basic distortion patterns and the comparison of inlet and engine indices become part of the final demonstration and substantiation of inlet-engine compatibility.

Thrust Reverser Effectiveness and Airframe Aerodynamic Interactions. Thrust reverser effectiveness (measured reverse thrust as a percentage of nozzle gross thrust in the forward thrust mode) is the prime responsibility of the engine company as indicated previously. The purpose of the present discussion is to outline broadly the efforts involved in achieving a target level of effectiveness. These efforts progressively lead from basic reverser into the more sophisticated phase of reverser development, namely, reverser-airframe aerodynamic interactions.

In anticipation of airplane program go-ahead, the engine company will conduct scale model tests of axially symmetric reverser cascades to determine reverser cascade thrust effectiveness and flow coefficients. Accordingly, force and flow measurements will be made. In addition, sufficient pressure measurements will be made to check the estimated turning losses used in preliminary reverser performance calculations and to relate turning losses and force measurements of reverse thrust.

Prior to program go-ahead, the engine company and airframe company will agree to a target level of thrust reverser effectiveness. The airframer will also supply estimates of the flow patterns from the reverser which are tolerable in terms of interference with airplane control and effects on lift and drag.

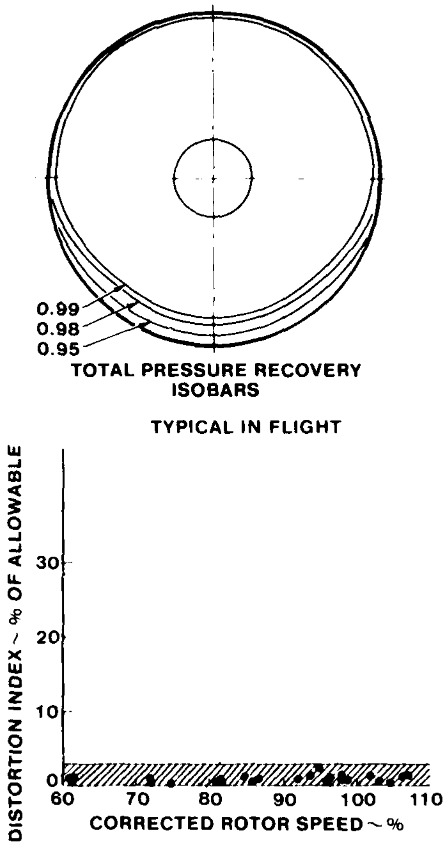


Figure 22. Inlet-Engine Compatibility

The next step in thrust reverser development testing by the engine company will involve blocking various areas in a scale model fan reverser to produce the tolerable flow patterns estimated by the airframer. This testing will indicate the degradation in reverser effectiveness due to cascade blockage.

Very early in the reverser development program, the airframe company will be testing a scale model of the complete airplane. Initially, testing with relatively small blockage of the reverser cascades will be done to determine effects of the reversers on airplane lift, drag, and stability and control for (1) ground operation, (2) in-flight, but in ground effect, and (3) in-flight. A number of reverser blockage patterns will then be tested. Photographs of a 1/20 scale wind tunnel model of the L-1011 used in wing and center engine reverser development testing are shown in Figure 23.

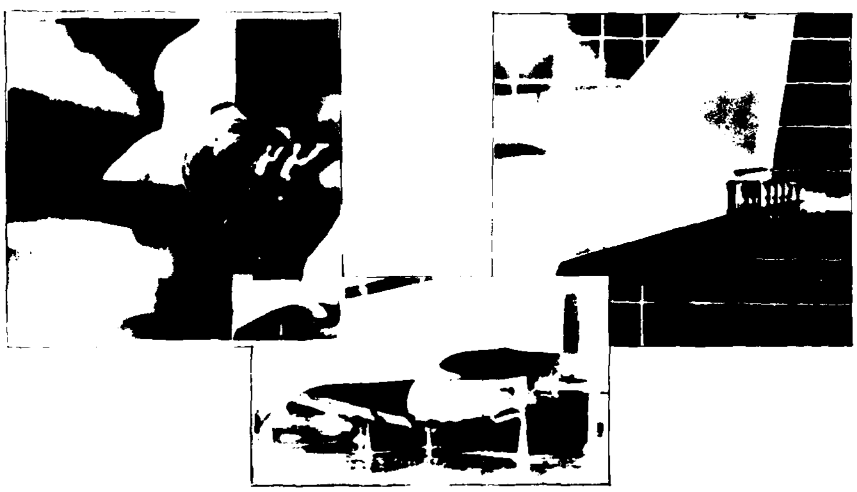


Figure 23. Thrust Reverser – Aircraft Interactions – Model Tests

In an essentially parallel effort, the engine company will be testing the more promising reverser blockage patterns from the airframer's wind tunnel model tests to obtain the effects on thrust reverser effectiveness. It is clear that thrust reverser development is an iterative process.

Flight tests will be conducted to check the thrust reverser performance. This will involve aircraft landing tests wherein the reverser is applied after landing gear touchdown and aircraft deceleration is measured during the landing run roll out. Knowing the aircraft drag from the previously described ground deceleration testing, the rolling friction coefficient, and aircraft weight, the net reverse thrust deceleration force is calculated. The results of this calculation are then compared with the predicted reverse thrust during the landing run roll out.

Foreign Object and Hot Gas Ingestion – Foreign object ingestion and hot gas ingestion during thrust reverser operation require attention reasonably early in the program to ensure a well integrated propulsion system-airframe combination and can influence the selection of reverser cascade pattern configuration. The engine company has prime responsibility in this development effort.

The airframe and engine companies will agree to a thrust reverser cancellation speed by the time of airplane program go-ahead. Major objectives such as thrust reverser effectiveness, airframe reverser aerodynamic interactions, structural loads, and back pressure distortion on the fan will have established the general reverser cascade flow patterns fairly early in the propulsion system development program. The engine company will then conduct scale model tests of a complete airplane model over speeds ranging from zero to approximately 100 miles per hour. Two tests will be performed: (1) simulated foreign object ingestion and (2) simulated hot gas ingestion.

In the simulated foreign object ingestion tests, flow visualization is achieved by using smoke or steam for the reverser flow medium. With the aid of this technique the degree of reverser flow penetration into the freestream can be visualized as a boundary on the wind tunnel floor (simulating the runway). The boundary moves forward at low speeds and aft at higher speeds as illustrated qualitatively in Figure 24. Small particles of appropriate scale are introduced along this boundary and the number of particles captured in the engine inlets are recorded. This test gives an approximate indication of the reverser cancellation speed for low probability of foreign object ingestion.

In the simulated hot gas ingestion tests, steam or hot air is used as the reverser flow medium. Thermocouples are placed at key areas in the inlets and around the nacelles, wing, fuselage, and landing gear. In general, at the reverser cancellation speed, hot gas ingestion is not a problem for current technology high bypass engines having conventional installations. The impingement of reverser air on surrounding aircraft surfaces has also not posed any problems.

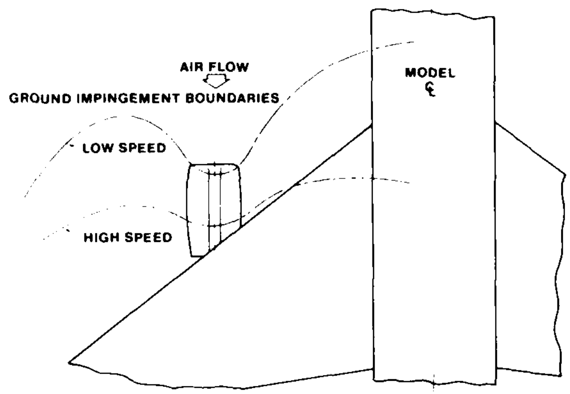


Figure 24. – Thrust Reverser Model Testing – Foreign Object Ingestion

CLOSING REMARKS

The highlights of a propulsion system performance development and demonstration program for a next generation aircraft and engine combination have been presented. Testing requirements and associated rationale were discussed for (1) basic engine performance and (2) propulsion system integration with airframe.

For a new aircraft-new engine combination, both incorporating advances in technology and undergoing development simultaneously, schedule is the single largest challenge. This follows from the fact that development of the full performance potential is an interactive, iterative process involving propulsion system and airframe. While the development of individual components proceed in parallel, the basic schedule challenge is one of developing/refining major components while simultaneously including the interactive effects between components. This involves innovative planning to ensure that the level of design/performance development remains consistent among the major components as the overall aircraft-engine development program proceeds.

This challenge can be met successfully by implementing (1) substantial preliminary design, analyses, and testing prior to program go-ahead and (2) extremely close coordination including planning and scheduling between airframe and engine companies in the areas of engine performance and propulsion system-airframe integration.

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DISCUSSION

M.Mihail, Bureau Veritas, Fr

(1) What is your opinion on the accuracy of the measuring instrumentation used on aircraft as compared to that used on the test stand at the time of certification or when the engines are delivered?

(2) Is there a correlation between on-the-test-stand vibration and on-aircraft vibration concerning the same engine? You spoke of maximal values. Do you think that, given the accuracy of the instrumentation which you mentioned, they are identical in both cases? This is a more specific instance of my first question.

Author's Reply

(1) In response to your first question, the accuracy of instrumentation on aircraft and on the test stand are very comparable in my opinion. We take great care in the calibration of the flight test instrumentation. The engine companies have, over the years, developed similar techniques so that it is my belief that the instrumentation accuracy on the aircraft and the test stand are very similar. I think the best check that one can get on the accuracy of any experimental data is to have several different sources of data and compare the results.

(2) Regarding your second question I did not discuss vibration in my talk. This question is out of my field of specialization and, therefore, I will leave it to the dynamicists.

CONTROLE DES PERFORMANCES ET ESSAIS DE QUALIFICATION DES MOTEURS CIVILS DANS LE CADRE DES ESSAIS AVIONNEURS

G. THERON

SNIAS - DIVISION AVIONS

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RESUME

La rentabilité économique d'un avion civil bi-réacteur repose pour une grande part sur ses performances de décollage ; il est donc très important de connaître l'évolution de la poussée installée des réacteurs durant cette phase de vol.

Celle-ci est caractérisée par une utilisation particulière des moteurs qui impose à l'avionneur :

- l'étude des relations liant la poussée au paramètre de conduite principal du moteur, après avionnage, aussi bien aux régimes élevés qu'à ceux des ralentis et de moulinet,

- l'étude de l'évolution de ce même paramètre de conduite lors du décollage (à position de manette constante) liée aux phénomènes dynamiques et thermiques auxquels est soumis le moteur.

Les systèmes de régulation actuellement mis en oeuvre contrôlent imparfaitement après l'ajustement de la manette ce paramètre de conduite provoquant une dispersion de ses évolutions. Pour la prochaine génération de moteurs, les motoristes proposent des régulations plus performantes, ce qui devrait permettre d'améliorer les performances et la durée de vie des moteurs.

Notations

FN	: poussée nette
WFE	: débit carburant
N1	: vitesse de rotation du corps basse pression
N2	: " " haute presssion
EGT	: température sortie turbine basse pression pour PWA
	haute pression pour GE

Indices

S	: calculé à l'aide du programme de calcul de simulation du fonctionnement stabilisé du moteur,
C	: calculé à l'aide des mesures effectuées en vol,
F	: estimé à l'aide des valeurs calculées (C) et de l'analyse de la polaire.

INTRODUCTION

La recherche des caractéristiques qui permettent de déterminer les performances d'un avion au décollage est l'un des buts des essais en vol effectués par un avionneur. En effet, l'importance commerciale de ces performances, associée aux exigences de sécurité, impose qu'il possède la meilleure connaissance possible des caractéristiques aérodynamiques de la cellule ainsi que des poussées disponibles au cours de cette phase de vol.

L'estimation de la poussée est rendue délicate par le fait que le décollage est la seule phase de vol pour laquelle l'équipage ne contrôle pas à tout moment le paramètre représentatif de la poussée. La manette des gaz est amenée à la position qui permet d'obtenir le EPR ou le régime fan désiré à $M \simeq 0,1$, puis elle est laissée fixe durant le décollage. L'évolution du paramètre de conduite étant, pendant cette seconde phase, influencée par la mise de gaz et par la mise en équilibre thermique des composants du moteur.

Il a donc été nécessaire de définir deux types d'essais nous permettant :

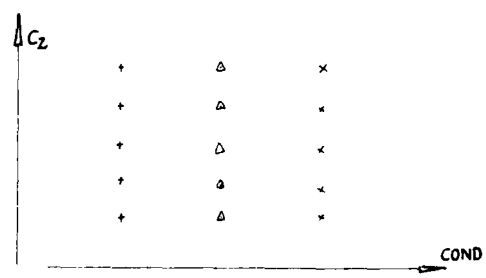
- de définir tout d'abord les relations liant la poussée au paramètre de conduite en fonctionnement stabilisé
- de définir enfin l'évolution de ce même paramètre de conduite au cours du décollage.

Ces essais permettront, d'une part de corriger le programme de calcul simulant le fonctionnement stabilisé du moteur et donnant ses performances, et d'autre part de déterminer l'influence des phénomènes dynamiques et thermiques sur la poussée.

1. - ETUDE DES RELATIONS LIANT LES CARACTERISTIQUES PRINCIPALES DU MOTEUR AU PARAMETRE DE CONDUITE EN FONCTIONNEMENT STABILISE

1.1 - Etude des régimes élevés

Dans des conditions de vol semblables et dans la même configuration aérodynamique de l'avion, sont effectués des essais balayant à la fois la zone utile de la polaire et les régimes utilisés par le moteur lors des décollages.



Durant ces essais, on enregistre les paramètres caractéristiques du moteur ainsi que les valeurs nécessaires au calcul de la poussée. Ces enregistrements nous permettent tout d'abord de comparer pour chaque point d'essais, les caractéristiques mesurées avec celles prévues par le programme de calcul, puis d'établir les corrections à apporter à ce dernier. (Voir Figure 1)

L'analyse de la poussée est plus délicate car elle n'est pas mesurée en vol mais calculée, d'une part à l'aide des pressions et des températures des flux, et d'autre part à l'aide des coefficients de tuyère estimés lors d'essais sur maquette et au banc statique. Mais du fait que les taux de détente trouvés lors des essais en vol sont généralement supérieurs à ceux qui existent au cours des essais effectués au banc statique, la précision de la poussée obtenue est liée à celle de l'extrapolation des coefficients de tuyère. Aussi est-il nécessaire de vérifier les poussées calculées à l'aide d'une approche utilisant les caractéristiques aérodynamiques de l'avion (Voir Figure 2).
Pour chaque niveau de poussée (ou valeur du paramètre de conduite) on détermine une polaire $Cx = f(Cz)$. On peut alors étudier l'évolution de ces polaires en fonction des diverses valeurs du paramètre de conduite.

Toute variation significative mettra alors en évidence une mauvaise estimation des coefficients de tuyère sur l'avion. Partant de l'hypothèse que la poussée calculée est correcte pour les taux de détente obtenus au banc, on définit la polaire de l'avion, puis on calcule les corrections à apporter à la poussée du moteur afin de rendre la polaire indépendante du paramètre de conduite.
Deux exemples de corrections définies après les essais en vol sont présentés sur la Figure 3.

1.2 - Etude des bas régimes

Afin de mieux déterminer les caractéristiques de l'avion,en particulier dans les phases d'envol, outre l'étude des poussées aux régimes élevés, il est également nécessaire de connaître avec précision la traînée de moulinet et les poussées aux bas régimes.

En effet, il est parfaitement possible d'effectuer les essais de mesure de pente de montée avec un moteur éteint, donc simulant la panne moteur. En revanche, tous les essais d'accélération monomoteur et d'envol peuvent uniquement être effectués avec un moteur au ralenti pour des raisons de sécurité.

Il sera donc nécessaire , lors de l'analyse des essais en vol, de retrancher la poussée de ralenti et d'ajouter dans l'établissement du Manuel de Vol la traînée du moteur éteint.
Pour déterminer la différence entre ces deux termes, nous effectuons, au cours du même vol lors de mesure de pente de montée, des essais balayant une polaire avec un moteur au moulinet puis avec ce moteur au régime ralenti. Si l'estimation des caractéristiques à bas régime a été correcte, les deux polaires seront confondues. Si cela n'est pas le cas, il faudra en admettant que la poussée au ralenti est correcte, par exemple, corriger l'autre pour obtenir leur coïncidence. On utilisera ces corrections pour dépouiller les essais en vol et établir le manuel de vol.

A l'aide de ces essais et de l'analyse effectuée, on dispose alors d'un programme simulant le moteur en fonctionnement stabilisé, et du moyen de corriger les essais effectués au ralenti. Il ne reste plus qu'à déterminer l'évolution de la poussée au cours du décollage.

2. - ETUDE DU DECOLLAGE

Lors du décollage,l'utilisation des moteurs actuels se divise en deux phases : la mise en gaz et le fonctionnement à angle manette constant. (Voir Figure 4).

Le pilote, après avoir lâché les freins, cherche à afficher en quelques secondes le régime nécessaire pour effectuer le décollage. Durant cette phase, l'évolution de la poussée dépend de la manière dont a été effectuée la mise des gaz. C'est pourquoi on prend en compte l'évolution qui correspond à la procédure préconisée dans le manuel de vol. En effet, toute variation par rapport à cette poussée n'a qu'une faible influence sur les performances de l'avion.

Après l'ajustement de la manette, le paramètre de commande (ou PLA) est stabilisé. Cependant, le moteur est soumis, d'une part aux inerties dynamiques et thermiques de ses composants provoqués par la mise de gaz rapide ; et, d'autre part, aux influences successives de l'accélération, puis de l'envol de l'avion. Aussi, l'évolution de ses caractéristiques diffère-t-elle de celle donnée par le programme de calcul simulant le moteur en fonctionnement stabilisé.

Afin de déterminer ces différences et d'évaluer la poussée disponible au cours du décollage, on effectue un nombre important de décollages pour chaque procédure de mise de gaz envisagée (manuelle et automatique) et pour plusieurs valeurs du paramètre à afficher.

Au cours de chaque décollage, les paramètres suivants sont enregistrés en fonction du temps : paramètre de conduite, angle manette, vitesse et altitude de l'avion, température ambiante. A l'aide de ces enregistrements, on détermine l'instant à partir duquel la position de la manette des gaz reste figée ; on définit ainsi le point d'ajustement. Le programme de calcul donne les caractéristiques stabilisées, l'angle manette qui correspond aux conditions de vol et à la valeur du paramètre de conduite enregistré à cet instant°. Il est ensuite calculé, à chaque instant du décollage, l'EPR ou N1 donné par ce programme pour les conditions réelles de vol et pour PLAo. On obtient ainsi l'évolution stabilisée qu'aurait eue ce paramètre de conduite et on la compare à son évolution réelle. (Voir Figure 5).

Il est donc possible de déterminer pour chaque procédure de mise de gaz ainsi que pour chaque niveau de poussée, l'influence des phénomènes dynamiques et thermiques sur l'évolution du paramètre de conduite. Le niveau et la dispersion des écarts obtenus nous permettent de choisir la procédure de mise de gaz la plus appropriée au moteur considéré, et de définir l'influence du niveau de poussée.
La Figure 6 montre les évolutions obtenues sur deux types de moteurs, jusqu'à 2 mn après l'ajustement de la manette. Ces évolutions sont étudiées sur l'un des moteurs maintenu au régime de décollage, l'autre étant au ralenti, pendant 5 et 10 mn afin de simuler les conditions réelles de vol rencontrées lors d'une panne.

A partir de l'évolution du paramètre de conduite et des relations qui lient la poussée à ce dernier, définies dans la première partie, nous déterminons une loi :

$\Delta FN/_{FN} = f(t, COND_{aff})$ (où $COND_{aff}$ est la valeur du paramètre de conduite affichée) pour chaque procédure de mise de gaz retenue, manuelle ou automatique.

On choisit l'écart de poussée le plus pénalisant pour l'utiliser dans les calculs qui définissent les performances de l'avion en vue de l'établissement du Manuel de Vol. Les poussées disponibles sont alors déterminées comme suit :

- lecture de la courbe de conduite et calcul des caractéristiques du moteur à M = 0,1 à l'aide du programme de calcul stabilisé corrigé
- définition du PLA correspondant à ce point
- calcul des performances du moteur stabilisé pour les nombres de Mach et les altitudes nécessaires
- prise en compte de l'influence des phénomènes dynamiques et thermiques en fonction du temps lors du calcul des performances avions.

3. - REGULATIONS FUTURES

Ces essais, tout en permettant d'estimer la poussée disponible lors d'un décollage, mettent aussi en évidence une certaine dispersion des évolutions de l'EPR ou du N1, et donc de l'EGT, pour une même procédure de mise de gaz. Il résulte de celle-ci que pour la plus grande majorité des cas, l'EGT obtenu est supérieur à celui qui correspond à la poussée utilisée pour l'établissement du Manuel de Vol. Ce phénomène détériore prématurément les moteurs de façon inutile. Les progrès de l'électronique ont conduit les motoristes à définir des systèmes qui permettront, sur les nouveaux moteurs, de réguler directement le paramètre de conduite représentatif de la poussée.

Quelles que soient l'altitude et la température du terrain, le pilote amènera la manette des gaz à une position fixe pour le décollage. Selon le constructeur, l'EEC ou le PMC réguleront à tout instant l'EPR ou le régime fan à partir de lois fonctions de la température, de l'altitude et du nombre de Mach, rendant ainsi leur évolution répétitive. Ces systèmes, même s'ils ne suppriment pas les essais, toujours nécessaires pour évaluer les évolutions naturelles des moteurs lors de la conduite manuelle, contribuent à augmenter leur durée de vie et/ou la poussée utilisable.

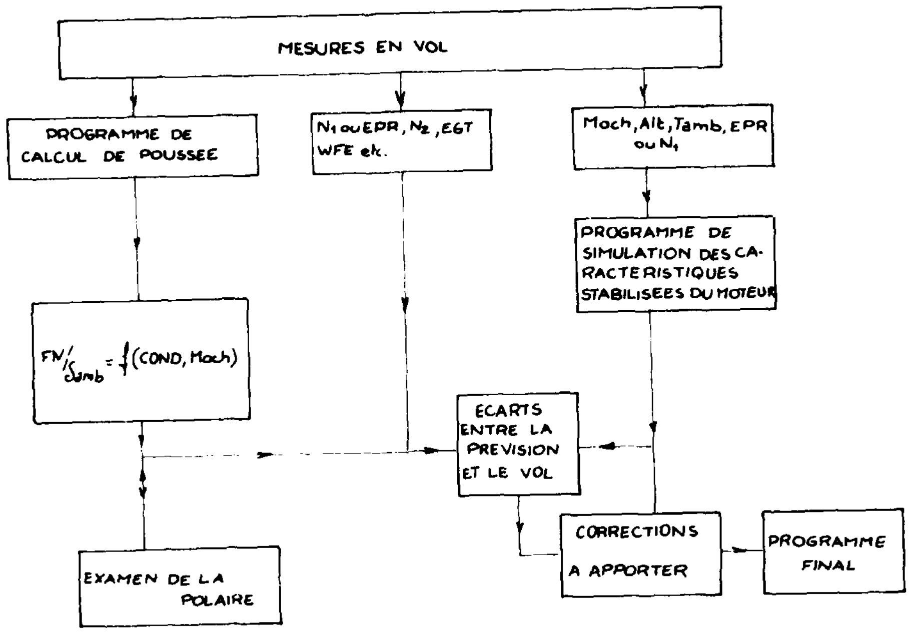


figure 1

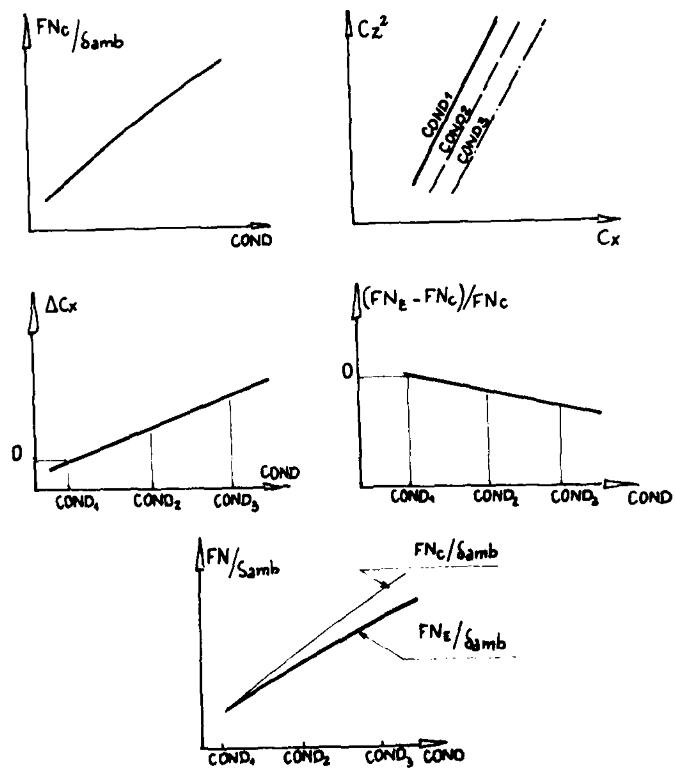


figure 2

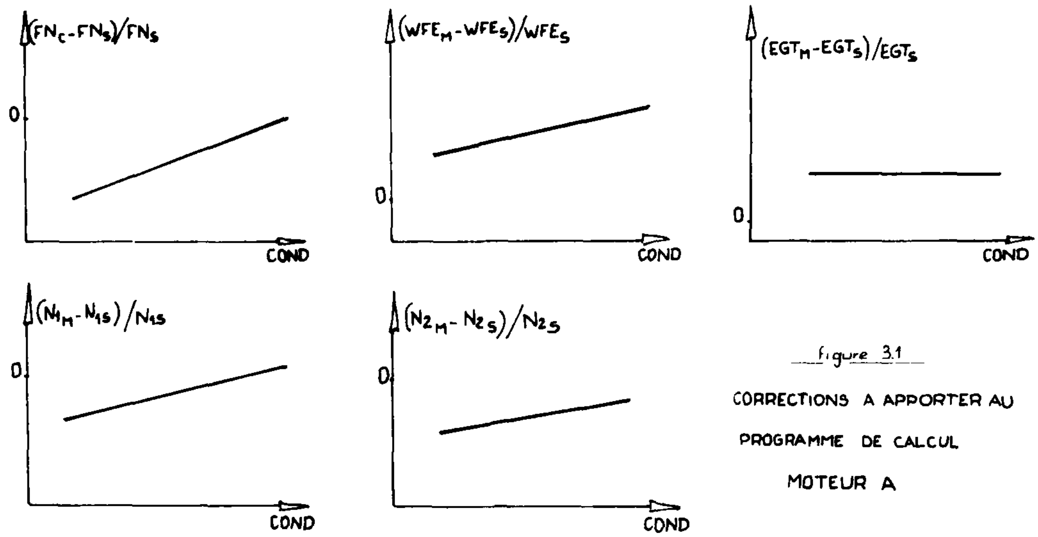


figure 3.1

CORRECTIONS A APPORTER AU PROGRAMME DE CALCUL MOTEUR A

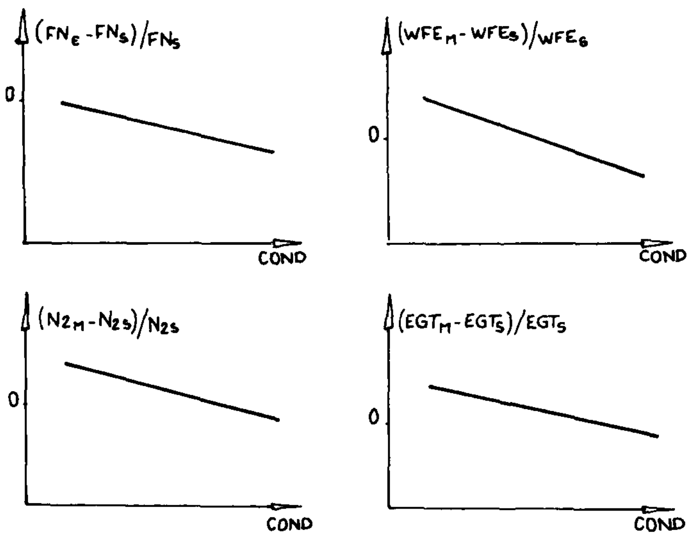


figure 3.2

CORRECTIONS A APPORTER AU PROGRAMME DE CALCUL MOTEUR B

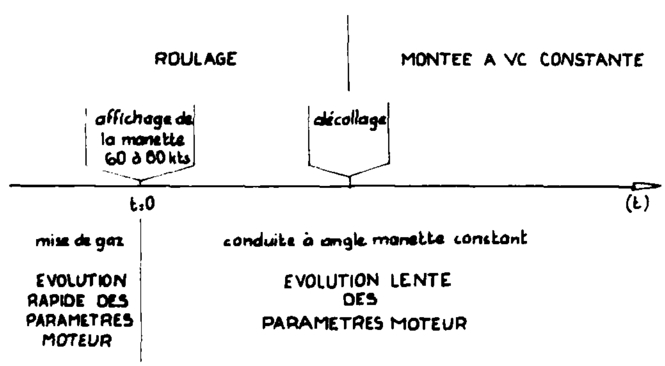


figure 4

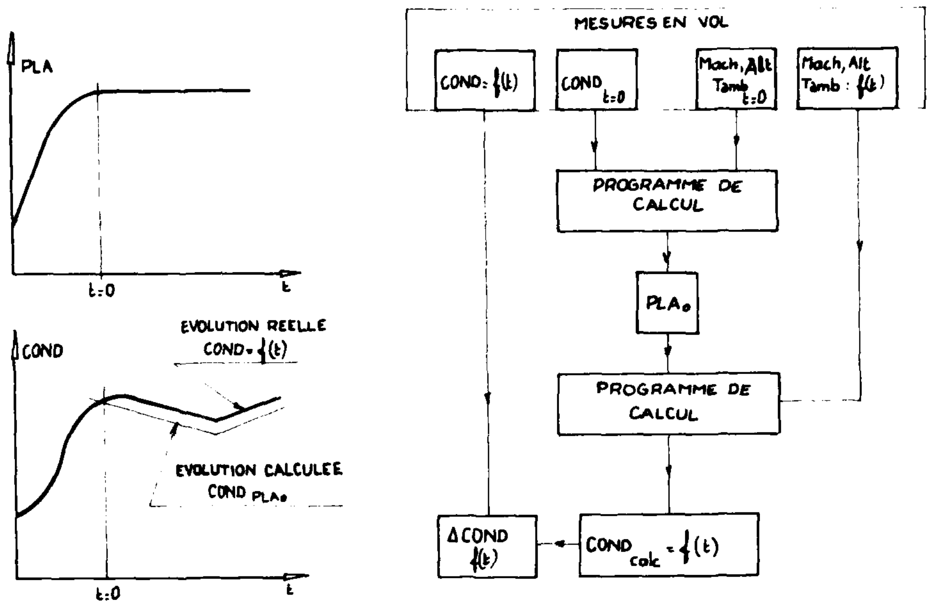


figure 5

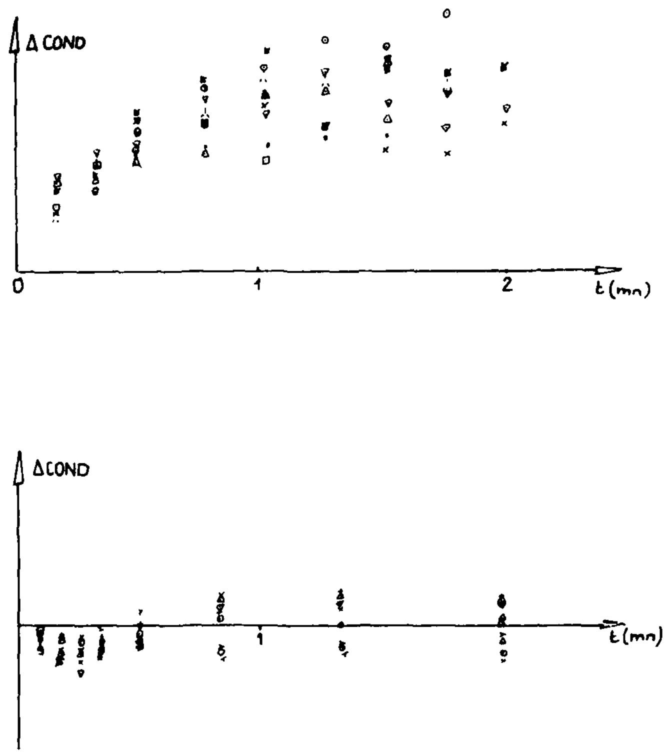


figure 6

DEVELOPMENT OF TEST REQUIREMENTS FOR CIVIL ENGINES

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SUMMARY

A summary is given of the objectives for the testing programme of modern civil transport engines. A typical modern testing programme is described covering timescale, test hours, number of engines and the different types of testing, such as performance, environmental, systems and mechanical testing. The effects on the programme of parts with long lead times, and the requirements for separate component testing are addressed. The testing conducted for manufacturer's purposes and that required by the certification authorities are compared, both for complete engines and for separate components and rigs. Typical test programmes in altitude test facilities and in flight are covered. Comparisons are drawn between modern programmes and associated techniques, and those in the recent past. The paper concludes with a look at the future, including the influence of new materials and the demands of higher fuel efficiency.

1. INTRODUCTION

This paper examines the Development Programme required to take a typical commercial fan jet from drawing board to Certification.

Before discussing the detail it is relevant to briefly review the background from which today's engines evolved and look in particular at the Rolls-Royce RB 211 which has provided the experience on which this paper is based.

The modern wide bodied airliner evolved to meet the need that became evident in the Mid '60's for an economical form of mass air transport which was compatible with existing airports and the communities around them. The engine manufacturers were faced with the challenge of a thrust requirement which was approximately twice that of the then current civil engines. Each engine manufacturer produced engines similar in size and cycle but differing in mechanical execution as befitted their differing background and experience. The Author's Company's solution was the Rolls-Royce RB 211 (Fig.1). This engine was marked from its competitors by having three shafts.

The two shaft gas generator is a design feature founded on experience from earlier engines which results in a compact engine with a short, very rigid carcase (Fig.2). This compactness provides for a more efficient integration of wing and propulsion system aerodynamics and forms the basis of the precise control of tip clearances between the rotating and static components which is so essential on high pressure ratio engines for minimum fuel consumption and good performance retention. The equal split of gas generator pressure ratio between intermediate and high pressure shafts minimises off-design compressor matching problems and 'variables' can be limited to a single stage of variable guide vane at entry to the IP compressor and low power bleed valves from IP compressor delivery and from the third stage of the six stage HP compressor. The RB 211 was the first Rolls-Royce engine to have 'modularity' designed into the engine at the earliest stages and has resulted in maintenance procedures which can significantly reduce unproductive 'down-time', both during commercial service and the Development Programme.

Since Service Entry in the Lockheed TriStar in 1972 at 42,000 lb. Take-off thrust (RB 211-22) the basic design has been developed to meet the further needs of the TriStar and the Boeing 747, evolving to a certificated thrust level of 51,500 lb. (RB 211-524 C2) with 53,000 lb. available in the near future. This has been achieved without increase in diameter and with a reduction in overall length, weight and specific fuel consumption. A version with a smaller fan diameter (the RB 211-535C), derived from the HP system of the -22B and with IP and LP systems closely based on those of the latest -524's is currently well advanced in its development programme. It will power the new Boeing 757 and gives 37,400 lb. Take-off thrust. (Fig.4).

It is this engine family and its development from the first run in April 1969 through to the current programmes on the -524 and -535 that form the basis of this paper.

2. SCOPE

The scope of this paper is defined as beginning at the completion of the Design Phase and finishing at achievement of the engine Type Certificate (Fig.5). It has been rather arbitrarily limited thus to maintain the task within manageable limits and, as such, addresses the phase which generates the most intense period of effort for the Development group. It commences with a Planning Phase where the opportunity has been taken to describe in some detail the types of testing required. This is followed by a Testing Phase where examples are given of particular tests. Post engine Certification, and hence outside this papers scope, is a phase of testing in the assigned aircraft which leads to Certification of the installed engine and aircraft in combination. This is usually accompanied by a programme of testing to demonstrate that the engine meets the terms of the commercial contract between engine and

aircraft manufacturer. This phase together with the engine manufacturer's calibration testing that precedes it would form the subject for a substantial paper in its own right, as would the final Development Phase, Service Support, which develops modifications for problems revealed in Commercial Service plus component life development. Mechanical and Aerodynamic rig test programmes parallel the Development period discussed and although very interactive with it are only considered briefly.

In th ise of all engines in the RB 211 family and many of their predecessors, Rolls-Royce has Design responsibility for the complete Propulsion System. This includes the intake, cowlings, thrust reverser, core engine fairings and nozzles - the nacelle - which in conjunction with the engine forms the Propulsion System. (Fig.6). This total responsibility leads to an unhindered integration of nacelle and engine design and development. It is therefore included in the paper.

Opportunities to plan and execute a Development Programme for a new engine type occur infrequently. In practice it is more usual to find that the programme requires the proving and Certification of a variant of an existing engine where perhaps one or more of the shafts incorporate significant change. Due to interactions it is still necessary to carry out many of the tasks that would be associated with a new design, however the risk of unexpected 'surprises' is lower. This paper is broadly based on experience of a number of such programmes and as such should provide a reasonably comprehensive review of a typical modern programme.

3. THE DEVELOPMENT PROGRAMME

It is convenient to divide the development task into two phases:-

'Planning' and 'Testing'. In describing the Planning Phase the opportunity is taken to cover in some detail the types of testing required. The Testing Phase is then given to description of examples of specific tests.

3.1 The Planning Phase

This Phase starts, typically, in parallel with the start of the main Design activity and becomes an ongoing task for the remainder of the programme. The Development Engineer will take as his basis the Design Specification a document which defines in some detail the design or design changes planned, and in conjunction with specialists from the Design and Technology staff, produce the Development Plan. This will address the following major items:

- Development test requirements.
- Development Engine Programme.
- Supply programme for Development engines and spares.
- Test Facility and Equipment requirements.
- Planned achievement of progress with 'milestones'.

Considering each in turn:

3.1.1 Development Test Requirements

The test requirements for the particular specification will be determined from the following typical list of test examples.

(a) Endurance and Cyclic

The former is testing to the 150 hr. endurance schedule specified by the Certification Authority for the Type Test, the latter simulates service operation between low and high engine power to a schedule chosen to gain experience in the shortest possible time compatible with the particular design aspect under review. It can include repeated engine starts and reverser operations. In combination this testing can normally be expected to rapidly reveal shortcomings in the durability of the new design features.

(b) Performance Development Testing and Noise

This testing anticipates a performance short-fall on the initial development engines. Even when the aerodynamic rig programme has demonstrated achievement of aerodynamic targets it is normal to find that development of the initial design is required to reduce clearances and leakages to design values, to correct gas annulus mis-matches between static and rotating components and to check out the effect on the engine performance of modifications that will be required for mechanical or functional reasons. A typical programme would have two Performance development engines. One engine would continually assess the effect of design changes on performance by frequent tests to assess individual or packaged modifications, with the emphasis naturally on continual improvement. This engine would be heavily instrumented to enable the performance of individual components to be assessed. The second engine would typically be maintained to the highest possible mechanical and aerodynamic standard to demonstrate the absolute performance level, thus serving as an accurate guide to the performance of the initial production engines. It's programme will require periodic calibrations in an Altitude Test Facility that is capable of simulating the range of altitude and Mach No. of the aircraft concerned. Testing at simulated cruise conditions will predominate. Additional tasks will include testing to establish optimum nozzle areas, the effect of air bleed on performance and performance under windmilling conditions. A detailed assessment of the engine noise characteristics will be conducted on an open air test stand specially designed to minimise extraneous

affects due to the ground or surrounding buildings. It is convenient to combine this test with an 'Open Air' engine performance calibration in both forward and reverse thrust. No corrections to the measured thrust are required so the test can be used to develop the corrections that have to be applied to measurement of thrust on the enclosed test beds. Both Noise and Open Air performance tests present difficulties, for both if they are to generate accurate results demand dry, still air. These are realised infrequently in the notoriously changeable English climate.

(c) Functional Testing

Functional testing is a 'catch all' title for the multitude of tests required to examine and confirm or correct the design requirements of the mechanical and operational functioning of the engine. Engines conducting such tests tend to be heavily instrumented, to have long and complicated builds and short duration tests. To be successful this type of test demands close and thorough liaison between the development engineer, the technical specialists who contribute to the specification and carry out the detailed analysis of the tests, the instrument designers and the measurement engineers who are responsible for the correct functioning of the special equipment.

Examples of Functional Tests include:

- Straingauging for measurement of direct and alternating stresses on most new static and rotating components. The degree of equipment sophistication will range from the simple case of, say, an engine mounting link, to the complex case of measurement of vibrational stresses in HP turbine blades with the associated hostile factors of temperatures, pressure and centrifugal field, plus the need for very specialised lead out equipment.

- Thermal paint test for temperature surveys of all major engine components and in particular to develop the cooling of combustion liners and HP and IP blades and vanes.

- Pressure and temperature surveys to ensure the correct operation of internal cooling and ventilation systems and pressure balance around bearing housings.

- Measurements of the transient thermal response of discs to provide information for accurate stress analysis and clearance matching. The HP system is of prime importance and presents similar difficulties to the straingauge case.

- Assessment of the correct operation of the engine oil system, including the efficient operation of the oil supply to and scavenge from the engine main bearings and gearboxes, and assessment of the oil cooling and filtration systems and their controls.

- Measurement of bearing end loads.

- Assessment of the engine starting performance using a special 'Cold Room' test facility for simulating starting down to -40°C. and hot day starting to simulated inlet temperatures up to +55°C. Similar testing is conducted in the Altitude Test Facility to assess inflight relighting. All such testing may be accompanied by fuel system 'tuning' to develop optimum starting performance.

- Assessment of the engines vibration characteristics to form the basis of Operational Limits and to enable an endurance engine to be built to simulate the worst service conditions.

- Assessment of engine handling at sea level and simulated altitude conditions to ensure optimum operation of compressor bleed valves and/or variable vanes to provide stall free operation under the most extreme throttle manoeuvres. This will include tests to artificially raise compressor operating points by injection of fuel or special operation of bleed valves to provoke stall and hence assess operating margins and thus margin for in-service deterioration.

- Assessment of the correct functioning of the nacelle. This includes the thrust reverser and its operating mechanism and control system, the nose cowl anti-icing, fire detection systems and the various ventilation zones.

- The above list addresses the major requirements but no list can be comprehensive. Additional special tests may be required to assess damage and engine integrity when ingesting birds, hailstones, and high water concentrations, the effect of ingested dust and internal oil leaks on the purity of the air supplied for cabin conditioning, etc.

(d) Certification Tests

Airworthiness Requirements demand that the engine manufacturer demonstrates a prescribed standard of engine integrity to the satisfaction of the Authority concerned. Satisfactory demonstration is marked by the granting of a Type Certificate. This major landmark in an engines development is required to enable the engine to proceed to the nest phase - the Flight Certification - and thence to Commercial Service. The requirements are demonstrated from a combination of engine and rig tests and technical analysis based on the results of these tests and, where appropriate, analogy with previous similar designs.

The scope of this paper does not include a detailed consideration of all the tests involved. Comment is thus limited to the observation that data from the Functional Testing plays a major part in the preparation of the Certification submission. Specific engine demonstration tests include:

- The Type Test.

  This is a 150 hr. endurance test to a schedule defined by the Airworthiness Authority. It is designed to subject the engine to a full range of conditions to expose any weaknesses in the design. As such it includes periods of running at all power levels with particular emphasis on maximum values of shaft speeds, gas temperatures and oil temperature, and minimum oil pressure. The testing at steady conditions is conducted with and without air and power off-takes and is interspersed with accelerations and decelerations and cyclic testing. The test includes a prescribed number of starts and thrust reverser operations. Satisfactory completion of this test as established by the thorough examination of the dismantled engine after completion to the satisfaction of the Authorities inspectors is mandatory. The maximum values of shaft speed and turbine temperature demonstrated together with other limiting parameters become the Maximum Operational limits in Commercial Service. Great care is thus necessary in planning and executing this test to ensure that the conditions tested are adequate to enable the engine to deliver its full performance.

- Overspeed and Overtemperature Tests.

  These are tests to demonstrate the ability of the engine to operate for short periods at conditions above the maximum normal limits. They again provide clearance for Operational Limits.

- Cold starting demonstration.
- Interrupted oil supply demonstration.
- Windmilling without oil.
- Icing demonstration.
- Smoke emission compliance demonstration.

3.1.2 Development Engine Programme

The assessment of the test requirements for the specification in question will provide the basis of the Development Engine Programme. To determine the order of tests a judgement will be made of the likely risk areas in the new design features. This leads to the choice of key functional tests at an early stage in the programme. Examples are straingauge surveys of new discs and checks on the functioning of the cooling systems of the high temperature turbine blades and vanes. An initial performance assessment will have high priority. Incorrect aerodynamic functioning can lead to time consuming diagnosis and correction with a resultant delay before endurance experience starts to accumulate on the final standard. In the interim a poor performance standard may preclude the attainment of full rotational speeds on endurance tests with the possibility that mechanical problems may remain unexposed.

In parallel with the above testing endurance tests will reveal unexpected design weaknesses at the earliest opportunity. Endurance testing will be of two types. Early 150 hr. endurance tests will explore the capabilities of the design at the more severe conditions of shaft speed and temperature and will also ensure that the engine is operated for a period at all speeds within the operating range to expose any unexpected resonances. Additionally it will serve as practice running for the formal Type Test later in the programme. Early cyclic testing is particularly relevant to establishing the durability of the combustion chamber and high temperature aerofoils and has demonstrated that it can rapidly show up durability and reliability problems in all parts of the engine more rapidly than by running to the 150 hr. endurance schedule.

The above testing will be followed by more routine functional testing, by detailed performance tuning and by continual endurance tests (both 150 hr. and cyclic) to accumulate hours and cycles on specific sets of parts and to test design changes that have been shown necessary by earlier testing. In the final stages the emphasis will be on the formal certification tests required to justify the granting of the Type Certificate.

The resulting programme will establish the number of development engines required, and test dates for specific tests and the required build up of test hours and cycles. (Fig.7).

3.1.3 The Development Supply Programme

The information contained in the Development Programme is the basis for the Supply Programme. This will consist of new development engines or conversion kits, spare modules to ensure rapid shop turnround on selected engines and spares. It will establish when orders need to be placed to ensure timely completion. In the initial stages it will concentrate on long lead time items. This phase is interactive with the design programme where designs involving long lead time components must be specified, at least in sufficient detail for material orders to be placed, at an early stage if the elapsed time for the total programme is to be competitive. As the programme progresses the Supply Programme will concentrate on getting new engines built for test and then on supplying modifications and spares for successive builds. A successful Development Programme is extremely dependent on a well planned timely Supply Programme.

3.1.4 Test Bed and Special Test Facility Requirements

The Development Programme will establish the number of engines and rate of testing. This will be related to the required Test Facility capacity and support services on the basis of previous experience. Test facilities for engines of the RB 211 type are large and, extremely expensive. A test bed with its exhaust equipment is some 200 ft. long, 30 ft. wide and 35 ft. high and with its special instrumentation currently represents a capital outlay of £2.0 million and an annual running cost of £0.7 million. Utilisation must be high consistent with adequate capacity to meet the Development Programme.

5.1.5 Special Test Equipment Requirements

This can be categorised as special equipment for use in conjunction with the engine and special measuring equipment on and in the engine. Examples of the former include airmeters, inlet flares, slave exhaust equipment, special inlets for noise testing, ground handling equipment for the engine and its modules etc. The latter include pressure and temperature rakes and tappings, straingauge lead out facilities, bearing load measurement equipment etc. The lists are far from comprehensive and all must be designed and procured in a timely manner.

5.1.6 Progress Monitors

To check that the development programme is proceeding at the required rate it is usual to establish during the planning phase monitoring parameters with targets consistent with the required rate of achievement.

The prime monitors are the build up of engine test hours and cycles. These will be backed by selected technical milestones. Fig.8. The selection of the latter is dependent on the particular specification and experience of the difficulties expected based on previous experience of similar programme. Fig.9 shows the averaged opinion of sixteen senior engineers from the Author's Company of the difficulty of satisfactorily completing various aspects of development testing as defined under fifteen categories. The response covered all aspects of development including all the rig test tasks but remains a good guide to the difficulties expected in the development engine programme. Mechanical reliability is marginally the most difficult, closely followed by Performance, which reflects in particular the intense competitive pressure to improve fuel consumption.

5.2 The Testing Phase

This phase of the programme typically consists of two years of intensive engine testing. It is not practical to discuss this in detail so a selection of examples has been taken to illustrate various aspects.

Example 1 - The Type Test

The regulations require that all Take-off and Maximum continuous running is carried out at the maximum shaft speeds and turbine gas temperature for the rating in question. Unfortunately the engine characteristics when operated to the limits of an aircraft flight plan do not result in these parameters reaching their maximum values simultaneously. The problem is compounded by having three shafts. It is therefore necessary to change the matching of the Type test engine to achieve simultaneous maximum values. The matching adjustments required at Take-off and Maximum Continuous are different. The maximum turbine temperature is only used in Service on hot days, if the Type Test engine were run to this temperature on ordinary day temperatures and particularly in cold winter weather the pressures in the engine would be far above maximum service conditions. Ways have to be found to overcome these difficulties. The high pressure case is normally avoided by heating the inlet air to the test bed. This requires that some 2,000 lb/sec. of air be raised in temperature by up to 25°C. This is expensive and the equipment is inflexible. More latterly the problem has been overcome by introducing a pressure loss at engine inlet with a screen in the inlet ducting. Fig.10. The speed/temperature matching case is solved by special adjustments to the engine. Considering the turbine temperature as fixed at it's required value the HP spool speed is raised to it's maximum by twists to the compressor blading front stages. The IP speed is then raised to its maximum by positive adjustment of the variable inlet vanes. The fan is then raised to its maximum speed by enlarging the exhaust nozzle. The gas generator adjustments reduce it's efficiency which can result in insufficient gas horse power to drive the fan at maximum speed with the largest possible exhaust nozzle. In this case it is necessary to clear the shaft speeds on two separate tests. Allowance must also be made for a drop in all speeds at a turbine gas temperature as the test progresses due to deterioration and accretion of atmospheric dirt. These factors make the running of the Type Test at the correct maximum conditions a very challenging task and its successful completion an occasion of great relief.

Example 2 - The Nacelle Leakage Test

Complete sealing of the nacelle is of prime importance on an engine of the RB 211 Type. Air that leaks through nacelle seals and joints instead of going through the propelling nozzles is wasted thrust and hence degraded fuel consumption. Ten square inches of leakage area is approximately equivalent to one percent of s.f.c. The test consists of sealing the inlet and exhaust nozzles of an engine with blanks and pressurising the engine interior with slave air. The leakage rate is measured and then individual leaks traced and sealed to determine their values. Leakage above the specified level will require design action. This simple test has proved most effective in reducing fuel consumption and is an area where the Author's Company has gained from their responsibility for the complete propulsion system. (Fig.11).

Example 3 - Bearing End Load Measurement

Gas Turbine main shaft bearings experience an end load which is the difference between a large rearward turbine load and a large forward compressor load. This difference between two large quantities is difficult to calculate accurately. If the load capacity of the bearing is exceeded the service life will be reduced, if it drops below a minimum value the bearing will skid with the same result. It is thus necessary to measure the end load on each bearing. This is achieved on the more difficult intershaft bearing which locates the LP shaft by the provision of equipment which can load the IP shaft hydraulically. By raising the oil pressure the bearing is made to move from its stop. The bearing load can then be established from the oil pressure and the area it is bearing on. This will be repeated at conditions throughout the Speed range. (Fig.12).

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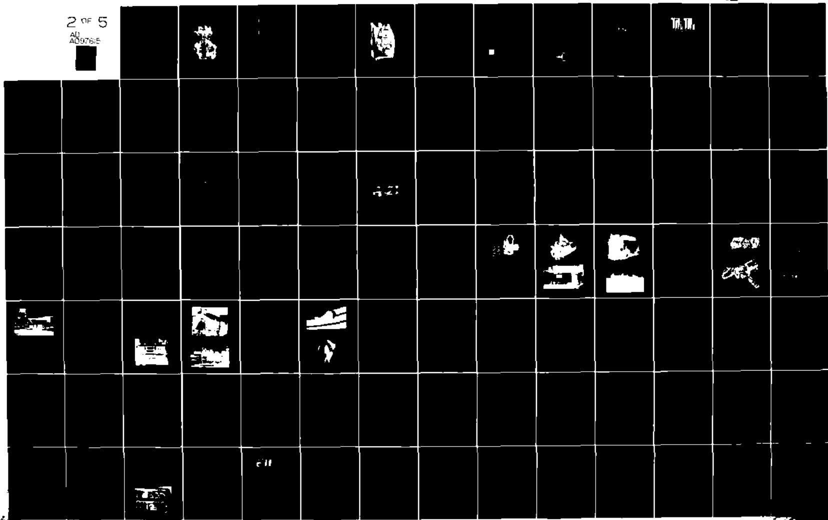

Example 4 - Measurement of Temperature and Stress on the HP Rotor

Measurements on the HP rotor present the most difficult challenge to the measurement engineer. It is necessary to transmit the signals from the rotating shaft (10,000 rpm) to a ground station. The environment is hostile (350°C) and space is limited. To be of practical value the equipment must be accurate over a wide range of conditions and very reliable. Fig.13 shows a system designed and developed by the Author's Company. It contains induction coupling rings for the power supply and control circuits and capacitive coupling for signal transmission. It can transmit data from a.c. or d.c. straingauges, thermocouples and thermal mats. Six transmission channels can be employed at any one time and by internal switching it has the capability of handling for example up to 64 thermocouples which can be calibrated on line. The unit is cooled to maintain acceptable operating temperatures. The ground station digitises the output signals and converts it to engineering units in a computer for on line presentation to the test engineer.

Example 5 - Cyclic Testing

Cyclic testing is used to expose design weaknesses. Fig.14 shows the cycle that has been developed to give the maximum experience for minimum elapsed time. This is important to both minimise the time required to expose problems and also to minimise the fuel bill which for an RB 211 is typically £750 per hour for this type of testing.

Example 6 - X-Ray Measurements

Performance is critically dependent on maintaining minimum clearances between rotating blade tip seals and the associated static members. The development of designs that meet this objective uses information of many types including theoretical analysis backed by measurements of engine pressures and temperatures. It is particularly aided by X-ray photographs. This technique which has been developed to have a capability of accurately showing the position of rotating blades under running conditions is illustrated by Fig.15 which shows an LP 1 turbine blade as photographed with the engine stopped and also at maximum conditions. With computerised analysis it is possible to accurately determine relative positions to within a few thousandths of an inch. The equipment is powerful to achieve the required penetration and safety precautions have to be rigorously observed.

4. CONCLUSIONS

The foregoing is a brief review of the intense period of development testing to Certification of a typical programme on one of today's high by-pass ratio fan jet.

It is conventional at this point to consider changes from past practices and anticipated developments for the future. Test practices have in fact changed little in basic principle over the last 20 years. The introduction of the large fan jet in the late 1960's required no major change to the Certification regulations. The step change in physical size with the attendant problems generated by the new test equipment required to accommodate it are now overcome. In some respects the increased size has advantages, for instance, providing more space for instrumentation. The progressive increase in pressure ratio with the accompanying increases in compressor delivery temperature and turbine entry temperature has required increased attention to turbine cooling systems and the introduction of new alloys, principally nickel based, with higher expansion coefficients than the ferritic alloys they replaced. This makes the control of clearances between static and rotating members more difficult at a time when increased stage loadings demanded that they be reduced. Development testing, backed by continually improving technical analysis has made a major contribution to the precise control achieved on today's engines. For the same reasons, thermal stresses, particularly in turbine discs, increased in significance and accurate life predictions have now become dependent on a precise knowledge of transient thermal gradients, again the development engineer has been required to provide accurate measurements. The introduction of computerised analysis techniques has led to quicker response to test results and the ability to plan more complex tests. The emerging environmental concern has led to new types of tests. Mandatory noise and smoke limits are established and today's planning now has to assume the introduction of exhaust pollutant limits. The above trends are expected to continue, perhaps at a diminishing rate. They will be accompanied by the introduction of new materials and manufacturing techniques which will include powder metallurgy for discs, superplastic forming for titanium alloys, novel casting techniques for turbine blades and increased use of composites, all driven by the competitive pressure to produce lower fuel consumption and reduce weight and cost. The development engineer will continue in his current major role in taking these innovations through to commercial service.

5. ACKNOWLEDGEMENTS

It remains for me to gratefully acknowledge permission of the Directors of Rolls-Royce Limited to present this paper and my colleagues support in its preparation.

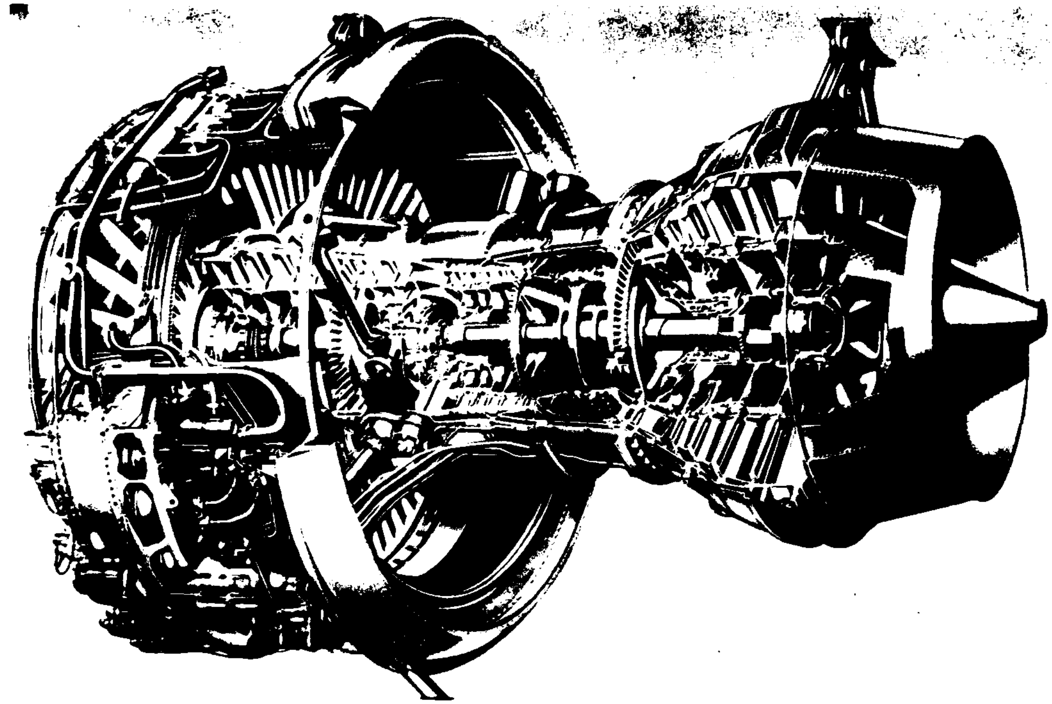

FIG.1 - ROLLS-ROYCE RB 211 TURBOFAN

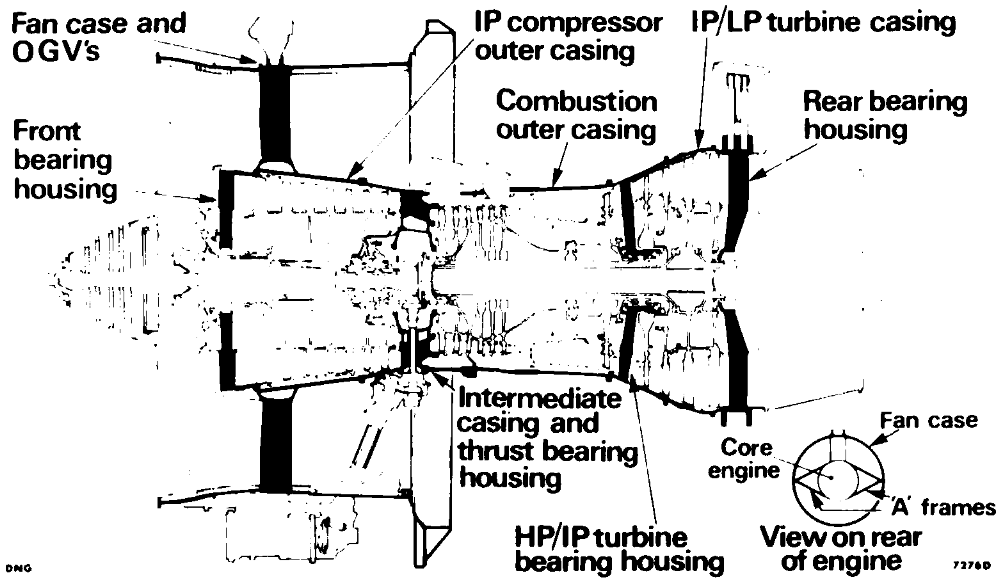

FIG.2 - RB 211 LOAD CARRYING STRUCTURE

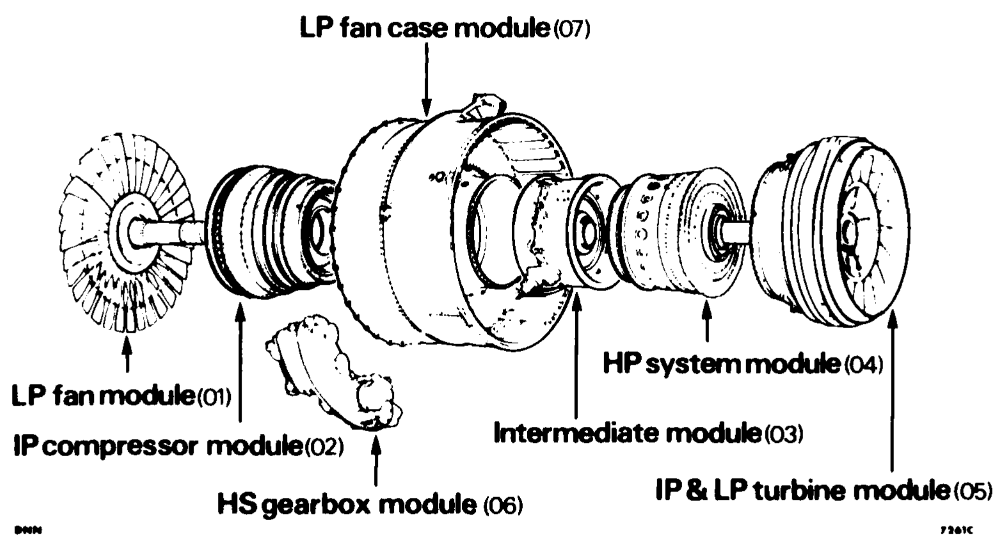

FIG.3 - RB 211 MODULAR CONSTRUCTION

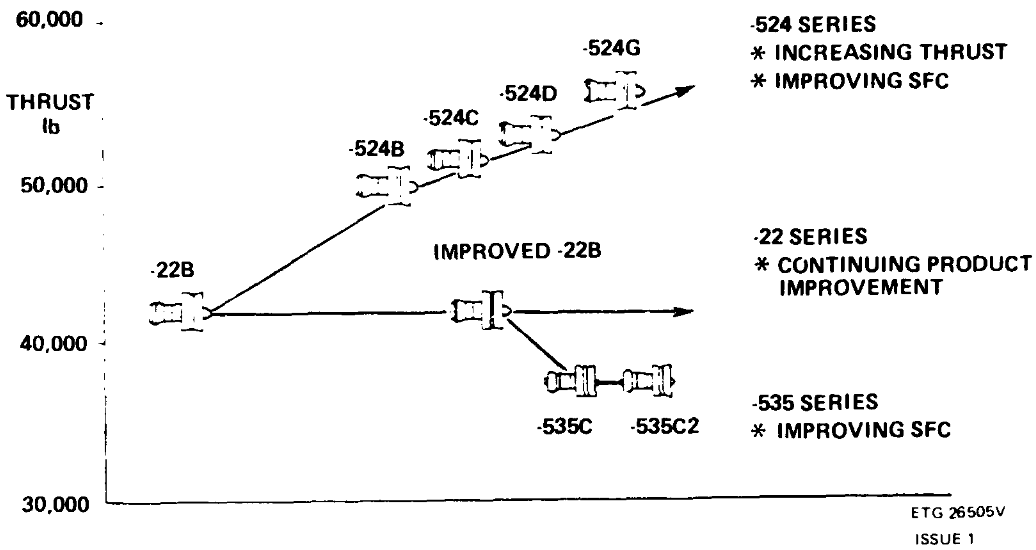

FIG.4 - THE RB 211 FAMILY

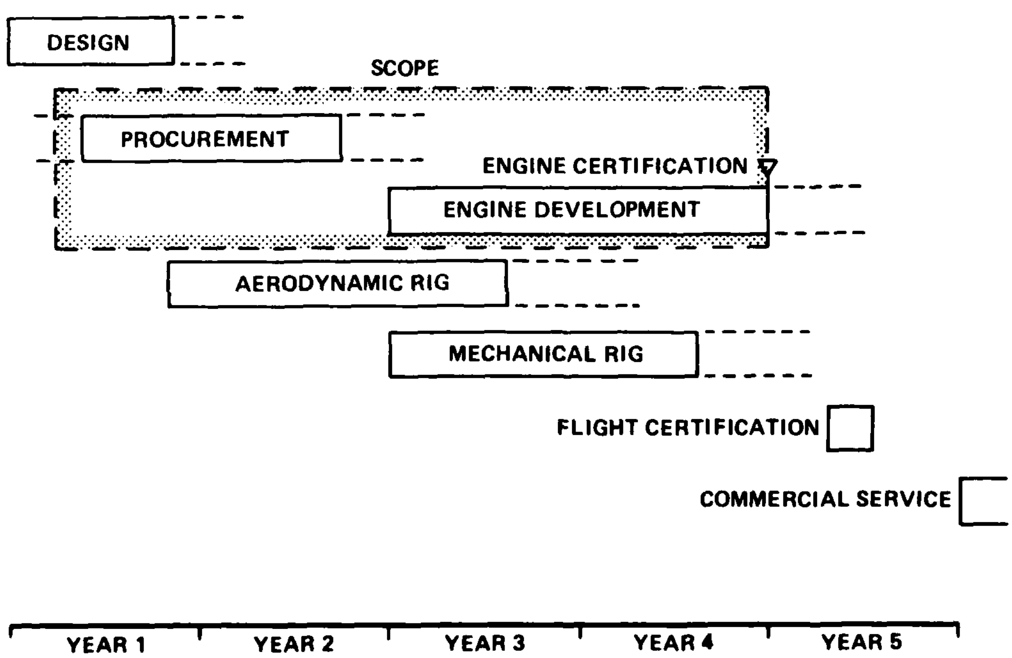

FIG.5 - SCOPE OF THE PAPER

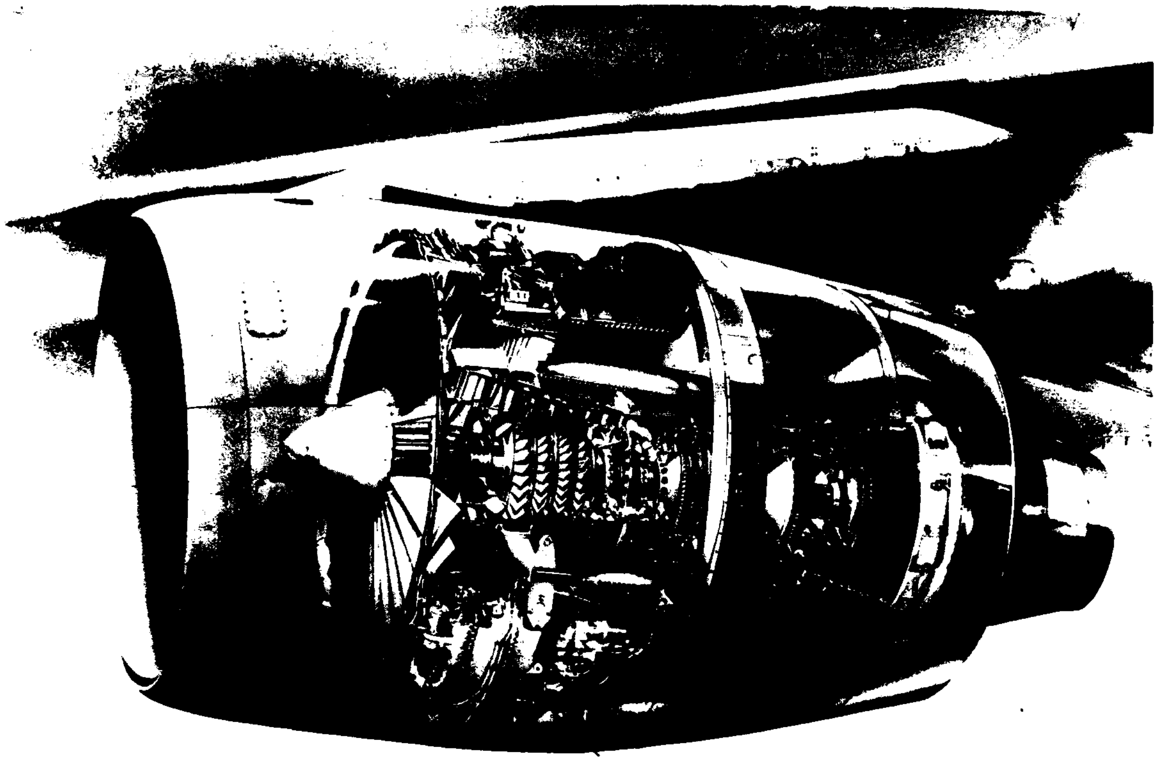

FIG.6 - ROLLS-ROYCE RB 211 PROPULSION SYSTEM

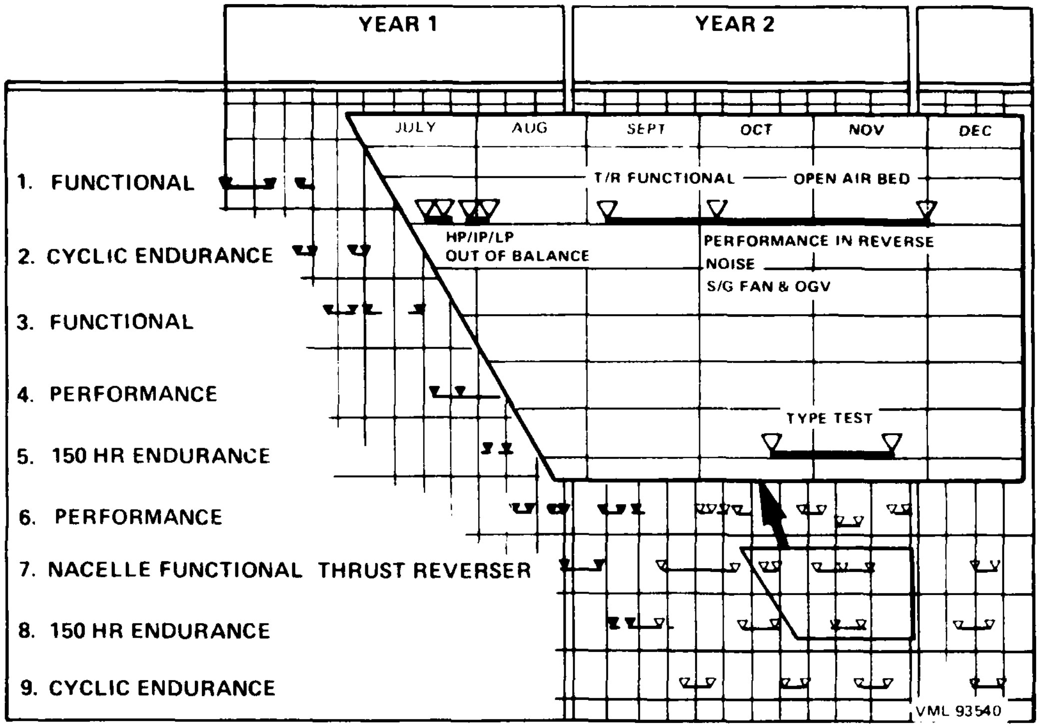

FIG.7 - TYPICAL DEVELOPMENT PROGRAMME

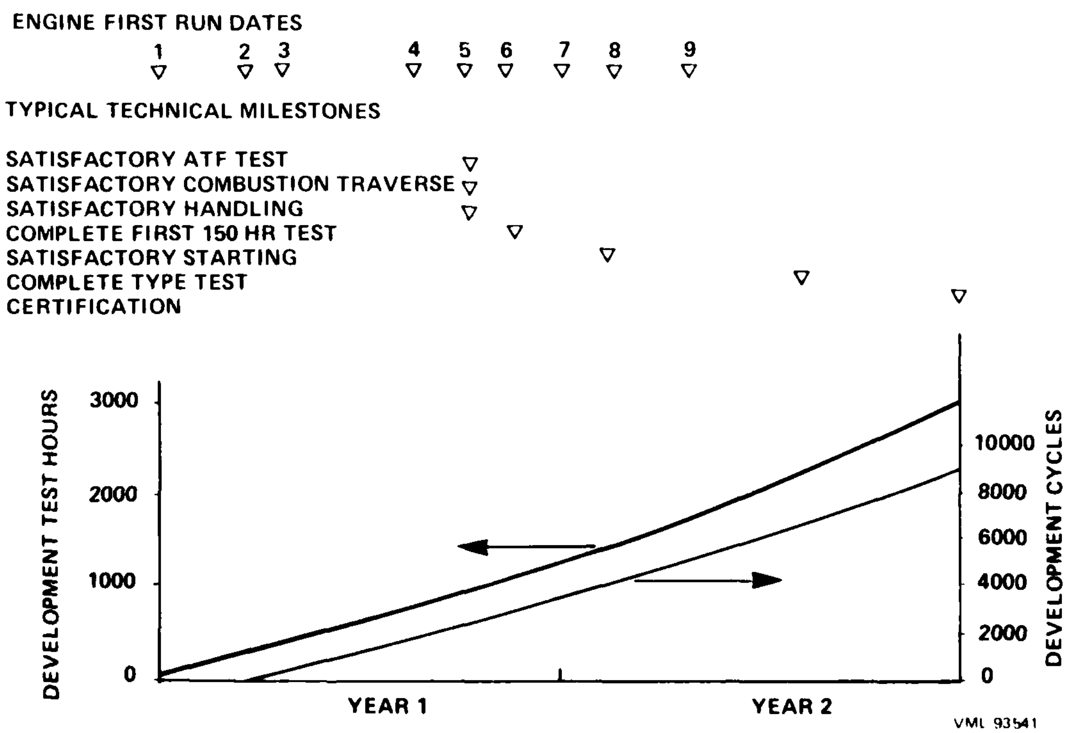

FIG.8 - DEVELOPMENT PROGRAMME MONITOR

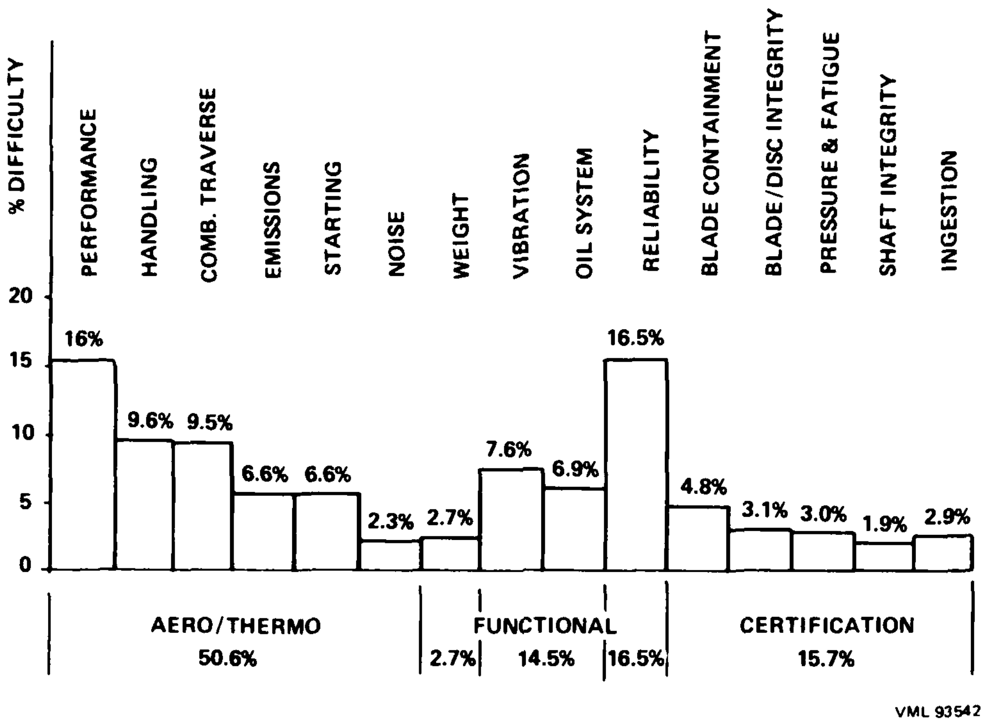

FIG.9 - RELATIVE DEVELOPMENT DIFFICULTY

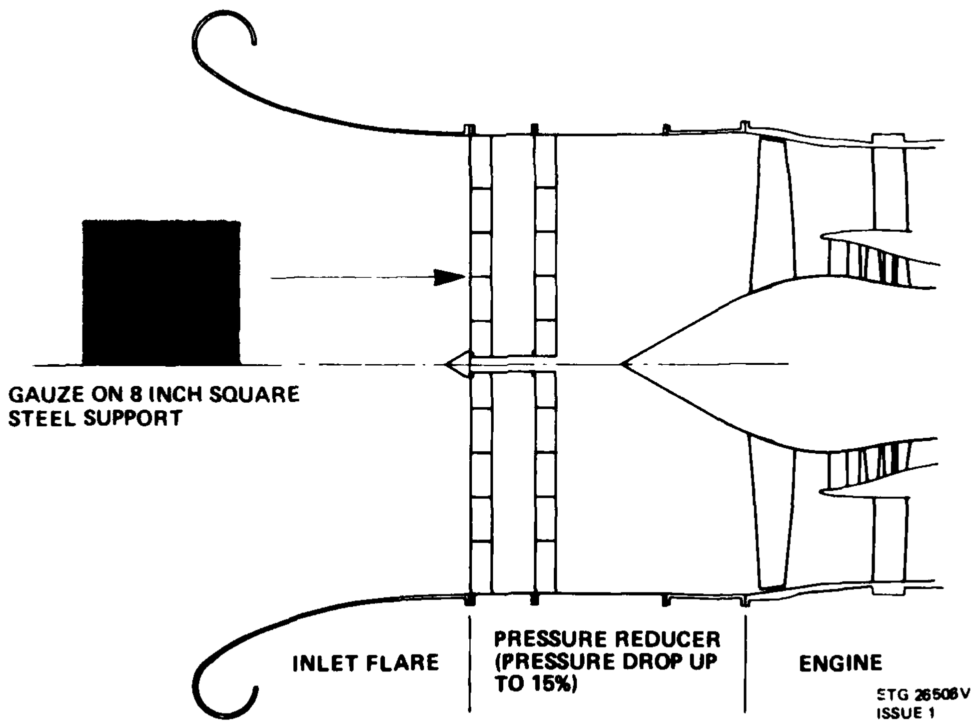

FIG.10 - INLET PRESSURE REDUCER FOR ENDURANCE TESTING

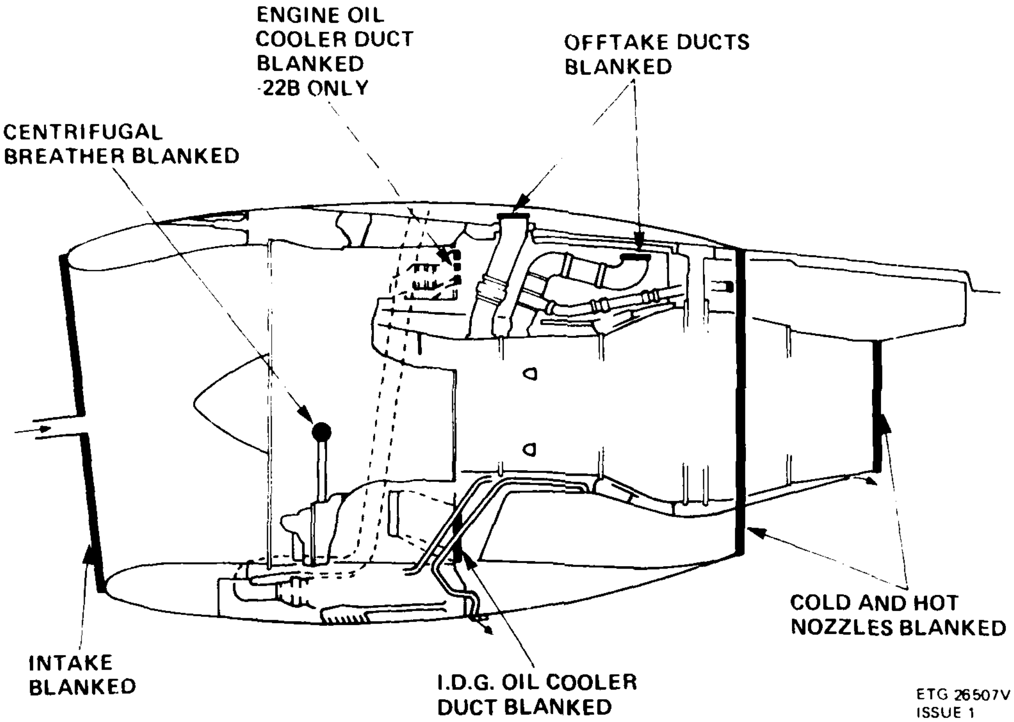

FIG.11 - RB 211 POD LEAKAGE TESTS

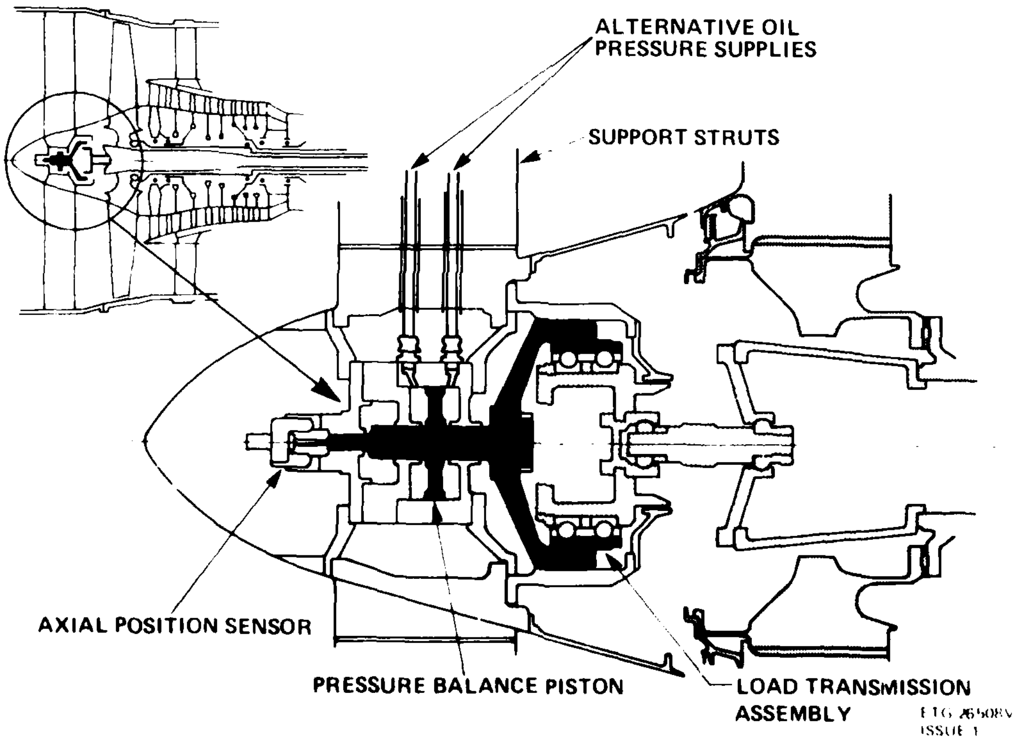

FIG.12 - THRUST BEARING AXIAL LOAD MEASUREMENT (IP AND IP/LP INTERSHAFT BEARINGS)

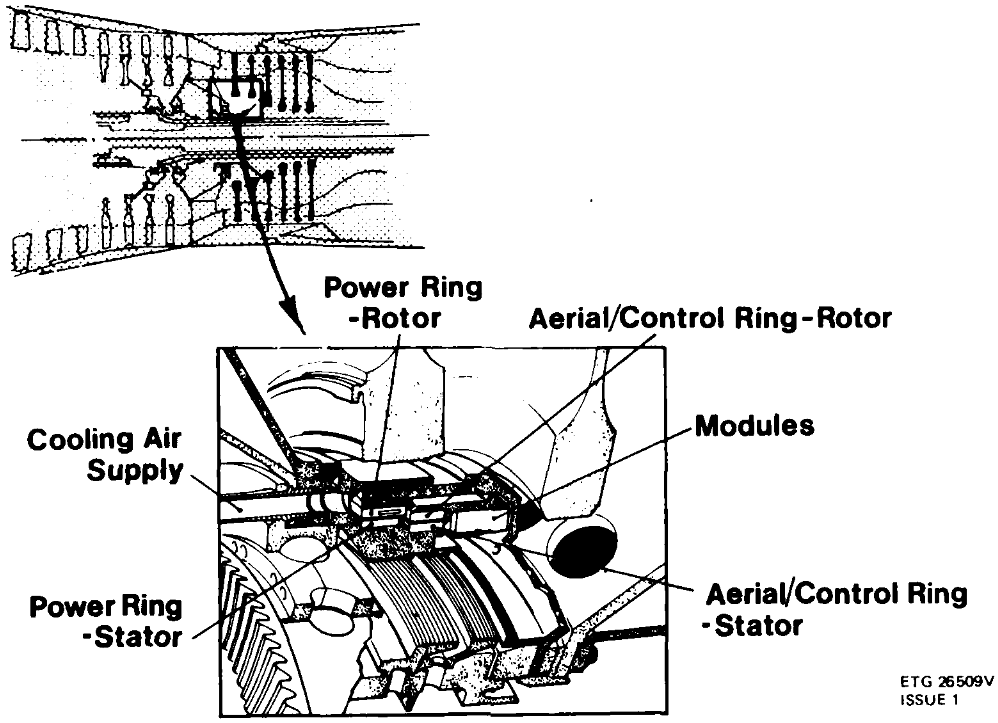

FIG.13 - HP SHAFT SIGNAL LEAD OUT EQUIPMENT

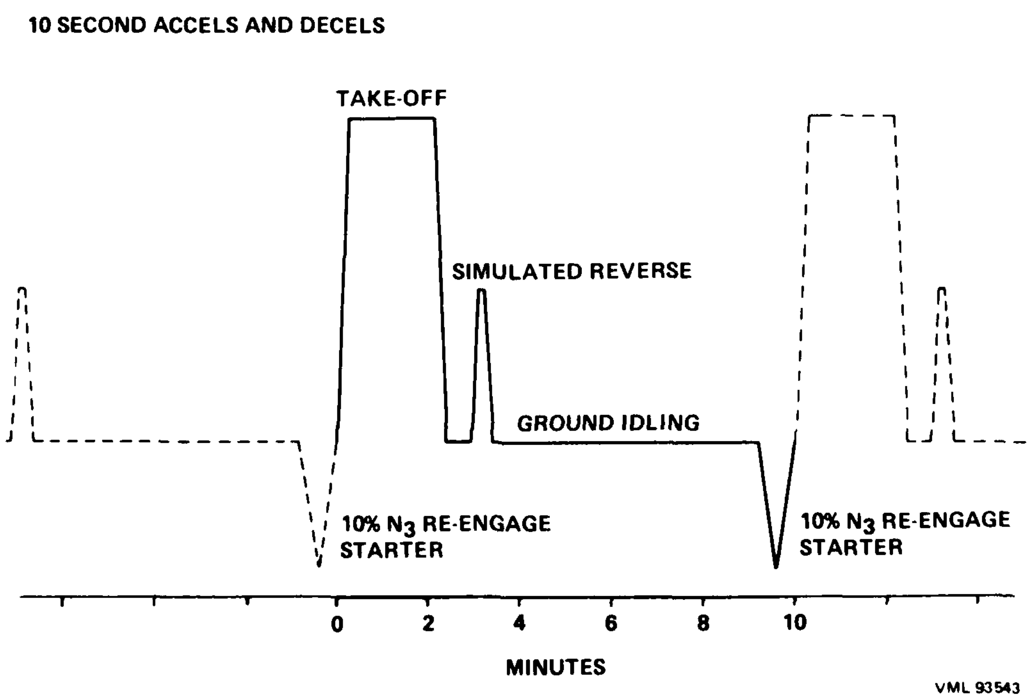

FIG.14 - CYCLIC ENDURANCE TEST SCHEDULE

RB211 *typical X-rays of turbine seals*

LP2

Engine static

Engine at high thrust

LP2 turbine blade X-rays

DNG

27165

FIG.15 - RB 211 TYPICAL X-RAYS OF TURBINE SEALS

DISCUSSION

M.Mihail, Bureau Veritas, Fr

You have pointed out the use of new materials. What is their influence, not over the performance, but over the longevity and viability? Are they intended for improving them, too, in a considerable manner?

Author's Reply

If by "performance" you mean all aspects of the competitiveness of an engine, like weight, fuel consumption, etc., I agree with you.

Paul Chetail, Air France, Fr

The author indicated identical degrees of difficulty (16%) regarding reliability and performance. Is this identity the result of a compromise chosen by the manufacturer or rather the result of the intrinsic characteristics of the RB211?

Author's Reply

The question refers to Figure 9 of the paper. The similar values of difficulty for performance and reliability are coincidental. While there is a small degree of interactivity, reliability is primarily concerned with mechanical design weaknesses leading to unpredicted failures. Performance is primarily concerned with clearance and cooling air optimisation and control.

DEVELOPMENT TEST REQUIREMENTS

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ABSTRACT

The propulsion system development process extends through deployment and must take into account maintenance and logistics support requirements and cost. A "systems" approach to the development process must consider the interrelationships of system subsystem mission usage and the impact on performance, operability, and durability.

Past propulsion system development programs have not sufficiently emphasized the impact of changes in weapon system characteristics and usage on engine performance, operability and life. These changes are more commonly characterized by increases in aircraft "takeoff gross weight" drag or variation in mission requirements. Failure to consider their impact on the propulsion system during the development process can lead to reduced system performance and increased maintenance cost.

It is the intent of this paper to introduce a methodology to consider these needs. The concept of baselining engine characteristics, including maintenance requirements based on evolving weapon system characteristics, will be considered.

Introduction

The capabilities of an aircraft weapon system are defined by the prescribed need or threat that the weapon system must meet. Acceptable levels of these capabilities are substantiated, demonstrated, and qualified through a comprehensive development process which encompasses design, test and development, and deployment. Because this process extends through the deployment phase, the development process must also include engine maintenance and logistics support plans. To be effective, these plans must be consistent with the actual operational characteristics and usage as well as the probable deployment structure. This paper deals with establishing propulsion system test requirements that will provide an operationally acceptable propulsion system, and maintenance and logistic support plans which are appropriate to the usage requirements.

To accomplish this, it is essential to use an approach in developing these plans that encompasses the total weapon system, hereinafter called the system approach. The interrelationships of system subsystem mission usage requirements must be thoroughly understood and translated into a data base (baseline) which will accurately reflect the impact of changes in system characteristics or mission requirements on subsystem capabilities. Furthermore, an effective procedure must be established to derive, develop, and track system-subsystem functional characteristics which can affect system capabilities. In the case of the engine (propulsion subsystem), key characteristics upon which projected system capabilities are based must be defined and demonstrated during the full-scale phase of the weapon system development program. Figure 1 indicates these key characteristics and relates them to weapon system functions.

In addition to delineating the key functional characteristics of the engine, Figure 1 emphasizes the requirements demanded by the system approach. Performance, operability and life functions of the engine must be defined in relation to system capabilities and validated for the projected operational environment. This allows full definition of system capability and the development of accurate maintenance and logistics support cost estimates appropriate to the system usage.

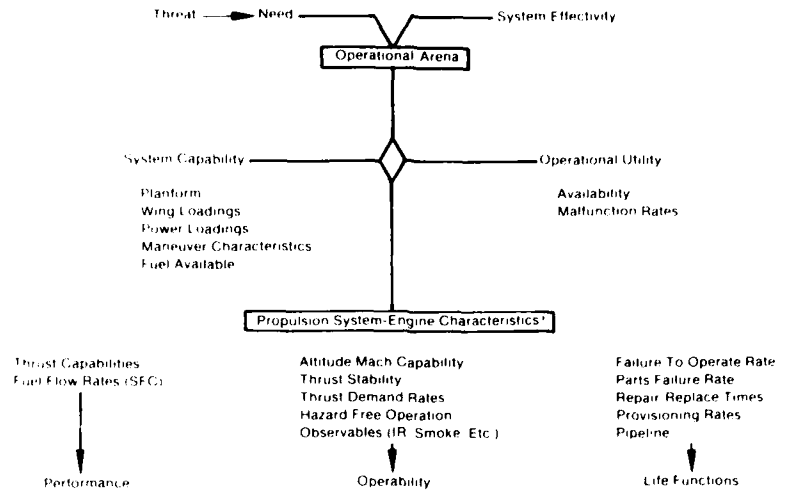
Background

A review of the past 10 to 15 years of military aircraft propulsion system history reveals an engine development process based on engine specifications which defined "engine" performance and durability standards and the testing required to "demonstrate" them. These practices had the effect of obscuring relationships between system and subsystem operational characteristics. As a result, the standards of engine acceptance became oriented around military specification compliances rather than satisfaction of evolving weapon system needs.

This approach to engine development and testing was characterized by a "system vs subsystem" noninteractive development process. Evolving system parameters, range, maximum Mach number, maneuvering envelopes and usage (which are dependent on both the aircraft and engine characteristics) were addressed only during the initial development stage. This approach contributed to deployed system operational problems, high maintenance cost and, in some instances, a dissatisfied system operator.

Current experience deals with inadequate mission profile definition in the engine model specification and highly structured and constrained military qualification tests designed to assure "acceptable service life." Table I shows the model specification requirements for the tactical fighter system selected as an example, as contrasted to the operation usage currently being experienced. It is clear that neither the development agencies nor the manufacturer anticipated that the engine-airframe system capabilities would be exploited by the user to yield aircraft missions that had such significant impacts on engine usage severity. After the onset of full scale development, engineering attention was oriented toward engine specification "requirements" rather than the systems data needed to project and update evolving weapon system characteristics. This lack of tracking of evolving usage factors and weapon system needs caused the engine durability test base to be targeted exclusively around the mil spec requirements. Post qualification tracking of flight experience, however, began to yield wide anomalies between specification test duty cycles and flight operational engine utilization. For instance, total time at military power was approximately 60% of that specified in the initial requirements. Actual high Mach time flown (above Mach 2) has evolved to less than 1[illegible] that amount required for the MQT test sequence. However, this reduced hot time severity is more than offset by at least a six fold increase in cyclic severity relative to the transient life demonstrated in the prescribed MQT. Increased cyclic usage severity, which was not adequately addressed during development and MQT, has resulted in substantial increases in maintenance costs. Such historical lessons are key ingredients in seeking to improve future development programming and testing.

Past trends are clear — few weapon systems developed in the past 15 years have entered service without extensive complaints from the user and logistic organizations. These complaints have dealt primarily — from the propulsion system *standpoint* — *with decreased* operability accompanied by maintenance costs far in excess of programmed resources. The conclusion is obvious — improved weapon system engine development and testing criteria is needed that will adequately baseline both system and propulsion system needs and reverse the trends toward unprogrammed increases in logistics and support cost.



FIGHTER ATTACK SYSTEM ENGINE FUNCTIONS

Function	System Capability	Engine Function	Criteria
Performance	• Range, Speed, Ordnance • Acceleration and Lifting • Fuel Available and How Utilized • Flexibility	• Thrust required vs Thrust Available • SFC at Thrust	• Combat Performance Advantages • Positioning Ps etc • Persistence Fuel
Operability	• Power on Demand Without Instabilities or Malfunction in the Operating and Maneuvering Envelopes/Tactics	• Power Demand and Response • Thrust Stability • Hazard Free Operations • Malfunction Definitions	• Combat Performance Advantages • Expanded Stall-Free Envelopes Definitions • Safety • Malfunction Rates and Impact On Operational Support
Operational Life	• Availability or Readiness	• Usage vs Component Failure Rates	• Shop Visit Rates for Specific Usages • Shop Visit Rates for Operability Malfunctions • Repair/Replace Times • Provisioning Needs

Figure 1 System Definition and Development

TABLE I. "DESIGN" REQUIREMENTS COMPARED TO OPERATIONAL USE

Design Parameter	*Design Requirements*	*Operational Use*
● Total Operating Time — Hours	3,000	2,000
● Full Power — Hours	525	235
● Mach 1.6 or Above — Hours	69	10
● Full Throttle Transients — Cycles	1,765	10,360

Discussion

Alternative engine development methods are available for the next generation of aircraft development programs. For example, minimum risk derivatives, commercial engine adaptations or highly matured core engines are being considered. However, the final selection will be based on which offering meets the defined "need" with predictable and affordable logistics support cost. To meet this objective, aircraft development requires a "system" approach to the development process that considers the interrelationships of system/subsystem/mission usage.

This approach includes the basic requirements listed below:

- Develop engineering philosophy where the relationships between engine/system/mission usage are defined and continuously tracked throughout the weapon system development and flight test phases.

- Develop define utilize methodologies to establish engine testing procedures for Performance/Operability/Life factors based on the usage relationships.

- Use this combined test and analysis approach to define the engine aircraft fundamental elements discussed below.

  Performance where system mission segment requirements for installed thrust, fuel flow rates are related to engine thrust and SFC.

  Operability where weapon system maneuverability envelopes are related to inlet/engine, augmentor, and control limit operation and stability margins.

  Durability/Life where weapon system characteristics and mission profile segment power requirements are used to establish testing methods relating usage to engine/component failure rates and maintenance and support factors.

Performance — The First Fundamental Element

The initial performance baseline characteristics of the engine must be related to the "selected" weapon system needs to establish basic threshold requirements. Aircraft characteristics such as configuration, weight, mission segment power requirements, fuel allowances, platform and L/D must be accounted for relative to engine thrust, weight and specific fuel consumption. In addition, the anticipated envelopes to be flown must be defined as accurately as possible. This requirement for thrust/drag accounting and operational envelope definition must be recognized if a responsive propulsion system is to be developed and made operationally successful.

Such an approach is necessary to prevent arbitrary insistence upon engine capability that result in minimal system gains. Further, system and subsystems performance characteristics must be continually updated, and matching changes in either the aircraft or propulsion system must be weighed relative to their effects on threshold system performance and operating/support characteristics. Particular care must be taken to ensure that engine performance and control schedules are also evaluated in light of installed power requirements which take into account the relative power sensitivity and stable operating characteristics of both inlet and exhaust subsystems.

Operability — The Second Element

Engineering methodologies dealing with inlet/engine compatibility were developed in the late 1960s. It is clear from the operational success of current tactical fighters that both the users and manufacturers understand the concept of inlet/engine compatibility and matching. What is perhaps not clear is the impact of the total "stability stack" inherent in the operability characteristics of the weapon system. It is necessary not only to define the inlet/engine interaction but also to address all elements of thrust stability requirements imposed by the evolving weapon system on the baseline engine characteristics. For example, as soon as aircraft characteristics begin to evolve, the effects of aircraft maneuverability in terms of inlet α (alpha) and β (beta) excursions (see figure 2, Operability Impact) must be matched against inlet recovery and distortion to evaluate the impact of augmentor ignition characteristics and power transients on both external and internal margin. Tracking of aircraft systems characteristics (i.e., inlet, exhaust, etc.) and engine characteristics and capability is required throughout the development and on into operational phases. It is also necessary that systems envelope limits be fully explored in the development and operational phase. Evolving combat tactics will extend the boundaries, and the impact on operability must be understood.

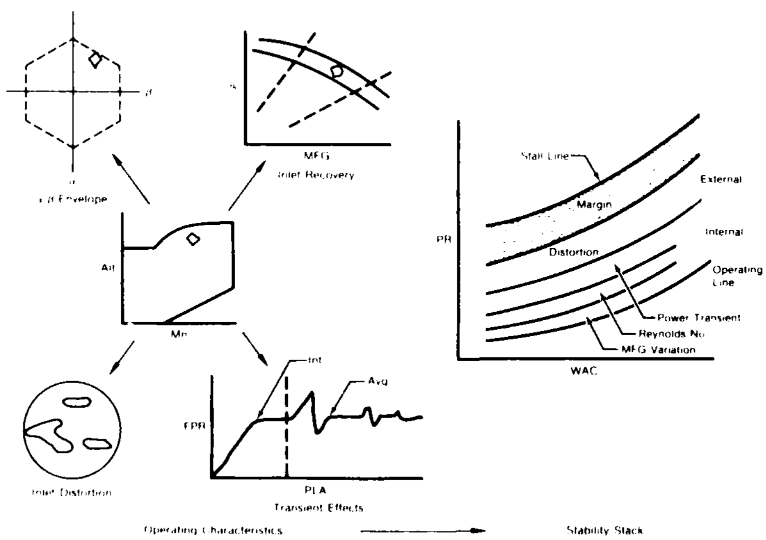
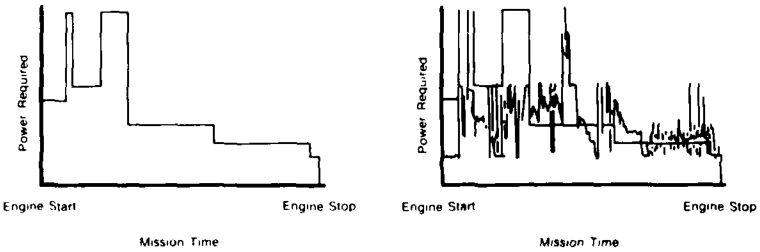


Figure 2 Operability Impact

Durability/Life — The Last Element

In today's environment of escalating logistic support cost, it is absolutely essential that durability be considered in developing program requirements. A review of previous development programs makes it clear that early estimates of weapon system usage that are defined by "stick mission" (see figure 3), usually reflect an optimized steady state estimate of actual system power usage. A more technically correct approach consists of estimating ranges of system power requirements vs power available at the onset of a new aircraft definition. Figure 4 is a typical mission defined in terms of range, altitude, and Mach number. Taking care to *segment* the mission properly and applying the rules of aircraft characteristics (i.e., power matching) yields the range of power available, power required, and variations that may be expected on any segment. Application of this concept for each of the proposed mission and mission segments yields the likely range of

transient power usage and thus defines more accurately the ranges of operational severity exposure of the engine components. Continued updating of this process is essential if the engine testing is to be effective in yielding a capability to project and program engine operating and logistics support requirements.



(a) Estimated Power (RFP) Required as a function of Mission Time for an Intercept Mission
- RFP
- Flight Test Data

(b) Actual Power (Flight Test Data) Required as Recorded on Instrumented Aircraft During an Intercept Mission

Impact on Engine Component Lives

Component	Original Design Life (Hours)	Revised Est Life (Hours)
1st Fan Disk	8700	900
10th Compressor Disk	4000	1400
1st Turbine Disk	5600	900

Figure 3. Power Required Profile

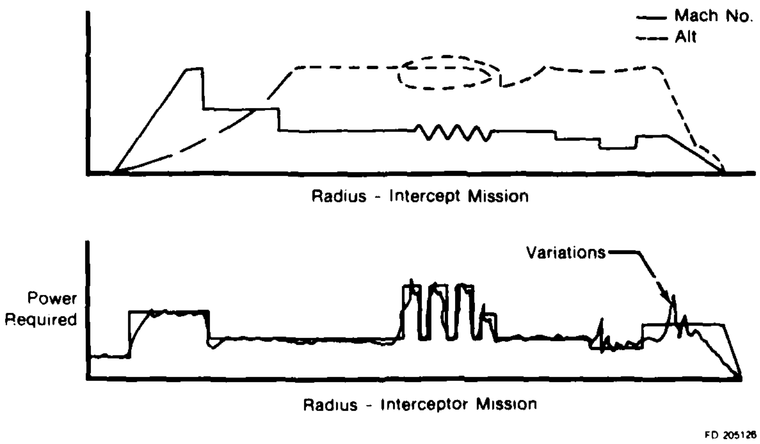


Figure 4 Power Matching

Test and Evaluation Procedures

From the outset of a development program, methodologies must be defined and utilized to derive engine testing needs for Performance/Operability/Life. Current test procedures, if tailored to system requirements, can deliver the maximum payoff for each engine test hour accomplished. This will also lead to improved evaluations and projections of system performance and operability characteristics and tradeoffs to control risks and costs for the evolving system. Emphasis will be placed on the expanded use of mathematical modeling procedures that translate engine testing and analyses results to durability data for estimating engine support needs. It is the intent of these methodologies to improve both the testing and resulting projections of engine durability/reliability. This approach will lead to timely definition of hardware failure modes as opposed to delaying them into the deployment phase. Figure 5 shows an example of a developing data base which can reduce the uncertainty in usage projections such as SVR and maintenance cost. Finally, this data base is used to make Weibull life projections (see figure 6), which can then be related to variations in system usage through life prorating techniques. It is interesting to note that this approach is consistent with established commercial system-engine practices and relates aircraft characteristics, mission, mission mix, ground time, flight time, and usage to both scheduled and unscheduled shop visit rates. These shop visit rates drive major elements of support costs. Such results are verifiable early in flight test phases and will be used to validate or adjust testing to project the impact of variations in usage on shop visit rates and maintenance cost factors.

Conclusion

It is necessary that a fundamental understanding of the functional relationships between weapon system requirements, and engine performance, operability, and life factors be achieved early in the aircraft development process. Any approach that lacks definition of the

evolving weapon systems needs will severely reduce engineering's capability to respond to the normal evaluation of the system, mission and production programs. It is also important that the engine aerodynamic and performance flow path be derived and defined and that operability and durability testing requirements be based on projected system engine operating exposures and usage factors. Efforts to baseline and balance engine durability and life sensitivities are essential if predictable levels of system capability, operability and engine support cost are to be achieved. A closing consideration is that flight test and initial deployment are only the beginning. If we are to achieve one of our prime objectives — an acceptable system and a "satisfied customer" — then, the tracking of usage and maintenance parameters must be initiated during the development process and continued throughout the life of the system. This approach forms the baseline of successful engine management and supports the longevity and productivity needs of the evolving tactical weapon system.

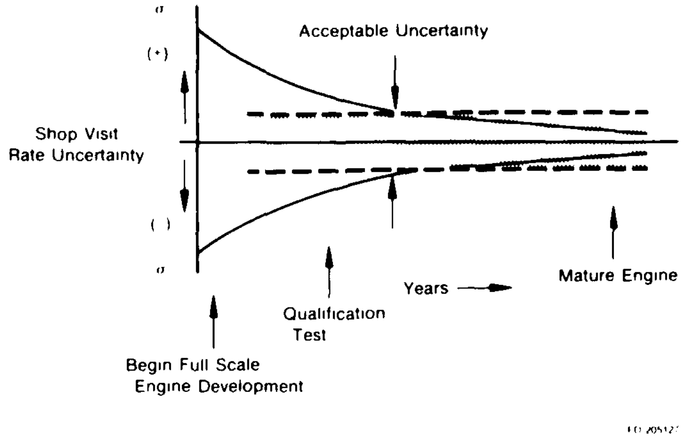


Figure 5. Reduced Usage Uncertainty

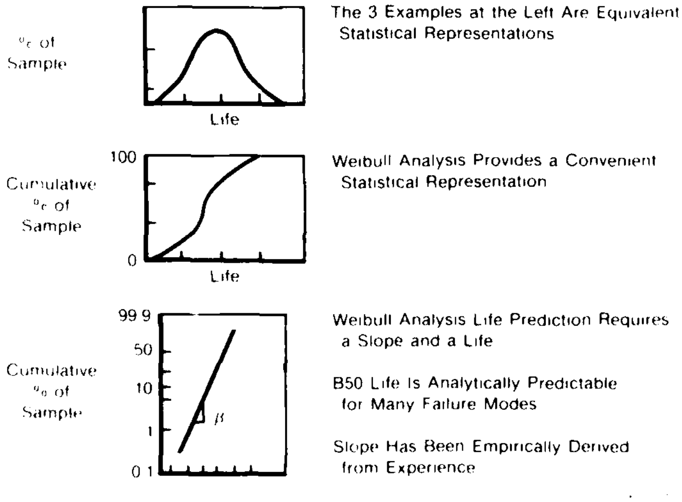


Figure 6. Engine Component Weibull Representations

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DISCUSSION

Paul Chetail, Air France, Fr

What is the role played by the different policies of maintenance in the author's mathematical model? Given the same intensity and level of deterioration, does the type of maintenance exert an influence on the maintenance cost?

Author's Reply

The "type" of maintenance in terms of degree of teardown and frequency of inspection certainty impacts the cost of operating any turbine engine. This impact is due to two factors — 1st, physical damage due to a primary failure (i.e. secondary damage); 2nd, increased scrappage due to operating components that are deteriorated with new, non-deteriorated components. There are other maintenance impacts due to trim and ground operation but normally remain of a secondary nature.

Once this operating scenario is established, in terms of the base line or statistical data needed for the operation and support simulator, the projections of shop visit rate or maintenance cost due to usage variations include the factors associated with "maintenance type".

Engine Life Development

Brian Brimelow
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Figures 1 and 2 (taken from reference 1) graphically illustrate how rapidly support costs have increased in a time span of two decades Figure 1 shows that Operating and Support costs (O&S) have grown from 35% of total life cycle cost to 80% of the total life cycle cost in the time period studied Figure 2 shows that, despite the rapid increases in fuel costs the elements driving this increased O&S cost are maintenance labor and parts replacement These increasing support costs of propulsion systems have generated for both government and industry a need to reassess their approach to engine development Most past engine developments have emphasized performance as the primary goal and used reliability criteria based on approaches which were developed for electronic systems (in those cases where reliability was specified at all) There has been little attempt to really understand the causes for high operating costs and to tailor engine development accordingly What has to be done is to set cost requirements, then prove by test and analysis that they can be met However before the development test effort can really be determined, a thorough understanding of the total system requirement, as it affects the engine, must be obtained This aspect has not been adequately addressed in the past

Acquisition Cost vs Operation and Support Cost

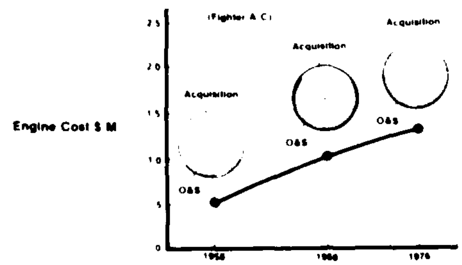

Figure 1

Maintenance Cost Drives O&S

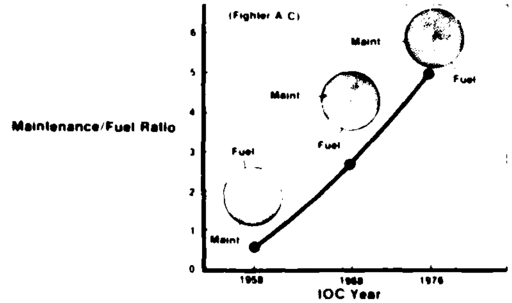

Figure 2

Requirements Definition

Requirements to the engine manufacturer come in the form of thrust, fuel consumption and weight that will allow an aircraft to meet a specific mission, as in Figure 3 From this can be derived a design mission power profile, i e X% time at max power, Y% at intermediate power, and Z% at cruise for various combinations of altitude and Mach number Unfortunately, closer examination of real engine usage shows that hidden behind this simple profile is a major engine variable the number of throttle transients to be expected in actual flight Figures 4 and 5 show the variation in engine usage taken from actual flight data to the same profile as Figure 3

Further complicating the engine usage issue is the fact that the profile and throttle transients for the design wartime missions are frequently much less severe than the training profile used to maintain pilot proficiency Lack of appreciation for the magnitude of these transients is reflected in past qualification requirements

Typical Mission Power Profile

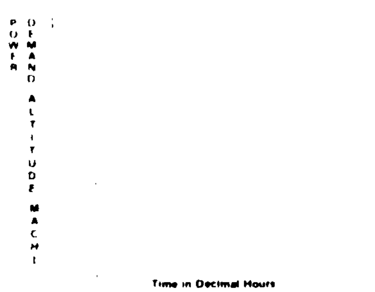

Figure 3

Minimum Severity Usage

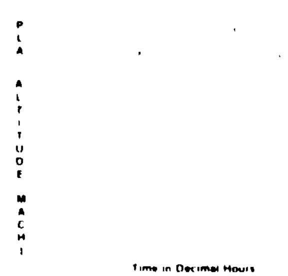

Figure 4

Table 1 compares the MIL 5007C endurance qualification which was used in the late 60's and early 70's to qualify fighter engines with a current endurance test based on flight data for a 1000 hours of operation. The comparison suggests reasonable correlation between earlier test and flight usage for steady state power settings but is not representative in terms of transient usage. As a result, engines which have undergone AMT (Accelerated Mission Test) can be expected to have superior lives in real service usage. A clearly defined engine usage spectrum is a prerequisite for any form of engine parts life requirement and an ability to estimate parts life is the cornerstone of support cost projections.

Maximum Severity Usage

Figure 5

F101-X/F404 Durability Testing

	Total Time (hrs)	Time at Max Power (hrs)	A B Lights	(LCF) Off Int Off Cycles	(FTC) Idle Int Idle Cycles
• PFRT (MIL-E-5007C)	60	26	78	10	140
• <u>QT</u> (MIL-E-5007C)	150	65	325	25	350
• <u>AMT</u>	430	127	2171	565	6499

Table 1

Mean Time Between Failure (MTBF) has been the classical measure of reliability and by inference, maintenance cost. Although more than 90% of maintenance actions are taken at organizational level (Flight Line Maintenance) less than 10% of the operating cost occurs at this level (Figure 6). Unlike avionics, a combination of high costs of some engine parts (e.g., cooled turbine buckets) plus a wide variation in the cost of gaining access to the failed part makes MTBF inadequate as a cost control requirement for the engine. Examination of logistic data shows that both parts and labor costs are directly relatable to the number of times an engine comes off wing and goes to an intermediate or depot maintenance shop (Figure 7). MTBF and Mean Time Between Maintenance Action (MTBMA) do have a part to play in the overall reliability picture in that they significantly impact two other critical areas — operational readiness and mission completions, however, support cost reduction requires a major effort to keep the engine out of the maintenance shop. Predominant causes for shop visits are life limited parts, system operability problems, and performance deterioration. Targets in each of these areas must be determined at the outset. In order to design to these targets, requirements must be clearly defined as to the weapon system objectives and how the engine will be used to achieve these objectives.

Propulsion System Cost Breakdown
(4 Year Average FY 75 - FY 79)

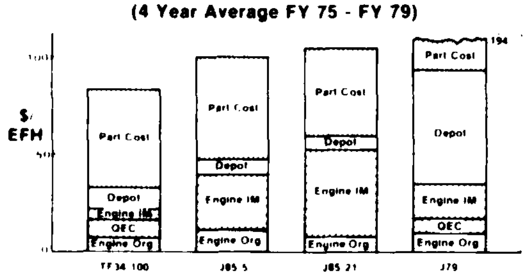

Figure 6

Engine Maintenance Cost vs Removal Rate

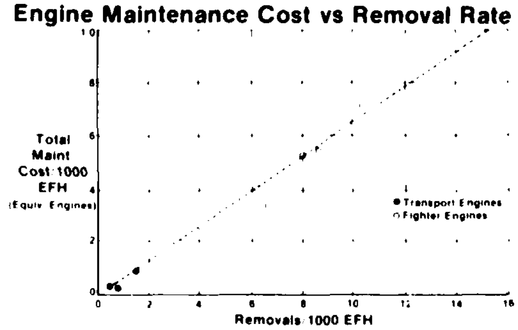

Figure 7

In the operability area it is equally important to define the range of conditions required for the aircraft to meet its mission requirements. The projected range of angle of attack, yaw, and inlet engine flow match must be determined for all flight conditions to which the aircraft can be taken in controlled condition (Figure 8) (Altitude and Mach number for these maneuver values may be in or out of the 1 g envelope.) Failure to fully establish these requirements will result in an inadequate test program followed by frequent engine/control removals because of adverse tolerances stack-ups of otherwise serviceable components. Given adequate definition of the aircraft requirements, shop visits for operability problems can and should be targeted so that refurbishments can be carried out at a time when the engine is removed for replacement of a life limited item.

A concensus amongst aircraft propulsion system specialists could quickly be reached about figures of merit for thrust to weight, or specific fuel consumption of a military engine but the benchmarks for the cost per operating hour could not be established nearly as readily. Similarly, the qualification of engines concentrates on ensuring that the engine is safe to fly and meets performance; however, meaningful test requirements for support cost projections are rare if not non-existent. For a major reduction in support costs to be achieved, requirements directed at these costs will have to be issued and tests to verify these projections made part of the qualification process.

In order to quantify these requirements, General Electric has a system of ten "Bottom Line" measures (Figure 9). These are used to track the operating costs of existing systems and so establish the design target or benchmarks for engines under development. The payoff or "Bottom Line" for making durability/reliability a prime factor in engine development comes in the form of:

- Lower Operating Costs
- Improved Readiness.
- More Mission Completions.

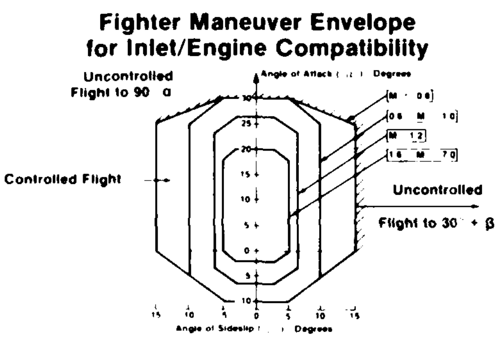

Figure 8

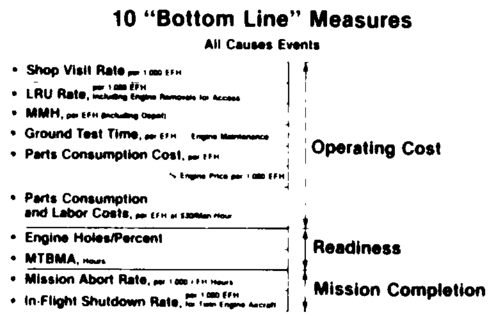

Figure 9

Development Program

Once the requirements are understood then a development program can be structured to prove by test that the durability/reliability requirements can be met. No single test will do though, there are many factors to consider. This program

- **Tests Parts to Failure**
  This is necessary to establish service limits and to establish progression rates for cracks, oxidation and other failure modes. Minimum failure rates are established by operational requirements. If the engine does not meet these rates, improvements must be made and proven.
- **Tests an Adequate Statistical Sample**
  It must be recognized that parts have individual life variations. An adequate test sample is necessary before a Weibull failure distribution can be plotted, to analyze and provide reliable life predictions

- **Simulates Actual Service Usage in Testing**

  To accurately evaluate durability, testing must simulate the actual mission in terms of hours of time at full temperature, number of equivalent full throttle cycles, number of start stop cycles, acceleration times, and a representative mix of maximum rotor speed. Additionally, pressures and temperatures must be typical of service usage across the flight envelope. Figure 10 shows the flight conditions simulated in a typical development test series. Endurance effort is concentrated at flight conditions typical of the major forecast usage. System functional tests are carried out at all points of the projected flight envelope. For realistic cost effective development, Accelerated Mission Testing (AMT) which incorporates extra severity and eliminates non-demanding portions of the mission is now the standard test for engine endurance.

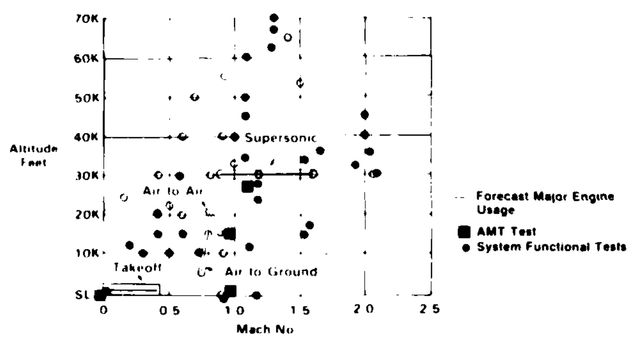

Figure 10

- **Evaluates All Key Failure Modes**

  High cycle fatigue, low cycle fatigue, rupture and creep, oxidation and burnout, erosion, and physical interference and impact are some of the key failure modes that engine and component test programs must consider. Different parts have different dominant failure modes.
- **Simulates All Flight Conditions in Test**

  Durability and reliability tests must include various altitude, sea level and ram engine test facilities, as well as many component facilities from spin pits to shake tables.
- **Demonstrates Life and Reliability Requirements Before Production to Assure Early Maturity**

  It is important that the first production engine meet initial service requirements for reliability and durability. These requirements must be established in the engine's specification and testing must assure requirements are met during development. For these requirements to be assessed, it is essential to track engine usage from flight test data and mission projections.
- **Makes Cost Effectiveness an On-Going Concern**

  There should be on-going CIP development to make cost-effective, in service improvements (such as improving repair techniques, maintenance methods, and cost reduction) that will assure that engines continue to meet Bottom Line Measures of in-service excellence, even after 10-15 years and at 3000-4000 hours age!

Summarized below are the major test techniques necessary to a successful development program:

1. **Accelerated Mission Testing**

   This is realistic endurance testing of an engine under conditions that closely simulate the actual usage it will be subjected to in field operations. By reducing missions to their cyclic components, equivalent operational engine running time is more rapidly accumulated.
2. **Other Engine Testing**
   - **Test Cell** - Other engine testing -- where all aspects of the complete engine characteristics are demonstrated prior to qualification. These include ruggedness (ingestion of foreign objects, ice, etc.) maintainability demonstrations, climate environment, etc.
   - **Altitude and Ram Facilities** - The purpose of these complete engine tests is to demonstrate capability and characteristics of operation over the complete flight envelope. Stall margins are measured with maximum inlet distortion and the interaction of controls and engine transient characteristics are determined and augmentor and burner stability envelopes are mapped. In addition, instrumented stress and structural capabilities are evaluated at the most severe extremes of the flight envelope.

3. **Special Engine/Component Testing**

Special engine component testing — these are the tests which supplement AMT endurance and other complete engine testing in order to evaluate component aerodynamics.

Table 2 shows the failure modes for different engine components

Table 3 shows the relationship between test technique and failure mode

Mechanical Failure Modes

	Age Related							Non Age Related					
	Low Cycle & Thermal Fatigue	Creep Rupture	Oxidation Burnout	Erosion Hot Corrosion	Thermal Distortion	Burst	Wear	Contamination	High Cycle Fatigue	Foreign Objects Birds Etc	Overspeed Overtemperature	Multiple Stall Compatibility	Maneuver Deflection
Compressor Section													
Blades							X	X	X		X	X	X
Vanes								X	X			X	X
Disks & Spacers	X						X				X		X
Casings	X					X							X
Structures													
Frames	X					X							X
Bearing Supports	X												X
Turbine & Combustor Sections													
Blades	X	X	X	X			X	X			X		X
Vanes	X		X		X			X			X		X
Shrouds			X				X						X
Disks & Spacers	X	X					X				X		X
Combustor	X		X										
Casings	X					X							X
Afterburner & Jet Nozzle													
Flameholder	X		X		X								
Liner	X		X		X								
Casing	X					X							X
Flaps & Seals			X		X		X						
Engine Subsystems													
Bearings							X						
Lube System (Pump Sumps Seals Etc)							X			X			X
Hydromechanical Control							X			X			
Electronic Control							X						
Fuel Pump							X			X			
Variable Geometry (Vanes Mechanisms)							X					X	
Accessory Gearbox & Drives							X						

Table 2

Evaluating Mechanical Failure Modes with Various Types of Testing

	How each mode is evaluated in engine development	Age Related							Non Age Related					
		Low Cycle & Thermal Fatigue	Creep Rupture	Oxidation Burnout	Erosion Hot Corrosion	Thermal Distortion	Burst	Wear	High Cycle Fatigue	Foreign Objects Birds Etc	Contamination	Overspeed Overtemperature	Multiple Stall Capability	Maneuver Deflection
AMT	AMT with mix of test cell heated inlet and ram	X	X	X	X	X		X	X					
Other Engine Testing	Altitude Facility major tool for Operability and Performance Testing						X		X					
	Special Engine Tests operational stress mapping failure mode testing ingestion etc								X	X	X	X	X	X
Special Component Engine Testing	Spin Pit rotor components disks spacers etc	X										X		
	Pressure Cyclic Test casing frames	X					X							
	Component Mapping rig tests of fan compressor combustor turbine etc								X				X	
	Distortion Mapping component and engine tests to measure stall margin and effect on stress								X				X	
	Bench Test control & lube rig test including vibration contamination etc							X			X			
	Special Component Tests special purpose tests on individual components								X	X		X		X

Table 3

How an Accelerated Mission Test (AMT) is Designed

Since the AMT test is now the backbone of durability development, it is worth looking at how such a test is designed.

Steps in creating an AMT

1. Airframer and military service define aircraft missions:
   - Flight conditions
   - Throttle usage – time at full power and number and type of cycles.
   - Time of each leg and mix of missions.
2. These missions are surveyed to separate the major elements affecting engine durability from those portions of the missions which have little or no effect:
   - Low cycle fatigue cycles.
   - Thermal cycles
   - Time at max power.
   - Afterburner light cycles.

Figure 11 compares the relative severity of the AMT duty cycle with the potential applications. While the AMT cycle is intended specifically to evaluate the engine in 2000 mission hours in the F-16, the severity of the test cycle will give excellent results applicable to both the F-15 and F-14.

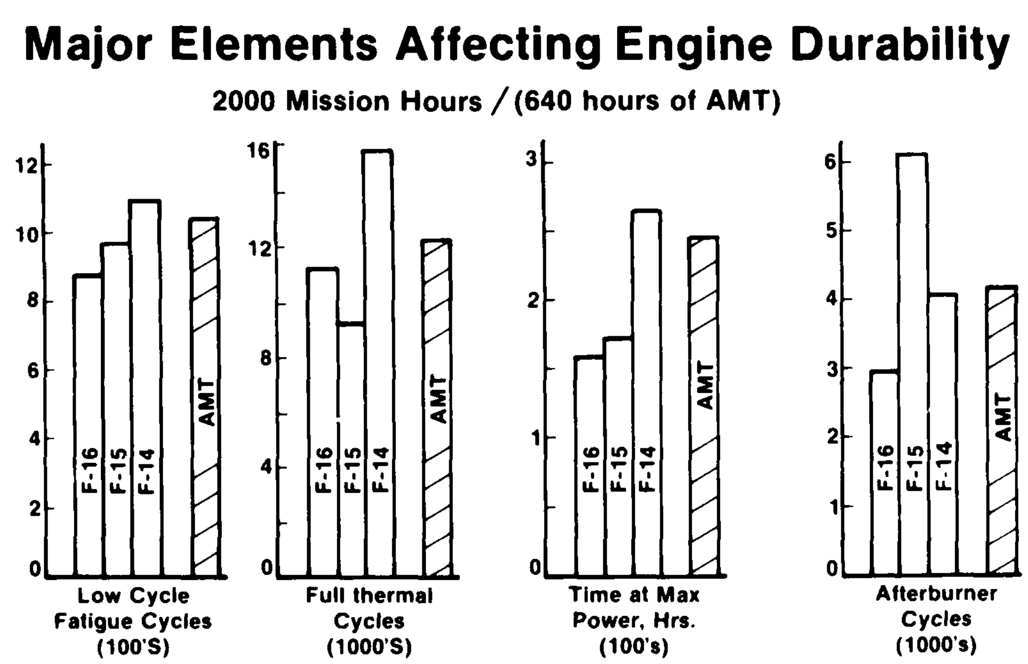

Figure 11

3 Once the major elements and mission severity are identified, they are translated into test cycles which simulate the same or harsher engine usage. Throttle movements approximate realistic pilot actions. The specific engine test cycles used in performing the AMT are illustrated in Figure 12.

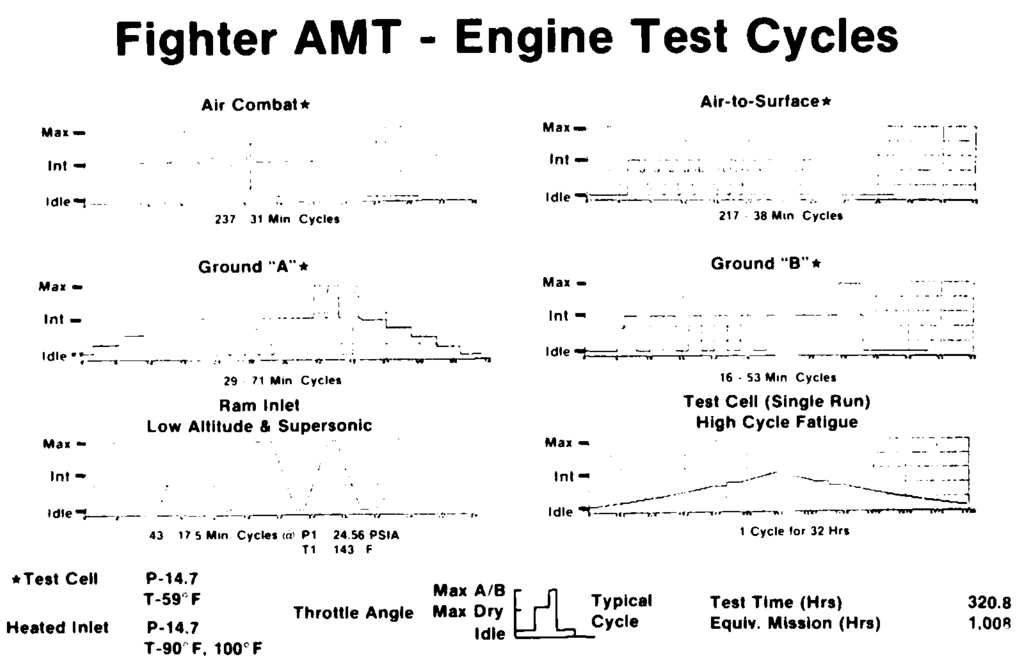

Figure 12

Clearly there has to be a system provided to link operating costs to the component lives as tested by AMT, spin pit, rig tests, etc. This is provided by logistic models combined with mission severity models. Data requirements for such a model are listed in Table 4. The mission severity model uses the test data base and projects service lives based on the projected engine usage (Figure 13). For non-life limited parts reliability growth is predicted based on past experience and the early development history. The logistic model takes into account the scenario infrastructure, number of bases, locations, flying rates, maintenance turnaround times, pipeline times, etc. It also should be capable of assessing secondary damage to be expected after the initial failures and the impact of the maintenance policies selected. In this way parts life and reliability can be converted in $'s per hour, thereby ensuring engineering requirements (parts life) which can be demonstrated by test and be used to verify cost requirements.

Logistic Model Data Requirements

- **Expected Component Life Data (Weibulls)**
  - AMT Test Experience
  - Mission Analysis by OPSEV
- **Random Failure Characteristics**
  - Ultimate Reliability · Similar Items
  - Initial Reliability · Today
  - Growth Dependent on Fix Cycle
- **Fleet Operation**
  - A/C Delivery Schedule
  - Utilization Rate
  - A/C Attrition

Table 4

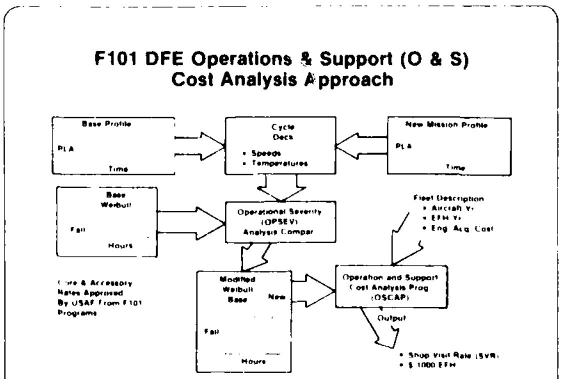

Figure 13

Operational Support Requirements

One thing is almost a certainty in a military development, that is by the time a system actually completes a development and is deployed the planned usage will change either because of a change in role or required tactics (Even if the role does not change, actual usage may differ from that projected.) To this end, although not discussed in detail in this paper, the ability to track actual engine life consumption in service on all engines is mandatory to an efficient logistics system.

Advanced logistic models referred to earlier can project spares needs based on these new life inputs, thereby allowing for forward looking logistic requirements rather than systems which look at the previous years' requirement to determine the next buy. With hot section lives capable of 3-6 years already being demonstrated on advanced engines, a backward looking logistic system is obviously the route to disastrous spares shortage when wearout does appear.

Life prediction based on real usage combined with an adequate test base and modern spares forecasting techniques allow rational judgments to be made on the changes necessary to spares provisioning. It also provides the key to any additional durability development.

Having established the need and the capability to project operating costs for engines, the major question remains — what numbers are realistic? Figure 14 shows the bottom line numbers for line existing in-service engines and compares them to the goals of GE's latest fighter engines . . . the F404 which is in the early stage of production and the F101 DFE which is in development.

10 "Bottom Line" Measures

All Causes, Events

	J79-17	TF34-100			F404	F101 DF
	Status	Field Status	O-1A Status	Maturity Goal	Maturity Goal	Maturity Goal
• **Shop Visit Rate** per 1,000 EFH	3.2	1.6	2.2	1.6	< 2	<2
• **LRU Rate,** per 1,000 EFH, Including Engine Removals for Access	5	2.2	1.9	1.5	< 2	<2
• **MMH,** per EFH (Including Depot)	3.1	9	1.0	1.2	1.2	1.9
• **Ground Test Time,** per EFH - Engine Maintenance	01	01	01	01	01	01
• **Parts Consumption Cost,** per EFH ($79)	$100	$25	$20	$77	<$75	$115
% Engine Price per 1,000 EFH	18.6	4.1	3.3	12.5	10	8.4
• **Parts Consumption and Labor Costs,** per EFH at $30/Man-Hour ($79)	$200	$52	$50	$113	~$110	~$170
• **Engine Holes/Percent**	0/0	0/0	0/0	0/0	0.0	0/0
• **MTBMA,** Hours	74	135	125 EST	300	175	175
• **Mission Abort Rate,** per 1,000 EFH Hours	73	1.2	55	1.2	< 5	< 3
• **In-Flight Shutdown Rate,** per 1,000 EFH for Twin Engine Aircraft	NA	22	11	12	1	< 1
Average age Hours	~ 2100	~ 400	~ 900	3000	3000	3000
Status as of	Dec 79	Jan 80	Jan 80			

Figure 14

The figures shown are for the mature goals. Equally important is to achieve this maturity early. Figure 15 is typical of the goal for these engines in terms of when maturity should be achieved. This early maturity is not only highly desirable to minimize early support cost and maximize operational availability, it is absolutely vital to avoid prohibitive modification and post-production development costs. This can be achieved by a reorientation of the development effort (Figure 16). This approach where required parts life is established prior to production combined with maximum effort fix cycle for early service reliability problems (some of which must be anticipated) can achieve the goal of acceptable support costs for advanced systems.

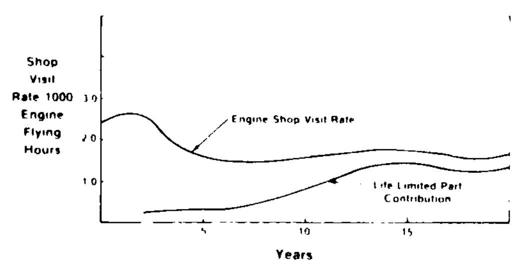

Figure 15

Engine Development — A New Approach

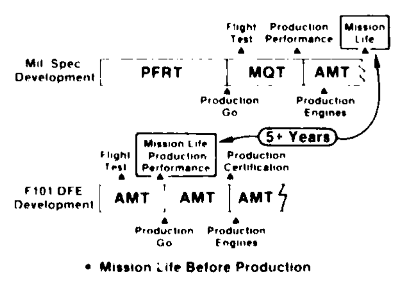

Figure 16

Reference 1

K. Franklin Byrd — General Electric Company, Military Engine Division, Evendale, Ohio

Wayne A. Tall — Air Force Aero. Propulsion Lab., Wright-Patterson AFB, Ohio

"Engine Life Definition Technique — An Approach to Logistics Support Management-Planning"

Technical Paper presented at ASME Winter Annual Meeting, San Francisco — December 10-15, 1978

Figure 1 — Page 2
Figure 2 — Page 3
Figure 7 — Page 3

DISCUSSION

Jack Sammans, Pratt & Whitney, US

One of your major points is that Operating and Support cost is driven by Shop Visit Rate. This was, of course, quite true in the case of earlier aircraft. With the more modern aircraft, the engines can be removed much faster than before. Often a decision must be made at the operational level whether to repair the engine in the aircraft and stand the aircraft down, or change engines and fly.

Do you expect $/EFH to correlate as well with SVR on the present generation of fighter aircraft?

Author's Reply

The data presented, I believe, contains two of our most recent fighter engines. I also believe the majority of the maintenance costs are accumulated at either the intermediate shop or at the depot. Costs associated with flight line maintenance are a minor part of the overall maintenance cost.

In summary, I do expect this correlation to continue. I might add it is necessary to take great care when using the USAF data system to measure engine costs, as there are several quirks associated with how the data is initially entered.

SPECIFICATION DES ESSAIS DE DEVELOPPEMENT POUR UN MOTEUR D'AVION DE COMBAT

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La SNECMA a une production diversifiée de moteurs civils et militaires. En ce dernier domaine, l'association des moteurs SNECMA *aux cellules des Avions Mirage de* Dassault a donné lieu à une famille d'avions dont la réputation sur le plan mondial n'est plus à faire. Rappelons que les moteurs Atar 9C équipent les avions Mirage III, les moteurs Atar 9K propulsent les Mirages IV de la Force Aérienne Stratégique Française, tandis que les moteurs Atar 9K50 sont montés sur Mirage F1 et depuis peu sur Mirage 50. Naturellement, l'évolution des moteurs Atar du 9C au 9K50 a essentiellement été une augmentation de poussée en sec et en PC puisque celles-ci sont passées respectivement de 4,4 T à 5 T et de 6,2 T à 7,2 T. La plus récente version d'avions Mirage, le 2000, ainsi que le prototype Mirage 4000 (biréacteur) sont équipés du moteur simple corps double flux M 53. Nous allons décrire la façon dont a été conduit le développement de ces deux moteurs.

1. STRUCTURE D'UN PROGRAMME DE DEVELOPPEMENT

Chaque type de moteur a son programme propre de développement, qui dépend des applications particulières du moteur lui-même, mais aussi de l'évolution dans la façon d'appréhender la conduite d'un développement moteur. Il existe cependant des catégories d'essais que l'on retrouve toujours dans un programme, mais qui peuvent prendre plus ou moins d'importance selon la manière dont, précisément est conduit ce programme.

En amont, on trouve les essais partiels par pièces, composants ou sous ensembles moteurs ; ce sont les essais de caractérisation des principaux éléments (compresseurs, turbines, chambres de combustion, réchauffe, tuyères, etc...) ainsi que des essais mécaniques destinés à vérifier la validité de conception de certaines pièces.

Dans le même temps, et un peu dans la même catégorie que les essais partiels, on trouve les essais d'équipements (régulation d'ensemble du moteur et tous les équipements associés). Ensuite, viennent les essais sur moteur complet, qui peuvent varier selon la nature de l'essai (essais de caractérisation fonctionnelle, mécanique, d'endurance, ou essais contractuels).

Enfin ont lieu les essais en vol, qui, comme les essais au sol, peuvent appartenir à plusieurs catégories selon les conditions d'essais (vols sur porteur subsonique, ou (et) sur banc volant rapide, puis sur l'avion de dotation)ou selon le caractère contractuel de l'essai.

2. EPREUVES CONTRACTUELLES

Chaque pays ayant une industrie de moteurs d'avions a ses propres catégories d'épreuves contractuelles. En France, tout moteur d'avion militaire a à satisfaire aux épreuves suivantes :

2.1. Epreuves contractuelles au banc sol

Epreuve de puissance (P) : cet essai a pour but de vérifier que le moteur réalise pendant 5 minutes la poussée maxi prévue au contrat. *Il peut y avoir plusieurs essais à* des niveaux croissants de puissance.

Epreuve de qualification pour vol (Q) : il s'agit d'un essai d'endurance de 50 heures réalisé selon des cycles définis dans le contrat. Le but est de démontrer une sécurité fonctionnelle satisfaisante pour autoriser des vols avec un potentiel limité.

Epreuve d'homologation (T) : c'est un essai d'endurance de 150 heures, comportant une partie d'essais dans les conditions sol, et une autre partie dans les conditions d'altitude simulée en caisson. Cette dernière est constituée de cycles basés sur les points de vol principaux de la mission de l'avion, dont les conditions aérothermodynamiques sont exactement restituées à l'entrée du moteur. Il s'agit donc en fait d'une première approche d'endurance dans les conditions de mission simulée.
L'objet de cet essai est de vérifier le comportement mécanique satisfaisant de la machine, dont tous les composants doivent être retrouvés en bon état général à la fin de l'essai. Si tel n'est pas le cas, quand les défauts sont mineurs, une prolongation limitée de l'essai peut être demandée par les services officiels.

Epreuves de performances : leur objet est de vérifier que les performances réelles du moteur sont supérieures aux valeurs de rejet contractuelles figurant dans les clauses techniques officielles.

2.2. Epreuve contractuelle en vol

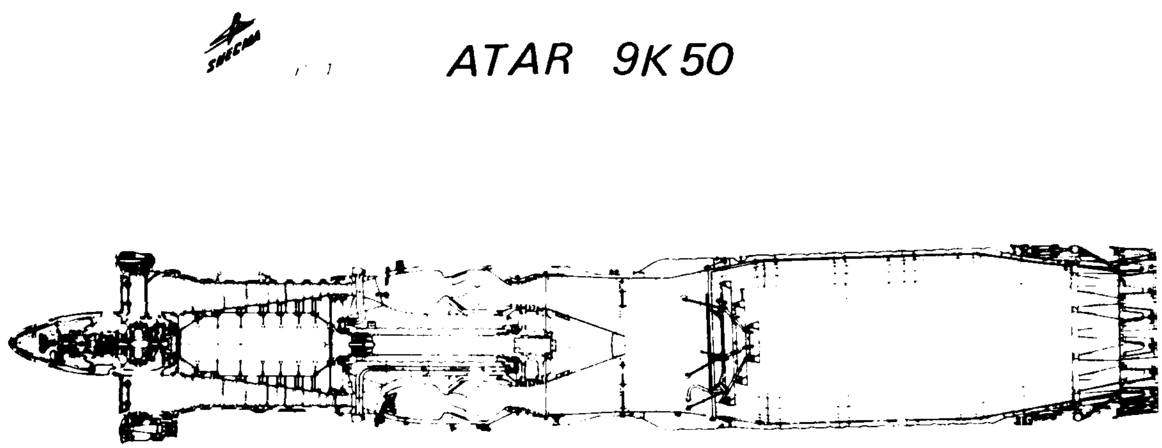
Bon de vol série : les essais sont réalisés sur avion d'arme. Leur but est de démontrer la compatibilité moteur avion et la capacité de l'ensemble à couvrir la totalité des spécifications avec du matériel de série. Autrement dit, à partir d'un échantillonnage de matériel réceptionné en série, les points critiques des spécifications sont vérifiés avec le matériel à priori le plus mal placé pour chacun de ces points critiques. Par exemple, les temps d'accélération sont évalués avec les moteurs dont les réglages sont les moins favorables à ce sujet, que ce soit au point de vue marge au pompage ou temps d'accélération proprement dit.

La campagne d'essais est organisée avec en gros, une période préparatoire sous responsabilité du constructeur avec suivi CEV, et une autre période sous responsabilité CEV uniquement. Il en résulte la définition de consignes officielles d'utilisation du matériel.

Ce type d'épreuve n'existe qu'en France. Sa durée s'étale sur plusieurs mois, car la masse d'essais et de vérifications est considérable. En contrepartie, il faut souligner les conditions remarquables dans lesquelles le matériel entame sa vie opérationnelle. Le Mirage F1-9K50 et l'alphajet-Larzac en témoignent.

3. DEVELOPPEMENT DU MOTEUR ATAR 9K50

Le moteur Atar 9K50 est une version dérivée du moteur Atar 9K propulsant le biréacteur Mirage IV (figure 1) et destiné à équiper l'avion polyvalent Mirage F1.



Les objectifs visés, par rapport à l'Atar 9K, étaient :

. une amélioration de la Cs à basse altitude (pénétration),
. une amélioration de la poussée en supersonique,
. l'adaptation à un monomoteur de combat avec des exigences particulières sur les plans manoeuvrabilité et sécurité de fonctionnement. Dans ce but, les principales modifications introduites ont été :
  - la modification de 2 grilles compresseur pour augmenter le débit et la marge au pompage,
  - une amélioration de la chambre de combustion pour adaptation des températures de parois et des répartitions sortie chambre à une température entrée turbine plus élevée,
  - une nouvelle turbine ainsi qu'un raccordement sortie turbine redessiné afin d'améliorer rendement et durée de vie,
  - également une modification de l'injection du carburant dans la réchauffe pour améliorer le rendement,
  - enfin, de nouveaux équipements pour améliorer les qualités opérationnelles ainsi que la fiabilité fonctionnelle.

Etant donné que le moteur Atar 9K50 est un dérivé d'un moteur existant, son programme de développement a été naturellement différent d'un moteur nouveau.

3.1. Essais partiels

Ils ont porté sur les composants modifiés cités plus haut, et ont été peu nombreux à cause du faible nombre de parties concernées, mais aussi parce que les modifications s'appliquaient à du matériel connu, ce qui est fort différent du cas où le matériel est complètement nouveau. C'est pourquoi la réalisation rapide d'essais moteur fut possible, et l'accent a donc été délibérément placé sur les essais de moteurs complets.

3.2. Essais moteur au banc

3.2.1. Constructeur

Les essais ont été nombreux : 450 essais ont été réalisés depuis la première rotation au banc le 7 février 1968, dans les phases "Développement" et "Exploitation", de nature très diverse :

. essais d'évaluation pour orienter certains choix technologiques,
. mise au point de la solution technologique retenue,
. essais de caractérisation, généralement sur 3 moteurs, pour établir une base de mesures moyennes,
. essais de rétention carter avec rupture volontaire d'une aube de turbine.

3.2.2. Epreuves contractuelles

L'enchaînement des épreuves contractuelles s'est effectué à un rythme très rapide, facilité naturellement par le fait qu'il s'agissait d'un moteur dérivé, et non d'un moteur nouveau. C'est ainsi que l'épreuve de puissance eut lieu deux jours après la première rotation du moteur, et l'épreuve de qualification pour vol fut satisfaite sept mois plus tard.

Quant à l'épreuve T, elle eut lieu un an après l'épreuve Q, dans les conditions mentionnées précédemment, sous contrôle des services officiels naturellement. Il faut souligner encore une fois l'intérêt de ce type d'épreuve, car l'expérience 9K50 en vie opérationnelle depuis maintenant 230 000 heures a montré que jamais un problème de fonctionnement moteur n'a été rencontré en vol.

Les épreuves de performances ont été basées sur des points contractuels types. Deux moteurs de développement ont d'abord été caractérisés, avec confirmation sur deux moteurs série ensuite.

3.3. Essais moteur en vol

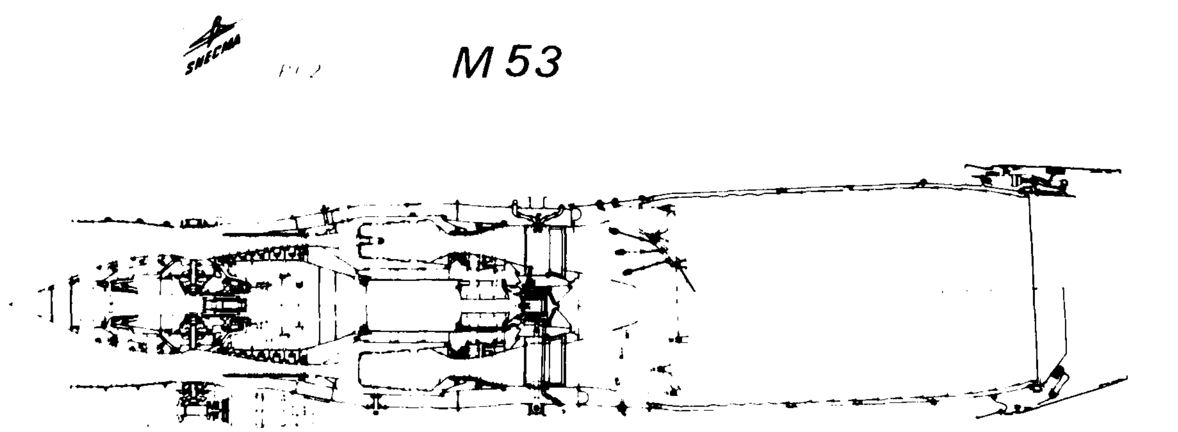
Les essais constructeur ont d'abord été effectués sur avion porteur Armagnac, surtout pour l'étude du comportement du compresseur en vol. Ensuite, un Atar 9K a été transformé pour recevoir les équipements du 9K50 et être essayé en vol. D'autre part, il a été procédé à divers essais de caractérisation en vol des problèmes d'interfaces avion - moteur.

Le "bon de vol prototype" a été réalisé sur avion Mirage III et acquis en février 1970. La campagne d'essais correspondant au "bon de vol série" s'est déroulée en 8 mois pleins répartis sur 2 ans : de novembre 1970 à décembre 1972.

4. DEVELOPPEMENT DU MOTEUR M 53

Le moteur M 53 est un moteur simple corps double flux (figure 2) dont la conception a été voulue simple dès le départ en raison des problèmes de coût et d'entretien.

Fig 2



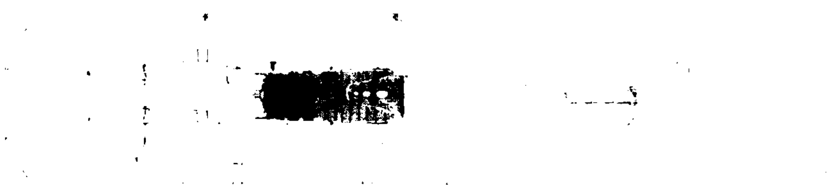
Il s'agit cette fois d'un moteur entièrement nouveau, et non d'un dérivé comme l'Atar 9K50. Sa conception remonte à 1967, et à cette époque l'accent a été mis sur un développement poussé des composants en essais partiels, dans l'idée de réduire les essais moteur.

4.1. Essais partiels

Ce point a donc été l'objet d'un effort particulier. Tous les éléments constitutifs principaux du moteur ont d'abord été essayés, caractérisés, développés en essais partiels. Que ce soient le compresseur, la chambre de combustion, les turbines, les équipements principaux, le circuit carburant, etc...

La plupart de ces types d'essais sont bien connus, nous allons décrire un peu plus en détails un montage moins répandu, utilisé pour la mise au point de la réchauffe. Un banc d'essais partiels a été spécialement construit au CEPr de Saclay pour permettre le développement de la réchauffe échelle grandeur d'une chambre de réchauffe et d'un système PC M 53 (figure 3).

Pl 3 **INSTALLATION POUR ESSAIS PARTIELS DE RECHAUFFE**



En amont ont été effectués les classiques essais de réchauffe en veine bidimensionnelle. La veine tridimensionnelle était donc constituée, pour le flux primaire d'une chambre de préchauffe, aux dimensions importantes compte tenu du débit-volume élevé, d'un dispositif mélangeur de manière à controler et uniformiser la répartition de température à l'entrée du système PC, et pour le flux secondaire d'une alimentation séparée par trois conduits se rejoignant dans une pièce commune pour la transition à une section annulaire à la confluence avec le flux chaud. Au-delà, la veine PC était constituée d'un véritable canal M 53, avec une tuyère à section variable. Les débits d'air et de carburant étaient contrôlés et adaptés aux conditions de vol désirées. Tout le domaine de vol M 53 pouvait être ainsi restitué en conditions amont PC

Les raisons qui ont conduit à réaliser ce banc ont été d'abord la possibilité de tester le système de réchauffe dans des conditions moteur avant la disponibilité de celui-ci. D'autre part, le M 53 posait un problème particulier à résoudre en raison du niveau élevé de température sortie flux primaire, et des problèmes thermiques et surtout des risques de cokéfaction du carburant dans les rampes d'injection que cela posait. C'est pourquoi la possibilité de fonctionner avec du carburant chaud a été réalisée. Ces risques ont donc été quantifiés grâce à l'homogénéité de la température sortie flux chaud, et les solutions appropriées ont été dégagées. Au-delà, bien entendu, ce banc a servi à résoudre d'autres problèmes apparus sur moteur, gagnant un temps appréciable puisque les essais n'étaient pas liés à la disponibilité du moteur.

La réchauffe du moteur M 53 est aujourd'hui bien au point, et la contribution apportée par ce banc a été très importante, ce qui rend probable une approche identique dans le programme du moteur M 88.

Un autre exemple est celui des équipements pour qui les essais de simulation ont été particulièrement poussés et ont constitué une aide appréciable à la conception et à la mise au point de la régulation.

Dans un premier stade, le moteur et tous les équipements ont été simulés. Le moyen mis en oeuvre a été un simulateur hybride capable d'évolutions aux "grands ordres" dont la partie numérique a réalisé la simulation du moteur, la partie analogique réalisant celle des équipements. Cette simulation a été un élément fondamental au stade initial du projet pour tester le comportement dynamique associé aux diverses lois de régulations envisagées, pour choisir la structure fonctionnelle la mieux adaptée et spécifier avec précision les équipements.

Dans un deuxième stade, le calculateur électronique de régulation en vraie grandeur a été couplé au simulateur hybride, moyennant une simulation des interfaces entre le calculateur de régulation et ses entrées/sorties. Il a été alors possible de tester le comportement dynamique obtenu avec le calculateur électronique réel, de mettre au point et d'optimiser les "réseaux correcteurs" et le séquencement des fonctions qu'il assure.

Dans un troisième stade, lorsque les régulateurs hydromécaniques ont été fabriqués, testés sur banc de régulateur et ont fait l'objet d'une mise au point préalable, les essais d'ensemble de tous les équipements importants (hydromécaniques et électroniques) ont été effectués sur un banc simulateur comportant :

. un simulateur du moteur, capable d'evolutions aux grands ordres,
. des servitudes permettant le fonctionnement des équipements hydromécaniques et électroniques,
. des convertisseurs transformant les commandes physiques (débit de carburant, position,...) délivrées par les équipements en des valeurs électriques compatibles avec le simulateur moteur et restituant aux équipements sous forme physique (vitesses de rotation, pressions pneumatiques,...) les paramètres moteur élaborés électriquement par le simulateur.

La mise en oeuvre des régulateurs hydromécaniques en vraie grandeur améliore la représentativité des travaux de simulation et a accru la finesse de la mise au point. Citons notamment la mise au point définitive du réseau correcteur de la régulation tachymétrique pour l'ensemble des conditions de vol, et l'étude des reconfigurations résultant d'une panne du calculateur électronique.

La mise à jour permanente du modèle de simulation du moteur, au fur et à mesure des évolutions du moteur pendant la phase de développement, a permis de préparer sur banc simulateur l'évolution résultante des réglages et l'introduction des nouvelles fonctions dans les équipements, et a entraîné une réduction notable des temps d'essais fonctionnels sur moteur, ainsi qu'une meilleure optimisation des réglages obtenus compte tenu de la possibilité de réaliser des explorations plus larges sur simulateur que sur moteur.

4.2. Essais moteur au banc

4.2.1. Essais constructeur

Le premier moteur M.53-2 a tourné au banc début 1970. Les performances nominales ont été très rapidement atteintes, et une première endurance de 50 heures a ainsi pu être effectuée pour mi-1970.

L'objet des premiers essais a été, en dehors des performances, la caractérisation du comportement mécanique d'ensemble de la machine, puis l'introduction d'éléments définitifs du moteur qui n'étaient pas présents au départ, comme par exemple le relai d'accessoires et la régulation du moteur.

4.2.2. Epreuves contractuelles

L'épreuve de puissance complète (sec et PC) fut effectuée en mai 1971.

Diverses épreuves de 50 heures (qualification pour vol) ont été réalisées ensuite, correspondant à différentes configurations du moteur, surtout au niveau du système de régulation, dont l'évolution était déterminée par les résultats des essais partiels évoqués plus haut.

Un moteur a passé avec succès l'épreuve Q pour vol sur Mirage F1, modifié en mai 1974.

Enfin l'épreuve T (150 heures) a été réalisée entre novembre et décembre 1974. Comme pour l'Atar 9K50, cette épreuve fut réalisée avec une partie de cycles simulant la mission en caisson d'altitude. Tout le domaine de vol a été couvert, avec même certaines excursions au-delà puisqu'un fonctionnement correspondant à Mach 2,4 36 000 ft eut lieu.

4.3. Essais moteur en vol

Les essais constructeur ont d'abord eu lieu sur avion porteur Caravelle, avec un premier vol en juillet 1973. Ils ont eu pour but d'affiner les réglages de la régulation du moteur, ainsi que la mise au point de la réchauffe, dans le domaine de vol de la Caravelle. Des altitudes supérieures à Z = 50 000 ft - V = 150 Knts,ont ainsi été explorées.

Des vols sur banc volant rapide Mirage F1 ont débuté en décembre 1974.

Le premier vol du Mirage 2000 a eu lieu en mars 1978. Le programme bon de vol série est en cours actuellement.

5. DEVELOPPEMENT D'UN MOTEUR MILITAIRE FUTUR

Les deux moteurs dont il vient d'être question ont été développés d'une manière "classique".

L'évolution des conditions d'utilisation des moteurs pour avions de combat, due à l'évolution des performances des avions, a pour conséquence une approche sensiblement différente d'un programme de développement de moteur nouveau, comme le M 88.

La principale caractéristique des avions de combat de la nouvelle génération est leur maniabilité, ce qui exige du moteur, entre autres choses :

. un rapport poussée/masse élevé, pour ne pas pénaliser ce même rapport au niveau de l'avion,
. une grande tolérance aux conditions de fonctionnement : distorsions à l'entrée du moteur, facteur de charge, etc...,
. des temps de réponse rapides en transitoire, ainsi qu'une résistance élevée aux sollicitations cycliques.

De plus, chaque pays possédant une industrie de moteurs d'avions a ses propres normes, et on assiste à une évolution de ces normes qui traduit ce qui vient d'être indiqué. Un exemple typique est celui de la norme US MIL 5000 7 D.

Afin de prendre en compte au mieux cette situation, et dans le but d'aider au choix des solutions techniques réalisant les meilleurs compromis, il a été établi une hiérarchisation des objectifs principaux pour le moteur M 88. Cette hiérarchisation est la suivante :

. La pilotabilité - Ceci signifie que le pilote n'a pas à s'occuper du ou des moteurs dans toute sa mission, aucune consigne particulière de conduite du ou des moteurs ne devant occuper son attention. Cela suppose en particulier des marges au pompage suffisantes pour chacun des compresseurs.

. La masse - ou plus exactement le rapport poussée/masse. C'est un paramètre important pour l'avion, et le ou les moteurs y contribuent pour beaucoup.

. Le coût - La complexité des systèmes d'armes modernes pose des problèmes financiers souvent graves, encore aggravés par les taux d'inflation. Les exemples, parfois spéctaculaires, ne manquent pas. Il est donc important de produire un matériel où les préoccupations de coût auront été largement prises en compte dès la conception.

. La maintenance - En prévoyant une conception adaptée à faciliter la maintenance durant la vie opérationnelle du matériel, c'est le coût global d'utilisation qu'il est ainsi tenté de réduire.

. Les performances - Il ne s'agit pas là des caractéristiques principales (poussée, consommation spécifique) pour les points les plus importants du domaine de vol qui doivent être impérativement respectés, mais des performances dans les zones marginales, ainsi que d'autres types de performances comme, par exemple, les domaines de réallumage en sec et en réchauffe. Tout ce qui est lié à la pilotabilité (temps d'accélération par exemple) fait naturellement partie du point numéro 1.

5.1. Incidence des nouvelles normes sur le développement d'un moteur nouveau

La principale conséquence de ces normes nouvelles est de préconiser des constructeurs de moteurs d'avion, des essais se rapprochant davantage des conditions de fonctionnement opérationnelles dans les cas les plus extrêmes. Comme par exemple :

. essais de démarrage de - 54°C à + 135°C,
. détermination des limites de distorsion à l'entrée (pour 5 configurations moteur - vol différentes dans la 5000 7 D),
. essais de 10 heures avec ingestion de sable,
. essais d'ingestion d'oiseaux plus sévères même que pour les moteurs civils,
. mesures de signatures bruit, radar, infra-rouge, etc...

La norme constitue en fait un canevas pour des spécifications techniques, chaque point devant être pris en compte avec beaucoup d'attention car il est clair que les compromis ne s'en trouvent pas facilités.

Néanmoins, une variété plus grande d'essais de développement au niveau du moteur complet devient ainsi nécessaire, ce qui implique que, pour ne pas augmenter exagérément le volume des essais moteurs, il est nécessaire que celui-ci soit déjà mur au niveau des composants avant leur intégration.

5.2. Essais d'éléments intégrés

Le programme de développement d'un moteur nouveau commence par des essais de composants au banc partiel bien entendu. Ainsi, est-ce le cas pour le M 88 de la SNECMA pour qui, par exemple, la même démarche que pour le M 53 sera effectuée, au sujet de la réchauffe aussi bien que pour les autres parties.

Mais l'évolution des conditions de fonctionnement évoquée plus haut, ainsi que la prise en compte des normes, font en sorte que les essais de composants ne suffisent plus, et qu'un stade intermédiaire vient se glisser entre les essais composants et les essais moteurs devenus plus variés, sous la forme d'essais d'éléments intégrés. Prenons deux exemples : le corps HP et le compresseur BP.

3.2.1. Essais de corps HP

Un paramètre thermodynamique important pour l'obtention d'un rapport poussée/masse élevé du moteur est la température entrée turbine. D'autre part, l'importance des transitoires accroît la difficulté pour les sollicitations thermiques qui en découlent dans les parties chaudes du moteur. Il est devenu nécessaire de tester ces parties chaudes dans les conditions les plus proches possible du fonctionnement moteur, aussi bien en transitoire qu'en stabilisé, en respectant :

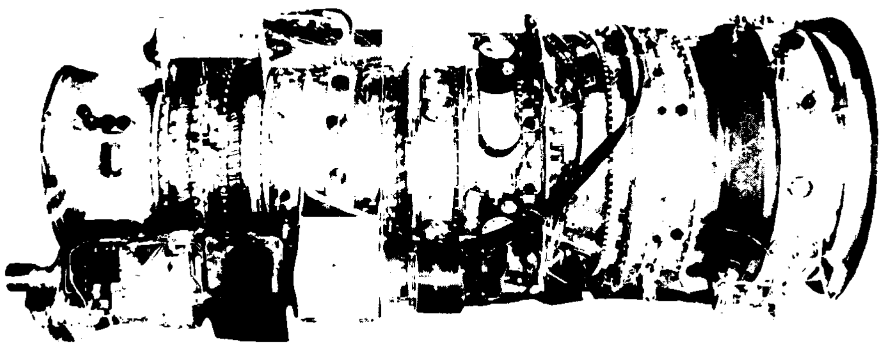
. Les conditions aérothermodynamiques exactes, ce que permettent rarement les bancs d'essais partiels (niveaux de pression par exemple),

. Les conditions limites. En particulier, il est important que la turbine fonctionne avec des répartitions de températures réalistes.

C'est pourquoi le corps HP complet est essayé isolément avec la possibilité de restituer dans un banc atmosphérisé, les niveaux de pression et température correspondant aux conditions sortie BP. C'est ce qui est en cours pour le futur moteur M 88. Il sera possible ainsi d'acquérir une certaine endurance sur cette partie essentielle du moteur avant que le moteur complet ait fonctionné.

En amont même, dans le cadre de programmes de recherches, de semblables montages sont effectués pour valider des technologies des parties chaudes, aubes de turbine en particulier, qui sont destinées à être utilisées dans un programme moteur. C'est le cas du montage "Dextre" (figure 4) à la SNECMA, qui est en fait un corps HP dont seules la chambre de combustion et la turbine sont représentatives d'une technologie de moteur avancé, le compresseur HP étant un composant de servitude.

F 4 DEXTRE



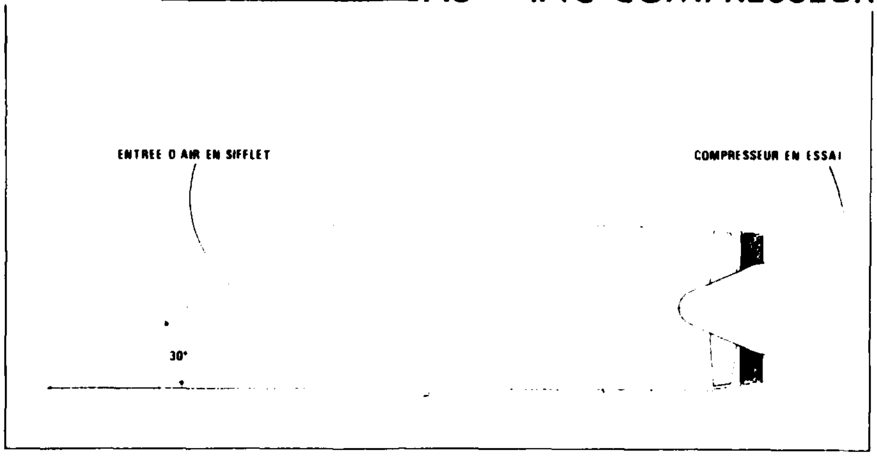
Un tel montage permet le fonctionnement d'un ensemble chambre de combustion-turbine HP dans des conditions rigoureusement identiques à celles d'un moteur évolué, aux points de vue niveaux de pression et température, et répartitions de température, permettant aussi d'effectuer des fonctionnements transitoires, donc des cycles d'endurance. Le montage Dextre va bientôt aborder une campagne d'essais avec cycles, assurant ainsi une étape importante dans le travail de développement d'une turbine à hautes performances.

3.2.2. Le compresseur BP

De même qu'initialement, le compresseur BP était essayé isolément au banc partiel, des tentatives de simulation des conditions de fonctionnement opérationnel sont de plus en plus poussées. Sous la forme d'abord de distorsions stationnaires, restituées par des dispositifs grilles, ce qui n'est pas nouveau. Les cartes de pressions à réaliser sont déterminées à partir des résultats d'essais d'entrée d'air avion en soufflerie, confirmées quand c'est possible par des mesures effectuées sur les mêmes entrées d'air en vol. Le problème des distorsions instationnaires est plus complexe, parce qu'il pose d'abord une difficulté de mesures en soufflerie et encore plus en vol, et d'autre part parce que même connaissant l'objectif à simuler, sa réalisation est plus délicate que pour du stationnaire. Au départ, en ne connaissant ni les objectifs à simuler, ni les dispositifs permettant la restitution des composants instationnaires, on se bornait à renforcer les distorsions stationnaires.

Il est envisagé, par exemple, des entrées d'air sur banc compresseur, taillées en sifflet (figure 5) qui génèrent des tourbillons dans l'alimentation du compresseur, simulant ainsi des distorsions instationnaires. La représentativité de ces distorsions, comme il l'a été souligné plus haut, reste bien sûr à vérifier et il reste beaucoup de travail à faire dans cette voie.

Pl 5 SCHEMA DE DISPOSITIF DE SIMULATION AU BANC COMPRESSEUR



Une expérimentation de telles entrées d'air a déjà été faite en soufflerie, et a donné des résultats encourageants. D'autre part, une expérience devant un compresseur à l'échelle 1 a été réalisée récemment.

5.3. Exemples d'essais nouveaux sur moteurs complets

5.3.1. Essais de simulation de fonctionnement opérationnel

Il est apparu que les essais classiques d'endurance tels qu'effectués sur Atar 9K50 et sur M 53 ne suffisaient pas toujours à représenter le comportement du moteur durant sa vie opérationnelle. De plus, les exigences nouvelles évoquées plus haut ne peuvent que contribuer à renforcer cet écart. D'où la recherche de formes d'endurances se rapprochant davantage des conditions réelles d'utilisation, celles-ci étant d'abord étroitement liées au type et à la mission de l'avion.

Un travail important a été réalisé aux USA à cet égard. On peut signaler par exemple, les travaux de Ogg et Taylor (accelerated Mission Testing of Gas turbine engine). De même les travaux de G.E. "Engine life usage expérience of YF 17-YJ 101 flight and ground testing" par TAOHA, ont conduit à définir un cycle pour essai d'endurance accélérée dans les conditions sol par simulation de la mission de l'avion YF 17. La norme MIL même, préconise un type de cycle pour essai MQT. Il est évident qu'une telle procédure sera désormais suivie pour un moteur nouveau.

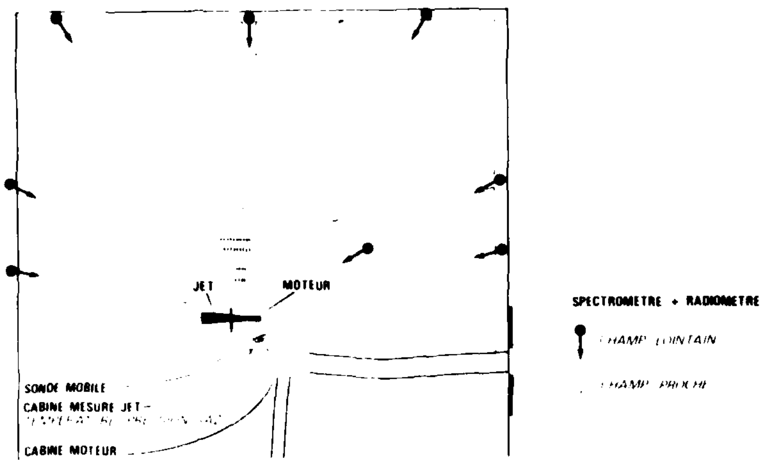
5.3.2. Mesures de signature infrarouge

Ce type d'essai fait partie de la norme MIL. Des mesures de signature IR ont été effectuées sur avions en vol et au sol, mais rarement, si même jamais, sur moteur isolé au banc sol. C'est pourquoi, disposant d'une installation de banc d'essais à l'air libre, celle-ci a été utilisée pour de telles mesures sur M.53.

L'installation est représentée (figure 6), et permet d'effectuer des mesures par spectromètres, plus précises que par radiomètres. Outre une caractérisation plus fine du spectre IR du moteur, ceci permet d'identifier plus précisément les sources, de les quantifier, et d'en tirer des données utilisables pour la conception des moteurs futurs.

Pl 6

BANC D'ESSAIS MOTEUR AU SOL POUR MESURES DE RAYONNEMENTS INFRA-ROUGES



6. CONCLUSIONS

Ce bref exposé a permis de montrer l'évolution qui a eu lieu en général, et à la SNECMA en particulier, dans la conduite du développement de moteurs pour avions de combat, au cours des dernières années. Le moteur M 88 destiné à la propulsion de la génération des avions de combat de la décennie 90 concrétisera pleinement cette évolution, qui s'est déjà traduite par la mise en place d'essais de composants intégrés, au niveau des parties chaudes en particulier (programme Dextre).

Ainsi, les essais de moteurs complets seront-ils plus spécialement orientés vers les simulations de fonctionnement opérationnel. Notre objectif final est d'aboutir ainsi à un propulseur qui répondra particulièrement bien aux exigences des nouveaux avions militaires, que ce soit sur le plan des performances opérationnelles ou sur celui des coûts.

DEVELOPMENT OF TEST REQUIREMENTS FOR CIVIL AND MILITARY AUXILIARY POWER UNITS

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SUMMARY

The history of the gas turbine Auxiliary Power Unit (APU) is reviewed to indicate the wide variety of design and usage requirements that have evolved. Particular emphasis is given to the requirement for unattended automatic operation responsive to multiple, variable power demands. Early units were of simple design. Increased utilization, particularly on board civil and military aircraft, led to need for higher performance and reliability, resulting in more sophisticated gas turbine designs and control systems.

In the initial applications with military aircraft, an APU start was followed by a limited period of APU operation. This resulted in a high starts-to-hours ratio of 10:1. The current ratio has decreased to 2:1 for civil applications.

Development and proof testing programs for civil and military requirements are formulated as a function of specific user needs and the similarity of the APU and its components to proven designs.

Each installation of the APU on-board aircraft requires tailored designs for such locations as engine nacelles, wheel wells, and tailcone sections. Compatibility and performance tests are conducted to confirm analysis and to develop the installed system. Proof testing at ground level is recommended to substantiate reliability, followed by the final flight test phase.

INTRODUCTION

The Auxiliary Power Unit (APU) is a gas turbine engine that has a variety of uses, both on-board and as ancillary equipment to support jet aircraft. APUs have been derived from propulsion engines, and propulsion engines have been designed using APU components as baselines. APU testing requirements depend upon the particular application and upon the expected life cycle. The primary design and operational criteria most significantly different from propulsion engine counterparts are:

(1) Operation is fully automatic and unattended and is responsive to a varying load demand.

(2) Power delivered is in the form of compressed air, electrical, hydraulic, and/or shaft power.

(3) Loads can be multiple and simultaneous, with priority given to one type.

BACKGROUND

During the late forties and early fifties, the increasing sizes of military turbine propulsion engines resulted in the development of the pneumatic starting system as a replacement for electric starting. Initial APU designs were simple integral-bleed radial-flow turbomachines of about 3:1 pressure ratio and were used for aircraft cold weather heating and turbine engine starting. The first on-board APUs were in Navy flying boats and provided electrical power only. The first on-board pneumatic units were installed in the C-130. APUs installed in ground carts provided starting air for the Air Force Century Series fighters, Navy fighters, bombers, and patrol aircraft. Some of these units are shown in Figures 1 through 6. It became evident during this period that utilization consisted of a large number of predominantly start/stop cycles with relatively short loading periods. Time between overhaul was relatively short and was expressed in hours and/or starts, whichever occurred first.

The advent of turbine-powered commercial airliners expanded the use of ground pneumatic starting units and created the need for the APUs with both bleed air and electric power capability. During the early sixties, the first on-board commercial airline APU installation was introduced. An FAA Technical Standard Order TSO-C77 was created for U.S. civil APU applications. It soon became evident that the APU usage in commercial applications had been significantly underestimated. The starts-to-hours ratio decreased, and operating time advanced to about 3000 to 3500 hours per year.

The first wide-body aircraft, the Boeing 747, required the design of the largest APU, the Garrett GTCP660. This is a 24-pound-per-second axial machine. A cross-section is shown in Figure 7. It is an integral-bleed engine with relatively low thermodynamic efficiency due to a pressure ratio of approximately 4:1 set by the aircraft bleed-system requirements.

The anticipated operational time of commercial APUs resulted in more consideration being given to higher efficiency for the next commercial applications, such as the McAir DC-10 and the Airbus A300 aircraft. These aircraft utilize the Garrett TSCP700, a sophisticated APU of unique two-spool design, shown in Figure 8. The engine develops 8:1 pressure ratio. The high-pressure spool drives the accessories and the 400-Hz electrical power generator and is controlled at constant speed with varying electric and pneumatic load demands. Variable turbine inlet guide vanes automatically control the low-pressure turbine and compressor speed in response to bleed air demand from either the air conditioning or main engine start system. The 4:1 pressure air for the aircraft pneumatic system is bled aft of the low-spool compressor.

A sophisticated approach to the cost-of-ownership determination for on-board APUs in the advanced Boeing and Airbus airliners has evolved from recent airline experience with APUs. Consideration for much higher fuel efficiency, longer life, higher dispatch reliability, minimum weight, and resulting life-cycle costs have led to the creation of a higher performance gas turbine design with advanced control and health monitoring systems. The APU selected to meet these criteria is structured from the power section of an existing production 840-hp turboprop engine of 11:1 pressure ratio, with minor modifications to provide higher cyclic life, and derated in speed and temperature. This basic power section, which has high-thermodynamic performance, drives a load compressor that provides pneumatic power at the desired 4:1 pressure ratio and an accessory gearbox with a reduction output pad for mounting an electrical generator. A cross-section of this engine, the Garrett GTCP331, is shown in Figure 9. A full-authority digital controller provides automatic unattended operation with electric power priority. Fault isolation monitoring and Built-in Test Equipment (BITE) are provided. The unit starts at up to 25,000 feet (7620m) and operates to 40,000 feet (12192m). The digital controller operates with the aircraft air conditioning system to minimize air and fuel usage. Air for main engine starting and for driving an air turbine motor hydraulic pump also is provided.

Select commuter and business propeller and fan jet aircraft also have chosen on-board APUs. Each installation is tailored to fit the needs of the aircraft.

Recent on-board military usage of APUs in the mid and late seventies include the new lightweight fighters, attack aircraft, and helicopters. This has called for new classes of advanced technology APUs in the smaller sizes, up to 250 hp. The need for higher thermodynamic performance presents special design technology requirements in this small size class (3 pounds/second) in order to meet the weight, performance, and reliability needs. APUs on board the F-15 and the A-10 aircraft are shown on Figures 10 and 11. The latest of these lightweight, high-performance units is the Garrett GTC36-200 APU designed for the F-18 fighter shown on Figure 12. The concept employed in this unit is similar to that described above--a power section driving a load compressor at shaft speed, with both the engine and the load compressor sharing a common air inlet. Variable inlet guide vanes automatically control the load compressor air delivery in response to a demand signal.

New requirements for aircraft ground support APUs are being met by units such as the Advanced Ground Power Unit (AGPU) shown on Figure 13, currently undergoing field operational testing for both military and civil usage. This unit has an APU installed in a self contained enclosure. The unit delivers compressed air for main engine starting, 400-Hz electrical power, dc electrical power, and 3000-psi hydraulic pressure for servicing. Batteries provide APU starting, and fuel for two-hours operation is included. Bleed-air hose and electrical cabling are stored within. Acoustic treatment is included in the exhaust and inlet, and inlet air filtering is also featured. A motor wheel drive provides power for aircraft ramp towing. The basic APU used in the AGPU enclosure is also utilized aboard business jets and military attack aircraft and helicopters.

Increased fuel costs have revived the need to consider the recovery of heat from the exhaust of APUs utilized in ground operations. APU heat recovery developments have been executed at various times since the mid-fifties. The small-sized APU with its low-pressure compressor is particularly suited to stationary or rotary recovery of exhaust heat. Figure 14 shows a recuperated 30-kw generator set with a stationary exhaust heat exchanger (recuperator) which has demonstrated 40-percent fuel savings at full load.

ENGINE TESTING

During the introductory period of ground and airborne gas turbine APUs, simplified design and qualification criteria were used. Although environmental design and testing requirements borrowed from propulsion engines were directly applicable, the early reliability development and qualification tests proved inadequate for the thermal-shock cyclic operation that was to prevail. Indeed, one of the first 200-hour qualification endurance tests was conducted with only one start and shutdown. Design life limits centered around the turbine with the general requirement of 1000-hours life at rated temperature with materials data judged inadequate by today's standards.

In the early fifties, field experience with pneumatic starting of turbojet engines resulted in a start/stop development cycle at AiResearch, depicted in Figure 15. Forced cooling with the engine non-operating and with up to two-hour shutdowns were used to

accelerate the thermal cycling. This cyclic loading pattern was eventually adopted as the reliability demonstration start/stop cycle of the military testing specification, MIL-P-8686. First use of this test cycle produced major failures within a few hundred starts. Design iterations with emphasis placed upon thermal transient analysis evolved to solve the thermal-shock problem. As a result, in 1955, a 10,000-start/stop qualification test (20,000 load applications) was successfully completed on the Air Force MA1A start cart.

This design and testing expertise was applied subsequently to the continuous line of APUs designed to meet specific operational requirements of both start cycles and hours of operation.

User specifications require variations of cycle/time reliability qualification testing. Examples of the requirements for production APUs for on-board military aircraft that have recently been qualified are shown below and on Tables I, II and III.

Garrett APU Model	Aircraft	Reliability Qualification
JFS-190	F-15	2000 starts; 150 hours
GTCP36-50	A-10	1250 starts; 1250 hours
GTCP36-200	F-18	8400 starts; 1000 hours

F-15 qualification was unique because the APU system is an integral part of the propulsion system. The F-15 APU system includes the Jet Fuel Starter (JFS), the gearboxes and drives for hydraulic pumps and alternators. The testing of the F-15 system did not follow the usual qualification requirement for other APUs. Many aircraft and propulsion system requirements had to be met because of the essentiality of its operation. These system requirements are presented in Table I.

Service operation of the A-10 has revealed a higher APU starts-to-hours ratio than shown above. Recent qualification testing of a modified fuel control system required 6000 starts and 1250 hours to the schedule shown on Table IV.

These tests are electronically programmed to operate the complete repetitive time and condition cycle automatically and unattended around the clock. Fault monitoring and shutdown are included. Manual shutdowns periods are executed to conduct prescribed timely thermodynamic performance calibrations or scheduled maintenance per operational and maintenance handbooks.

Civil certification requirements of the APUs are satisfied by conformation to Technical Standard Order-C77 (TSO-C77) and the Joint Airworthiness Requirements (JARs) and are applicable worldwide. Special conditions are sometimes added by customers. Table V shows these requirements, for both essential and non-essential applications. Figure 16 shows how requirements are combined to satisfy simultaneous military/civil applications of an APU.

Special attention is directed toward rotor integrity testing. This testing is conducted in the operating engine at maximum temperatures and at 105 percent of overspeed protective shutdown speeds which range between 107 and 110 percent of rated speed.

Strain measurements are performed upon the critical compressor and turbine rotors. Locations of these gauges are determined by component tests coupled with analysis.

Containment testing of blades and rims of high-speed rotating components is common to most turbomachinery; however, hub containment testing is unique to APUs. This type test has been developed over a period of many years, requiring extensive expertise in providing maximum containment capability at minimum weight penalty.

To determine that the objectives of the hub containment test are met, the test is run at maximum allowable speed, temperature, and stabilized conditions to assure that all parameters are met or exceeded under the most adverse conditions that would be met in service. Determination of the fusing method is made by analytical methods followed by empirical whirlpit tests to fine-tune the speed and temperature at which component will separate. Final tests are run in a fully operating engine.

APU development test programs are tailored to fit the needs of the application and the final proof testing requirements. The program hardware content and quantity of testing are a function of several variables. The baseline consideration is the degree to which an extension of the state of art is applied to given components for system, thermodynamic, or reliability performance. Minimum risk is achieved with existing or modestly scaled components. Many different APU models have been developed and produced by Garrett since 1948 as shown on Table VI. After unique initial design and manufacturing knowledge was acquired and proven for this small-size high-speed turbomachinery, accelerated development became possible. Utilizing a mix of existing and/or modest upgrades of turbine, compressor, and accessory designs from an ever-increasing pool resulted in low-risk, relatively short test programs which required modest hardware investments. In most of these instances, four to six equivalent program engines would fill the need. Two thousand to 3000 hours of engine system development with the associated ratio of about 6:1 start cycles-to-operating hour were completed prior to proof testing.

Developments which require new or major departure from existing components require more extensive program content. In these cases, thermodynamic and mechanical rig component tests are required to a degree commensurate with the departure from existing tested hardware. Engine development in this case can require up to 10 or more program engines and up to 10,000 hours of testing. The initial step is mechanical checkout and operating capability. As soon as the control system is operational and performance is close to or meeting requirements, initial full envelope testing must be explored. This has been achieved on most engines within four months of initial run/build. The unique requirements for static altitude starting and operating up to 40,000 feet after cold soaking and -65°F at sea level poses a severe range of problems for fuel insertion, combustion, controller and start system needs. These component designs must be established as soon as possible to maximize usefulness of all other program testing. The envelope limits set the acceleration fuel schedules which are major life determinants for the turbine section. Obviously, changes that are applied during the development program that affect starting or thermodynamic performance require repeated altitude environmental checkout. As few as two and as many as 10 altitude environmental tests are required including final proof testing. Garrett has several high-altitude, environmental chambers which are used extensively for these purposes.

Continually increasing demands have been placed upon improving the thermodynamic performance, life, and life cycle cost of the APU, along with the expectations of all users regarding minimum attention be paid to the APU during its operational life. These needs require special control system developments, especially when coupled with the requirement for unattended operation in the mode of responding or reacting to unscheduled load demands of different types.

Control systems have evolved from simple hydromechanical governing with acceleration fuel scheduling via pneumatic temperatures limiting, through various degrees of electromechanical controls to the latest design--a full authority, digital electronic engine controller.

The two most recent APU developments at Garrett utilize gas turbines driving load compressors that require advanced control systems to extract maximum performance with minimum fuel consumption. One of these, the Garrett GTC36-200 for the F-18, employs a load compressor whose bleed air output is modulated by an actuator that controls inlet guide vane positions. Compressor surge protection is provided by a modulating surge bleed air valve controlled by an airflow sensor. The other is the Garrett GTCP331-200/-250 being developed for the advanced airliners under development by Airbus Industrie and Boeing. This APU is also a load compressor type utilizing a full authority digital electronic controller. Closed loop feedback systems are employed in engine acceleration and compressor load control, which can be adjusted by electrical signal levels from the aircraft airconditioning system. An electronically controlled surge protection bleed valve is also used.

These two programs, for example, require major consideration for the development of the controls for engine/aircraft system operation. Several engines must be available for development of the various components in system operation and the selection of the locations of the required sensors. The system must be responsive to sudden load demand changes, both electrical and pneumatic with priority given to the electrical requirements. Consequently, considerable iteration of transient response characteristics must be developed within the full operating envelope (altitude and temperature). Concurrently, starting and acceleration schedules must be fine-tuned. Since the APU must operate unattended, failure mode analyses result in several protective shutdown features, some of which are intentionally by-passed under the emergency in-flight operating mode. In addition, a fault monitoring and panel indicator is provided as shown in Figure 17. This particular APU development program utilizes about 15 engines and will complete approximately 8500 hours and 20,000 starts.

Two production engines will be placed on extended reliability testing with a load schedule to simulate the intended airline life cycle.

One important characteristic of the commercial airline operation that is impractical to simulate in the test cell on a long term basis is the normal condition wherein the APU operates on the ramp for main engine starting and airconditioning and is then shut down at takeoff. The APU is inoperative during most of its service life at the flight altitude, soaking at the extreme cold temperatures. The aircraft descends, lands, and the APU is started immediately before or after landing. This provides for an extension of the thermal-shock envelope which must be accommodated in service.

COMPONENT TESTING

Mechanical

New accessory and controls components are kept to a minimum. Successful designs are modestly scaled when necessary. Judgments are made whether substantiation by similarity is adequate or actual bench component testing is needed. When tests are required, the established military tests are used as baselines. Where practical, new components from multiple sources are subjected to continuous unattended cyclic testing.

The fuel controller is always tested for performance range. When a new design or major redesign is used, it is subjected to full or major portions of military specification tests.

Air-to-oil and air seal rigs have been used extensively to develop and establish performance through the operating ranges. Endurance tests of air-oil seals are no longer performed on rigs but accomplished during engine testing.

Bearing design expertise has advanced to the point where anti-friction or journal rigs are not generally used. Instead, the entire rotor system development is utilized. An example is shown on Figure 18. Most Garrett APUs operate above the second shaft critical speed. In one case, the engine operating speed is above the third critical. Past rotor systems utilized fixed bearing mounts and mechanically spring mounted systems. Current designs exclusively use the viscous damped squeeze film design. The rotor rigs are utilized for sizing the system elements, confirming the critical speeds, and tailoring the oil damping needs.

Structural tests are conducted on new designs for rigidity and pressure vessel capabilities. Structural and system vibration tests are conducted in a special facility with the engine non-operational. A shaker table is vibrated with the engine mounted per installation requirements through the specified frequency and amplitudes.

Rotor integrity is verified via actual temperature and strain measurements in the operating engine. Bench testing of rotors includes the current state of art stress vibration and burst requirements.

The smaller APUs provide the advantage of using cast rotors, thereby having advantages of the their high-temperature properties for turbine wheel use. These cast wheels are subjected to overspeed and burst testing in spin pits to confirm design and material capabilities in the actual part. In addition, new designs are subjected to accelerated thermal cycling by "dunking" the wheel into a fluidized bed. These repeated step thermal loads reveal configuration capability to absorb high-compression/tensile cycling at various locations of the wheel. Although it is not a quantitative measure per se, experience provides an excellent relative measure of design capability and survivability in engine cyclic operation.

Aerodynamic

Many combinations of radial and axial configurations have been utilized in Garrett APUs. These configurations have consisted of radial compressor(s) with axial turbine(s), axial compressors with radial turbine(s), and combinations of axial and radial compressor with combinations of axial and radial turbines. In all cases, the actual engine housings and rotors are used in the thermodynamic testing. The turbines have been confined to cold testing, which has proven quite satisfactory and is conducted only when there are significant sizing or configuration changes.

Compressor testing is conducted on rigs with separate isolated turbine drivers. These are also full scale utilizing engine hardware. Particular attention is paid to the inlet configuration and its effect upon performance. For performance and safety reasons, the air must be ducted to the inlet of the compressor. For minimum weight, single entry ducts are desirable; therefore, the inlet plenum must be designed carefully to avoid pressure distortion and unfavorable swirl at the compressor entry. Specification distortion limits are provided at the plenum entry simulating limited air supply. The large bank of compressor and turbine design data available allow contingency testing only in some engine programs.

INSTALLATION COMPATIBILITY AND FLIGHT TESTING

On board APUs are installed in various aircraft locations such as tail-cone section, wheel wells, and engine nacelles.

In order to ascertain the suitability of an on-board APU installation, it is necessary for the airframe manufacturer to conduct a series of tests. These fall into two general categories: compatibility tests (normally conducted with a compatibility test fixture or with the aircraft on the ground) and flight tests.

The purpose of these tests is to determine the compatibility of the APU with the aircraft installation and to ascertain that the installation environment is in accordance with Garrett requirements for satisfactory operation of the APU and associated equipment. The aircraft installation consists primarily of the APU enclosure, generator and controls, inlet and exhaust ducting systems, actuators and doors, fire extinguishing equipment, acoustic treatment, accessory cooling system, bleed-air system, fuel supply system, d-c power supply, and aircraft-furnished elements of the APU starting system.

The tests are intended to provide advance information concerning the effect of aircraft equipment on the APU operational characteristics and to reveal at the earliest practical time whether design changes are required to achieve a satisfactory installation. The recommended compatibility tests are listed on Table VII. Complete coordination and liaison between the airframe company and Garrett are required in preparation of the test program and in conducting of the testing.

Recommended testing defines ground test requirements for a complete compatibility checkout. Generally, the installation designer will find it economically advantageous and more expeditious to make use of a compatibility test fixture or APU compartment mock-up for this testing. Some examples are shown on Figures 19 and 20.

Normally, this test fixture will be used in lieu of the actual airplane for most of the compatibility testing. The most significant advantage of conducting the compatibility tests in a fixture as early as possible is to reveal deficiencies in sufficient time to permit corrective action and the substantiation thereof, and to enable the necessary changes to be made in production aircraft. This test fixture offers the additional multiple advantages of:

- Accomplishing testing prior to aircraft availability
- Decreasing amount of time that flight test aircraft are required for ground test operations
- Providing the capability of more complete instrumentation as shown on Figure 21
- Facilitating use of automatic programming for cyclic proof testing.

Usually, the aircraft manufacturer will supply the compatibility test fixture to Garrett and arrange for the testing to be accomplished in the AiResearch test facility.

The compatibility test fixture consists of the actual section of the aircraft or a full-scale replica, encompassing the APU compartment and all aircraft systems interfacing with the APU. It should be constructed of identical materials and type of structure and should include the engine mounting system, driven accessories, identical wiring and piping, inlet and exhaust ducting, fire detection, and fire extinguishing components. It should also include fuel system, boost pump, starting system (electric or hydraulic), electrical system (ac and dc), battery, and battery-charging system.

After a satisfactory installation has been proven by test, an accelerated service cycle test is recommended, particularly if the aircraft is to be used in scheduled airline service. This extension of the installation compatibility test has proven worthwhile in the past for a number of APU installations, by revealing design or equipment deficiencies under simulated service conditions. Thus, corrective action may be expedited prior to delivery of aircraft. Compatability testing has been conducted on many civil and military engines as shown on Table VIII.

Flight tests are required to determine if starting and operation of the APU, as it is installed in the aircraft and in operating and non-operating APU environments, are satisfactory and in accordance with the APU model specification throughout the required operating envelope of the aircraft. The recommended tests are listed on Table IX. It may also be desirable to combine some of these tests with aircraft certification testing, particularly those concerned with certification of the APU installation.

Instrumentation locations used for flight tests should duplicate those in the compatibility tests, wherever possible, in order to provide a comparison between static and flight conditions.

Flight testing in many cases has resulted in installation or aircraft surface changes. Inlet and exhaust pressure conditions are measured under varying attitude and flight speed conditions. If these are not favorable at the inlet or exhaust, modifications are necessary to provide proper skin surface pressure/flow conditions for starting operating or non-operating modes. Figure 22 shows some typical aircraft modifications that have been made.

For those installations where starting and operation of the APU in flight is not required, flight test requirements are confined to determination of the presence of windmilling, the speed and direction of the windmilling, the vibratory environment the loads at APU mount points, and whether leakage from unrelated aircraft systems can cause flammable fluids to enter the APU inlet or exhaust systems.

SYSTEM CYCLE TESTING

After the aircraft installation compatability test is completed, the test fixture may be used for a system cycle test which is recommended to uncover APU installation system problems before the production aircraft is available.

A typical 1000-start cycle is as follows:

(a) The air inlet and exhaust doors, are required, are operated as in the installation.

(b) The gas turbine is started and accelerated to governed speed.

(c) The gas turbine is loaded in accordance with the following schedule:

No load for 1 minute Full load for 5 minutes No load for 1 minute	To simulate actual shaft and bleed air loads in aircraft

Full load for this test is defined full electrical load with sufficient bleed-air load to give the rated turbine temperature.

(d) The gas turbine is shut down for 5 minutes.

All doors are closed as in the installation. An appropriate soaking down time(s) is exercised.

(e) Steps (a) through (e) are repeated to accumulate a total of 1000 cycles and 100 hours.

Prior to and upon completion of the 1000-cycle test, the installed gas turbine is required to demonstrate the sea-level performance output ratings of the APU Specification, adjusted to cover the installation losses.

CONCLUSIONS

The Auxiliary Power Unit is a gas turbine engine that is designed for a wide variety of applications.

Operational criteria most significantly different from propulsion engine counterpart, are:

(1) Operation is fully automatic and unattended and is responsive to a varying load demand.

(2) Power delivered is in the form of compressed air, electrical, hydraulic, and/or shaft power.

(3) Loads can be multiple and simultaneous, with priority given to one type.

Baseline military and civil qualification tests exist as MIL-P-8686, ADS-17, FAA TSO and JAR's.

Supplementary qualification tests are invariably added by customers.

Complete test requirements are formulated unique to each APU and are based upon the qualification proof requirements plus:

(1) Design maturity.

(2) Reliability requirements.

(3) Intended usage.

(4) Installation conditions.

TABLE I. MILITARY SPECIFICATION APPLICABLE TO THE JET FUEL STARTER 190 USED ON THE F-15 AIRCRAFT

Military Specification Number	Title
MIL-E-5007C-1 11 October 1966	Engines, Aircraft, Turbojet and Turbofan, General Specification for
MIL-G-6641B 3 May 1968	Gearbox, Aircraft Accessory Drive, General Specification for
MIL-C-45662A 9 February 1962	Calibration System Requirements
MIL-STD-210A-1 30 November 1958	Climatic Extremes for Military Equipment
MIL-STD-704A-1 30 November 1968	Electric Power, Aircraft, Characteristics and Utilization of
MIL-STD-781B-1 29 July 1969	Reliable Tests: Exponential Distribution
MIL-STD-810B-1 28 July 1969	Military Standard; Environmental Test Methods for Aerospace and Ground Equipment
MIL-STD-882 15 July 1969	System Safety Program for Systems and Associated Subsystems and Equipment; Requirements for

TABLE II. F-18 GTCP36-200 LIFE CYCLE/RELIABILITY DEMONSTRATION CYCLE

(A) Conduct Steps A1 through A6 (28 Times)		(C) Conduct C1 through C6 (3 Times)	
A1 Unit Start		C1 Unit Start	
A2 Ready to Load	15 seconds	C2 Ready to Load	10 seconds
A3 Main Engine Start Mode	36 seconds	C3 Main Engine Start Mode (Motoring)	240 seconds
A4 Ready to Load	15 seconds	C4 Ready Load	10 seconds
A5 Main Engine Start Mode	36 seconds	C5 Main Engine Start Mode (motoring)	240 seconds
A6 Unit Shutdown	*	C6 Unit Shutdown	
(B) Conduct B1 through B4 (4 Times)		(D) Conduct D1 through D4 (7 Times)	
B1 Unit Start		D1 Unit Start	
B2 Ready to Load	15 seconds	D2 Ready to Load	120 seconds
B3 Main Engine Start Mode	36 seconds	D3 Subsystem Checkout Mode	1800 seconds
B4 Unit Shutdown	*	D4 Unit Shutdown	*

*Time between unit shutdown and subsequent start will be two minutes minimum.

Steps A through D (5 hours) shall be repeated 200 times to total of 1000 hours and 8400 starts

TABLE III. A-10 GTCP36-50 ORIGINAL QUALIFICATION TEST

- o 1250 hours per sequence shown
- o Sequence is 10 hours long
- o 1250 starts required
  - o 30 starts must be preceded by 2 hour shutdown
  - o 20 starts must be made following rolldown

SEQUENCE

Step No.		Step No.	
1	5 min at no-load	14	5 min at no-load
2	1 hr, 45 min, 30 sec at 25%	15	2 min, 15 sec at max
3	2 min, 15 sec at max	16	5 min at no-load
4	5 min at no-load	17	1 hr, 45 min, 30 sec at 100%
5	2 min, 15 sec at max	18	2 min, 15 sec at max
6	5 min at no-load	19	5 min at no-load
7	1 hr, 45 min, 30 sec at 50%	20	2 min, 15 sec at max
8	2 min, 15 sec at max	21	5 min at no-load
9	5 min at no-load	22	1 hr, 45 min, 30 sec at norm shaft/min bleed
10	2 min, 15 sec at max	23	2 min, 15 sec at max
11	5 min at no-load	24	5 min at no-load
12	1 hr, 45 min, 30 sec at 75%	25	2 min, 15 sec at max
13	2 min, 15 sec at max		

Steps 1 through 25 = 10 hours; therefore, run sequence 125 times.

TABLE IV. A-10 GTCP36-50 REQUALIFICTION TEST

- o 1250 hours per sequence shown
- o Sequence is 12.5 minutes long
- o 6000 starts required

Step No.	Condition
1	Start APU
2	10 seconds no-load
3	60 seconds full-bleed load
4	10 seconds no-load
5	60 seconds full-bleed load
6	120 seconds no-load
7	370 seconds part-bleed load (700 50°F)
8	120 seconds no-load
9	Shut down

TABLE V. TEST REQUIREMENTS FOR ESSENTIAL AND NONESSENTIAL UNITS FOR CIVIL APPLICATION

TSO Part 37 Subpart B Section 37.183 Paragraph	Subject	Applicability	
		Essential Units	Nonessential Units
4.3	Attitude Conditions	X	X
4.4	Magnetic and Electronic Interference	X	X
4.5	Operating Characteristics (see TSO for required data).	X	Not required
4.7	Negative Acceleration	X	X
5.0	Design and Construction	X	X
5.2	Air Intake (Icing Characteristics)	X	Not required
5.2.1	Foreign-Object Ingestion	X	Not required
5.2.2	Inlet Air Pressure Drop (Velocity Distribution)	X	Not required
5.3.4.4	Oil Tank Pressure Test	X	X
5.8	Drive Attachments	X	X
5.11	Safety Devices	X	X
5.13	Rotor Blade Failure Protection	(Note 2)	
	(1) Rotor containment, nonessential	Optional	X
	(2) Blade portions likely to occur	X	X
	(3) Blade containment (critical stage)	X	X
	(4) Critical stage (see 3.23 for definition)	X	X
5.14	High-Energy Rotors		
	(1) Containment	Optional	X
	or Integrity	X	Optional
5.15	Vibration (Rotors and Highly Stressed Parts)	X	Optional
5.16	Stress Rupture and Start/Stop-Cycle Fatigue	X	Optional
5.17	Control of Unit Rotation (if included--Manufacturer's Option)	Optional	Optional
6.0	Block Tests		
	(1) Unit calibration	X	X
	(2) Rotor Integrity	X	Optional
	(3) Rotor containment	Optional	X
	and Rotor blade containment	X	X

NOTES:

(1) This table includes all paragraphs of the TSO and the Flight Standard where testing, analysis or documentation is specified or implied. The design requirements of all other paragraphs are satisfied by inspection or review of manufacturing drawings.

(2) The manufacturer may elect to demonstrate rotor integrity in lieu of full rotor containment on nonessential units.

(3) The FAA may specify other special tests if deemed necessary.

TABLE VI. HISTORICAL SUMMARY OF AIRESEARCH APU INSTALLATIONS

COMMERCIAL AIRBORNE APU INSTALLATIONS

APU MODEL	INSTALLATION
GTP30-51A -70	UTA DC-7C
GTP30-54	Fairchild F27
	Vickers Viscount
	Lockheed Constellation
	Goodyear Blimp
GTCP30-56	Texaco Jetstar
GTCP30-92	DeHavilland DH125
	Lockheed Jetstar I & II
	Dassault Falcon 20
	Rockwell Jet Commander
GTP30-95 -121 -141	Convair Dart 600 640
GTP30-100	Del Mar Helicopter
GTP30-142C	HFB 320 Hansa Jet
GTCP36-4A	Fokker F28
GTCP36-6	Grumman Gulfstream II
GTCP36-16	NAMCO YS-11
GTCP36-16A	Fiat Aeritalia G222
GTCP36-28	VFW 614
GTCP36-100[A]	Dassault Falcon 50
GTCP36-100[H]	Hawker-Siddeley HS125-700
GTCP36-100[E]	Canadair CL-600
GTCP36-100[G]	Grumman Gulfstream III
GTP70-4	Vickers Viscount
GTP70-17	Douglas DC-6B
GTC85-37	Grumman Gulfstream I
GTC85-62 -65 -71	Lockheed L-100
GTC85-90 90F	Convair 580
GTC85-90E	Lockheed Electra
GTCP85-91C -291C	Convair 580 (Stinger)
GTCP85-91S	S.E. Caravelle
GTCP85-98 -98CK	Boeing 727-100 -200
GTCP85-98CK	Boeing 707 720 (VIP)
GTCP85-98CK[A]	Lockheed L-100
GTCP85-98CK[B]	Boeing 707 (VIP)
GTCP85-98D DCK	McDonnell Douglas
	DC-9,-10,-20,-30,-50
GTCP85-99	Sud Caravelle
GTCP85-115 -115CK	BAC1-11 500
GTCP85-115H	Hawker Siddeley
	Trident 3B
GTCP85-129	Boeing 737
	Convair 990
	Boeing 707 (VIP)
GTCP85-139H	Hawker Siddeley Trident
	I & II
GTCP85-163CK	Dassault Mercure
GTCP85-185L	Lockheed L-100
GTCP85-201	Westinghouse DC-6
GTCP85-202	Garrett Convair 240
GTCP660-4	Boeing 747
TSCP700-4B	McDonnell Douglas DC-10
TSCP700-5	Airbus A-300B

MILITARY AIRBORNE APU INSTALLATION

APU MODEL	SERVICE	INSTALLATION
GTC43 44	USN	Convair XP5Y R3Y
	USN	Airborne Pod
GTP30-52	RCAF	DeHavilland Caribou Generator Set
GTCP30-59	GAF FAF	HFB Noratlas
GTP30-106	USA	Lockheed Cheyenne Helicopter
GTP30-142C	USAF	Northrop A-9
GTCP36-50	USAF	Fairchild-Republic A-10
GTCP36-55[C]	USA	Hughes YAH-64
GTCP36-100[C]	USCG	Dassault HU-25A (Falcon 20)
CTCP36-100[K]	BAF	DeHavilland Buffalo
GTC36-200	USAF	McDonnell Douglas Northrop
		F-18
GTP70-6A -9	USAF	Fairchild C118
GTP70-10	USN	Martin P5M
GTP70-15	USAF	Boeing KC-135
GTU85-3	USAF	Douglas C-133
GTC85-56 -71	USN USAF	Lockheed C-130
GTCP85-97C	USAF	Boeing VC-135
GTCP85-98	USAF	Boeing VC-137 (A.F. One)
GTCP85-98D	USAF	McDonnell Douglas C9A B
GTCP85-98J	JAF	NAMCO C1
GTCP85-100	FAF	Dassault Atlantique
GTCP85-106	USAF	Lockheed C-141
GTCP85-116	USN	Grumman C-2A
GTCP85-127A	USAF	Fairchild AC119
GTCP85-131J	JAF	Shin Meiwa PS-1
GTCP85-134	USN	Grumman TC-4C
GTCP85-160	GAF	VFW Transall 160
GTCP85-180L	USAF	Lockheed C130H
GTCP95-2 -3	USN	Lockheed P3C
GTCP95-4	USAF	Lockheed EC121
GTCP100-1	USN	Martin P6M
GTCP165-1	USAF	Lockheed C5A
	USAF	Boeing E-3A AWACS
GTCP165-7	USAF	Rockwell B-1
GTCP660-4	USAF	Boeing E-4A AANCP
JFS100-4	SWAF	SAAB Viggen J-37
JFS100-13A	USAF	Vought A7D
JFS100-34	USN	McDonnell Douglas A4M
JFS190 SPS	USAF	McDonnell Douglas F-15

OTHER APU INSTALLATION

APU MODEL	SERVICE	INSTALLATION
GTP30-1	COMML	Generator Set
GTP30-11	COMML	UAL Generator Set
GTCP30-17	COMML	RTA Generator Set
GTCP30-18	USAF	Generator Set
GTCP30-24	USAF	Generator Set
GTP30-47	USA	PRIL Generator Set
GTP30-50	USN USAF	Wheelbarrow APU
GTP30-51	USAF	Generator Set
GTCP30-65	USA	APU
GTP30-66	USAF	Generator Set
GTP30-67	USAF	EMU 12 E Generator Set
GTP30-67B	COMML	Shell EMU29 Generator Set
GTCP30-71	COMML	Swedish Wheelbarrow APU
GTP30-96	COMML	Wolverine APU
GTP30-97	COMML	Toyota APU
GTP30-103	USA	PRIL Generator Set
GTP30-150P	COMML	Williams & Lane Gen. Set
GTP70-19	USA	30-KW Generator Set
PTP70-26	USAF	Fuel Pumper
GTP70-15	USA	60-KW Generator Set
GTP70-50	USA	40-KW Generator Set
GTP70-52	USAF	60-KW Generator Set
GTCP85-70	USAF	MAIA Ground Cart
GTCP85-72	USN	Ground Cart & Pod
GTC85-90	COMML	Ground Cart
GTCP85-97	RCAF	Ground Cart
GTCP85-102	USA	Pershing Missile
GTCP85-103	USN	USS Tennison Hydrofoil
GTCP85-125T	USN	Gunboat
GTCP85-126	NASA	NASA Van
GTCP85-127	USA	MUST Hospital Power Pack
GTCP85-180	USAF	M32A60A Ground Cart
GTCP85-184	COMML	Airline Ground Cart
	USN	FFG-7 G.M. Frigate
	USN	PCG-511 Patrol Gunboat
IE85-184	USN	U.S.S. Osprey
	Korea	Patrol Gunboat
	USN	PCG-612 Patrol Gunboat
	USN	Aerojet "Jeff A" SES
GTCP85-197	RCAF	Ground Cart
GTCP85-291	COMML	Ground Cart
GTCP85-391	RCAF	F-5 Ground Cart
GTCP85-397	USAF	M32A60 Ground Cart
GTC100-53	USN	Aircraft Carrier CVA-66
GTC100-54	USN	Ground Cart
GTC100-56	USN	Ground Cart
GTCP105-2	USAF	Ground Cart
GTCP105-6	USAF	Ground Cart
GTP431-41	USAF	Generator Set
GTP431-41F	USN	FDO Sea Sled
IE831-200	USAF	Generator Set
ME831-800	USN	Boeing Hydrofoil PHM

TABLE VII. APU INSTALLATION COMPATIBILITY TESTS

Test Item	Type of Test	Description of Test
1	Compatibility and Calibration*	Utilizing the bare APU configuration, determine that the APU and the aircraft-furnished, shaft-driven equipment are compatible and capable of specified performance.
2	Fuel Drain Check	Conduct wet start attempt with ignition deactivated to demonstrate acceptability of fuel drainage from turbine plenum, exhaust system, and APU compartment.
3	Installation Effects on Operation	Determine installation effects on APU starting and performance.
4	Compartment Temperature Survey	Conduct a survey to determine component surface temperatures and compartment air temperatures during APU operation and following shutdown (soakback effect).
5	Electrical System Compatibility	(a) Establish that the aircraft and APU electrical system controls interact in a compatible and suitable manner. (b) Conduct electrical tests to ensure that the voltage, frequency, and fault-clearing controls are satisfactory. (c) Determine adequacy of electrical power supply to APU starter (or hydraulic power supply if hydraulic start system is used). (d) Determine adequacy of electrical voltage for actuation of relays and solenoids simultaneously with maximum voltage requirement of the APU starter.
6	Fire Detection and Extinguishing System	Conduct checkout test of the fire detection and extinguishing system.
7	Bleed-Air System Compatibility	Conduct pneumatic system tests with actual or simulated aircraft ducting to determine compatibility of the APU with the aircraft main engine pneumatic starters and/or environmental control system.
8	Exhaust Wake Temperature Survey	Determine that exhaust wake temperature do not create excessive skin and structural temperatures on adjacent parts of aircraft and that exhaust gases are not ingested into the APU inlet.
9	Service Cycle Test	Conduct a cyclic test of operable installation equipment to simulate actual service usage. This involves opening and closing inlet and exhaust doors, starting and operating the APU at a load schedule, followed by shutdown.
10	Acoustics (optional)	Determine the acoustic characteristics of the installation and/or effectiveness of the attenuation.
11	Maintainability (optional)	Demonstrate the maintenance and servicing of line replaceable units (LRUs), installation and removal of the APU, and use of AGE equipment.

*To be conducted in a standard engine test facility.

TABLE VIII. AIRESEARCH CONDUCTED COMPATIBILITY TESTS

Installation	APU	Installation	APU
A.300	TSCP700-5	Lockheed Electra	GTCP85-291E
DC-10	TSCP700-4B	Fairchild AC-119	GTP85-127A
DC-9	GTCP85-98D	Grumman TC-4C	GTCP85-134
Dassault Mercure	GTCP85-163CK	MA-1A Trailer	GTC85-70
727	GTCP85-98	MUST	GTCP85-127
747	GTCP660-4	M32A60A	GTCP85-180
Fiat G222	GTCP36-16A	TMC105-6	GTCP100-52
Gulfstream II	GTCP36-6	CVA 62	GTCP100-56
Lockheed P3A	GTCP95-2	EMU 12-E	GTP30-67
HSA Trident	GTCP85-139H	GTGE70-6	GTP70-50
Namco YS-11	GTCP36-16	EMU 14	GTP70-52
		PUPE70-1	GTP70-21

TABLE IX. APU INSTALLATION FLIGHT TESTS

Test Item	Type of Test	Description of Test
1	Inlet Temperature/Pressure Survey	Obtain APU operating and non-operating data, including the starting cycle, also ascertain ΔP between air inlet port and exhaust outlet port.
2	Compartment Temperature Survey	Determine APU compartment temperatures and temperature of various APU and critical components while operating on the ground, in flight, and following APU shutdown (soakback).
3	Exhaust Wake Temperature Survey	Determine that exhaust wake temperatures do not create excessive skin and structural temperatures on adjacent parts of the aircraft.
4	Installation Effects	Determine in-flight installation effects on starting characteristics and performance, on flight-operable and FAA "essential" APUs.
5	Overboard Drains	Check for proper function of the overboard drain systems.
6	Vibration Survey	Determine vibration characteristics.
7	Operational Data	Record various APU operational data.
8	Inlet Icing	For FAA "essential" APU, start and operate throughout the icing envelope defined in Appendix C of FAR, Part 25.
9	Acoustic Noise Survey	Determine presence of acoustic noise above 150 dB in the APU inlets and APU compartment, both in flight and on the ground with APU operating and non-operating.

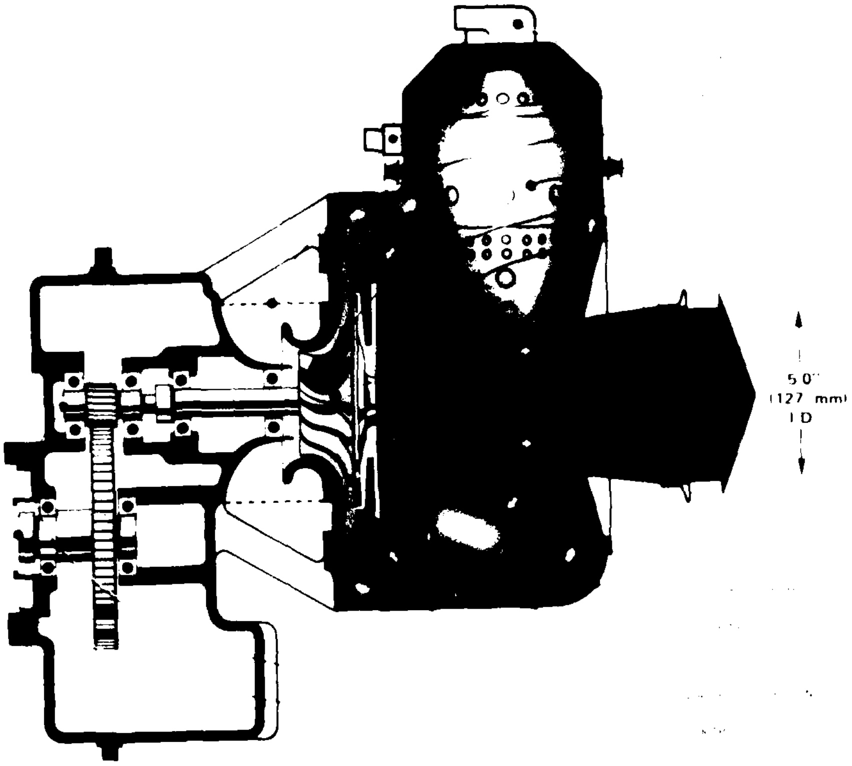


Figure 1. Model GTCP30 Cross Section.

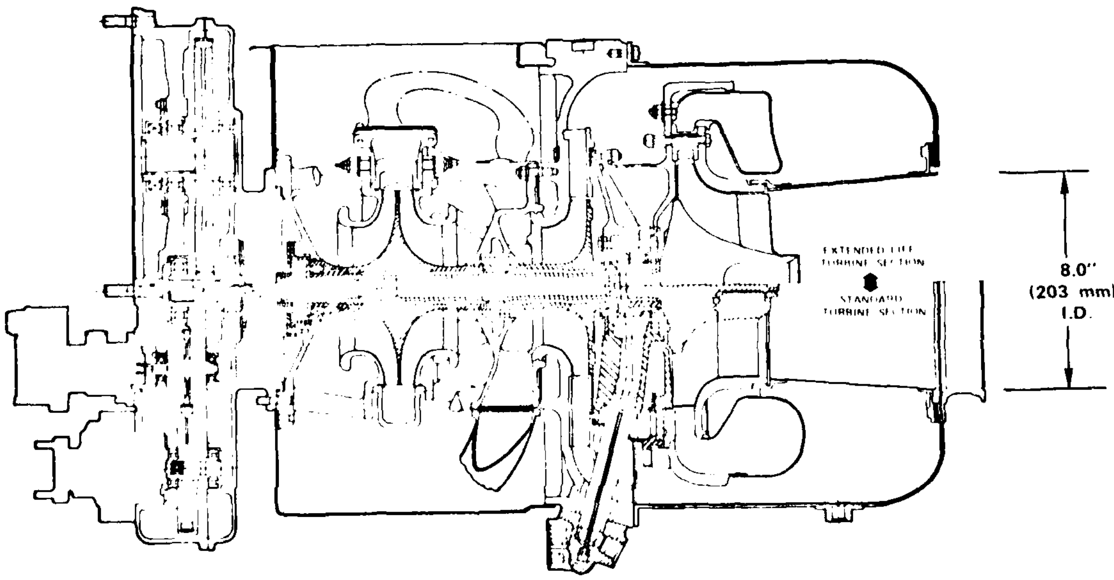


Figure 2. Model GTC85 Cross Section.

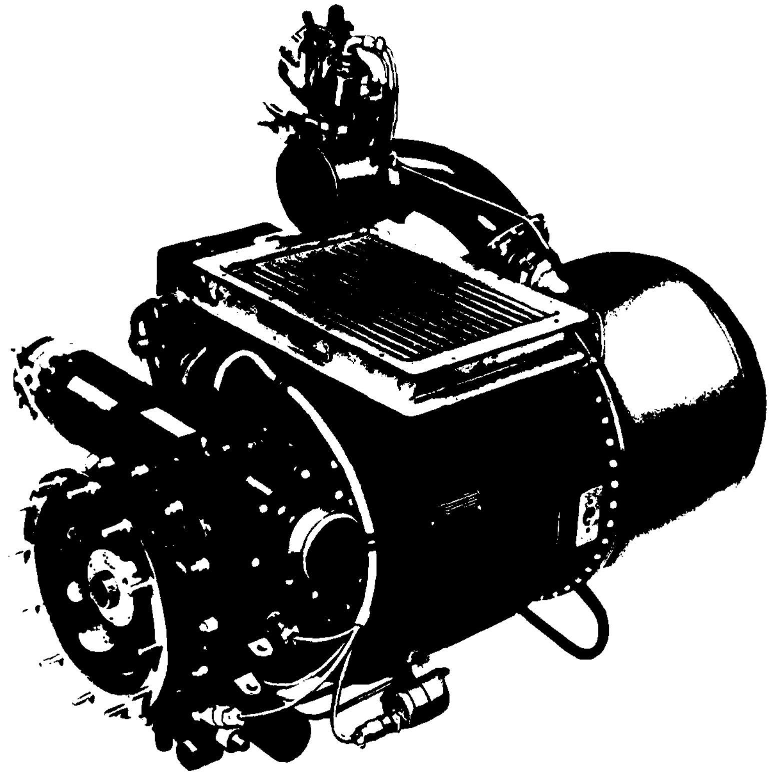


Figure 3. Model GTCP85.

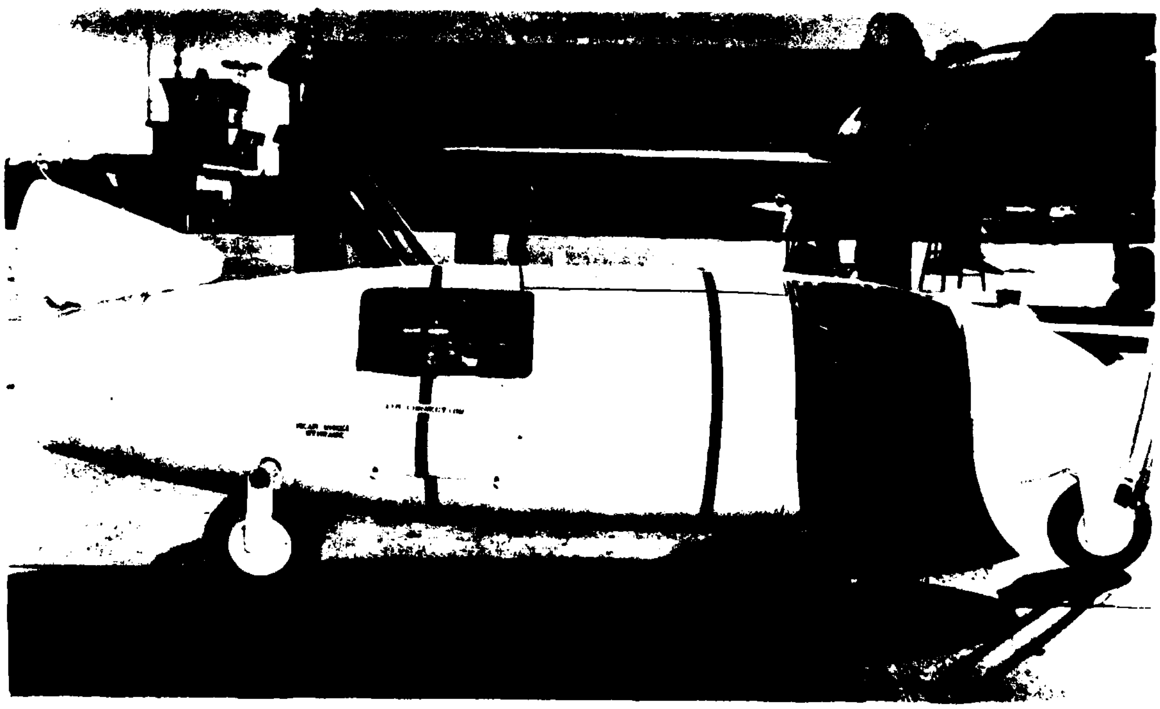


Figure 4. U. S. Navy Start POD.

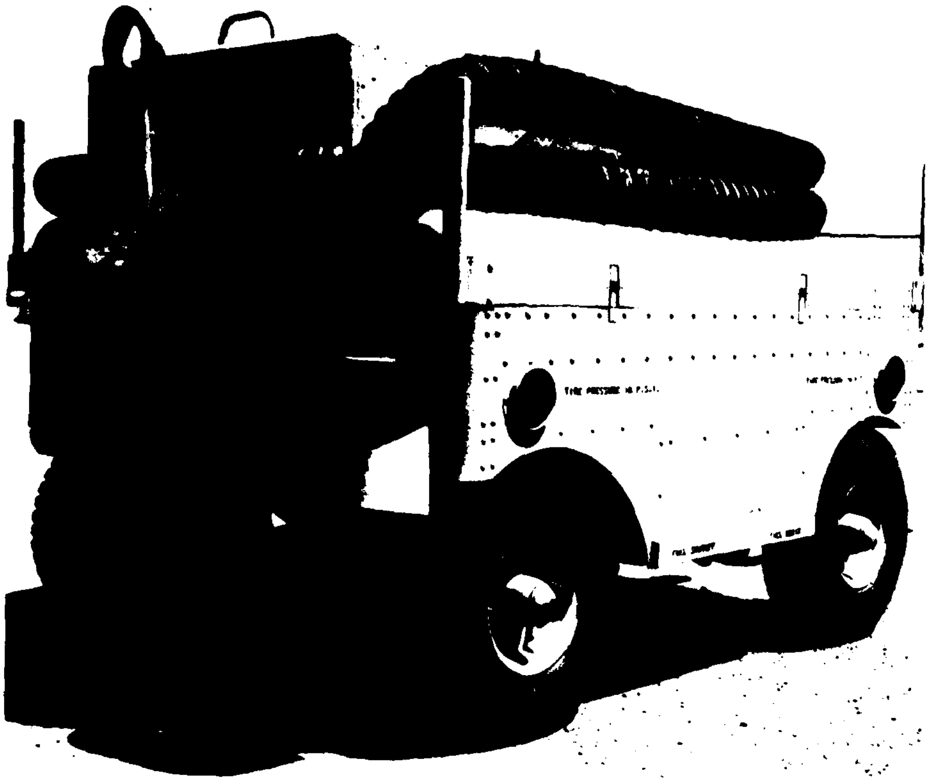


Figure 5. U. S. Air Force MA-1A Ground Cart.

Figure 6. U. S. Navy RCCP 105 Auxiliary Power Unit.

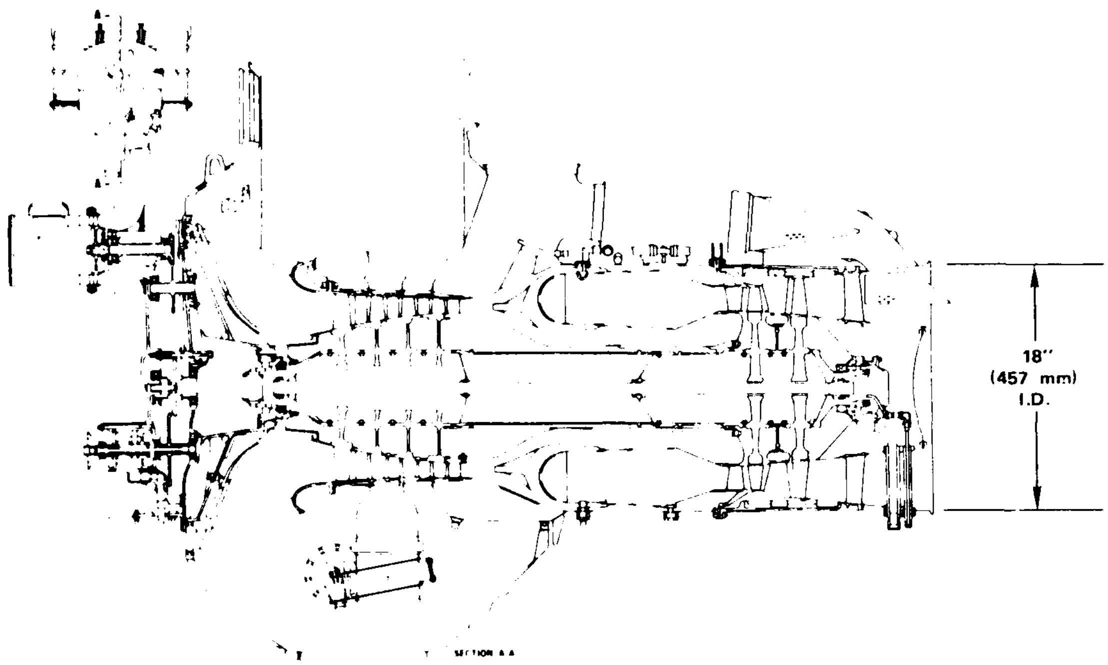


Figure 7. Model GTCP660 Cross Section.

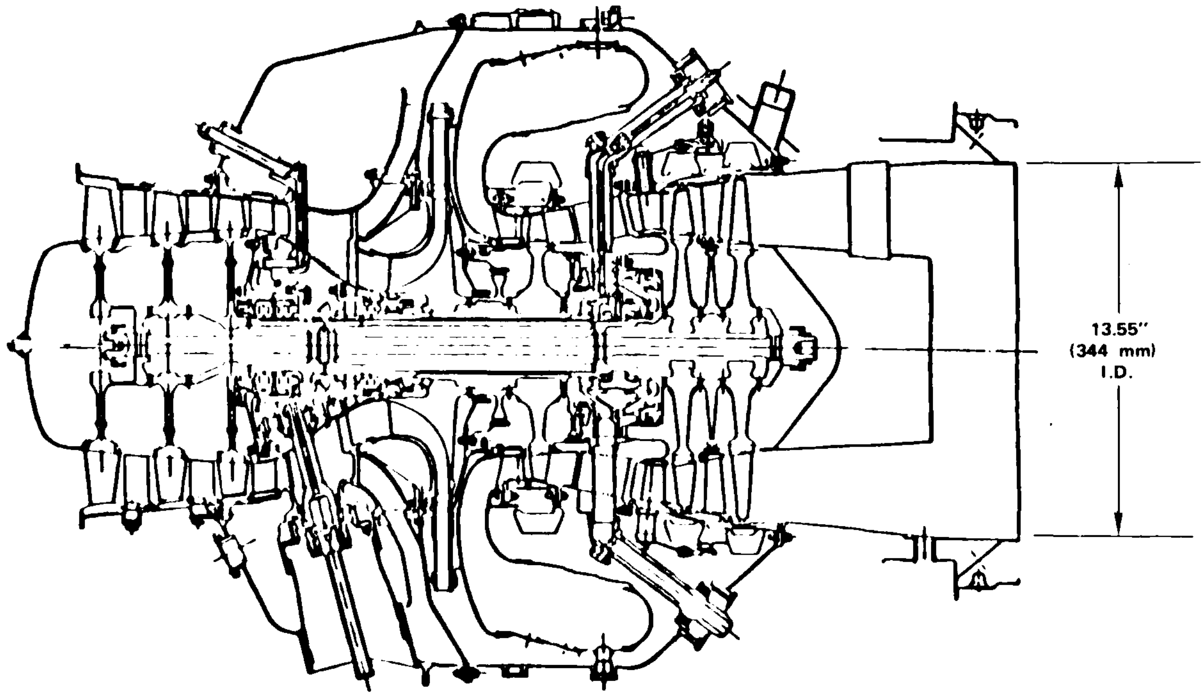


Figure 8. Model TSCP700 Cross Section.

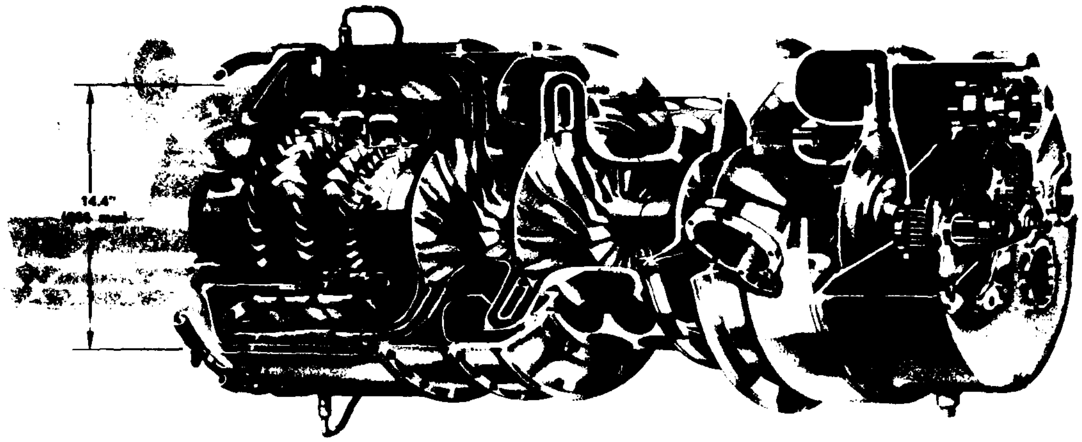


Figure 9. Model GTCP331 Cutaway.

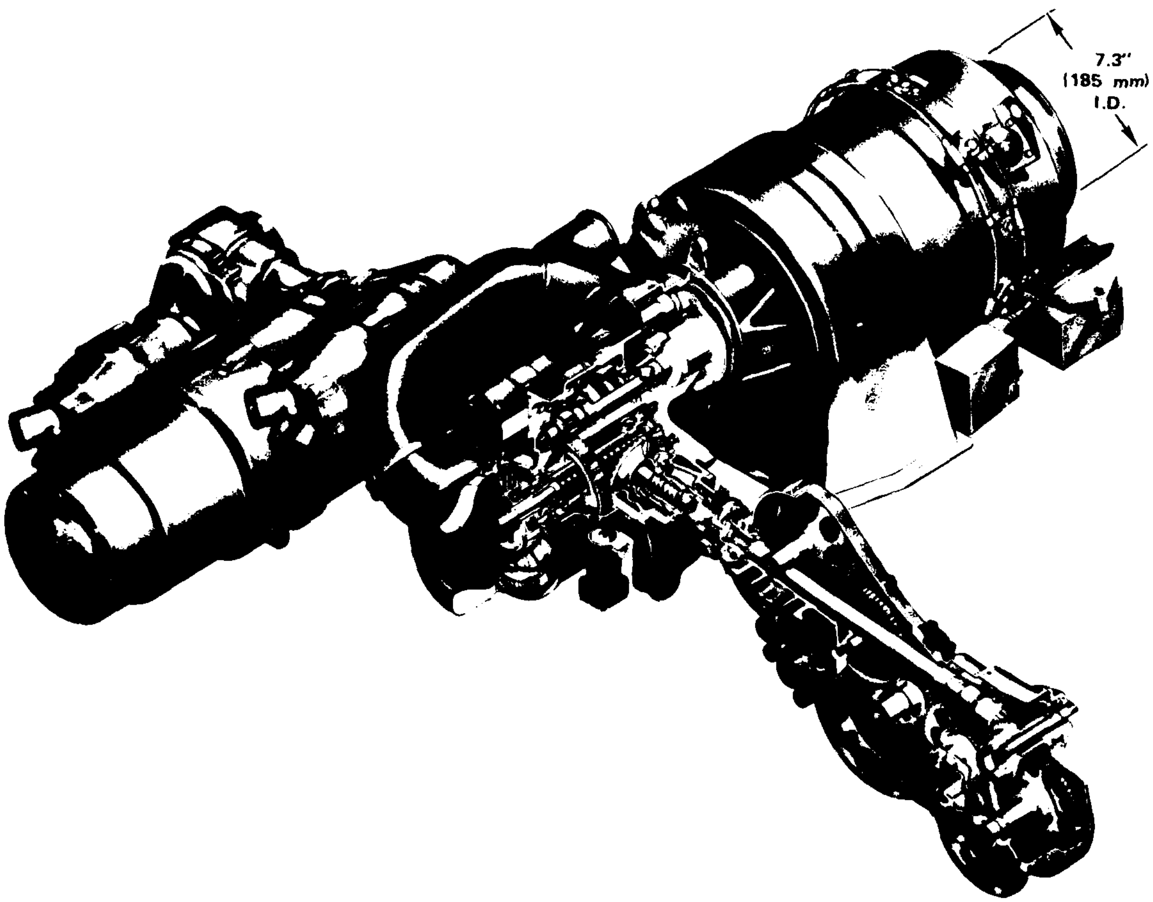


Figure 10. F-15 Accessory Drive System.

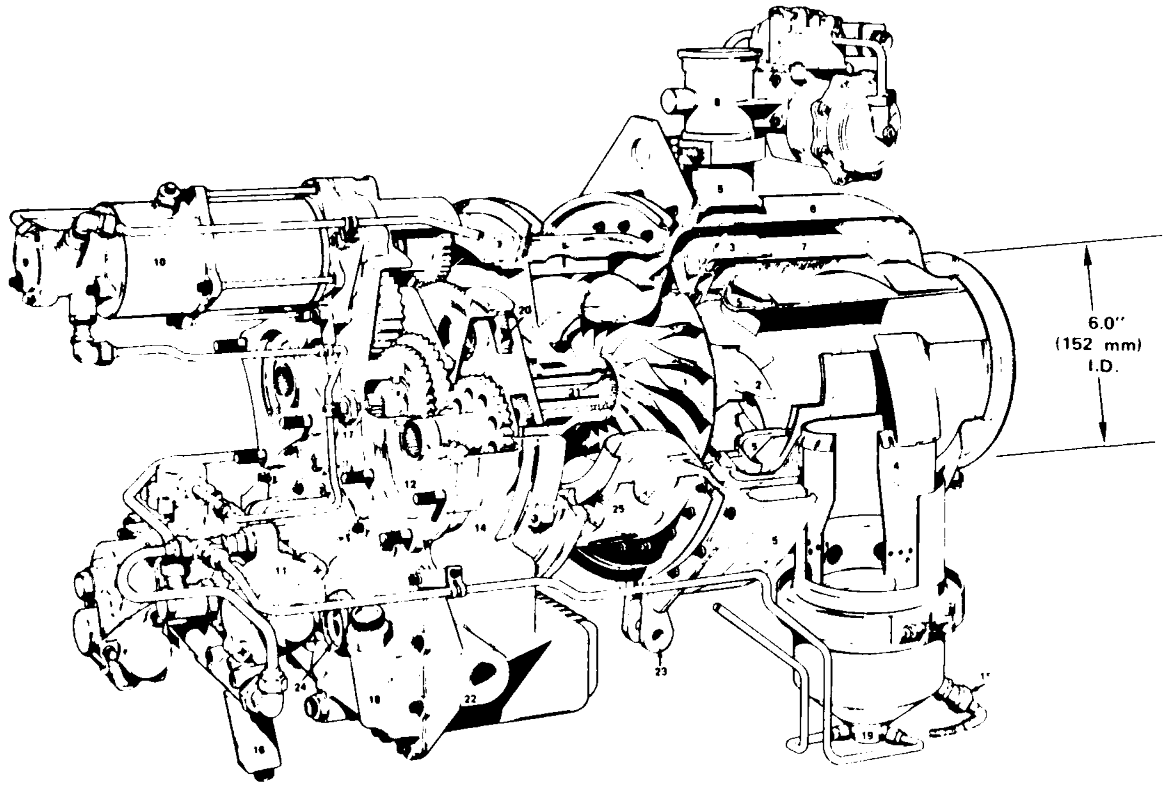


Figure 11. Model GTC36-50/100 Cutaway.

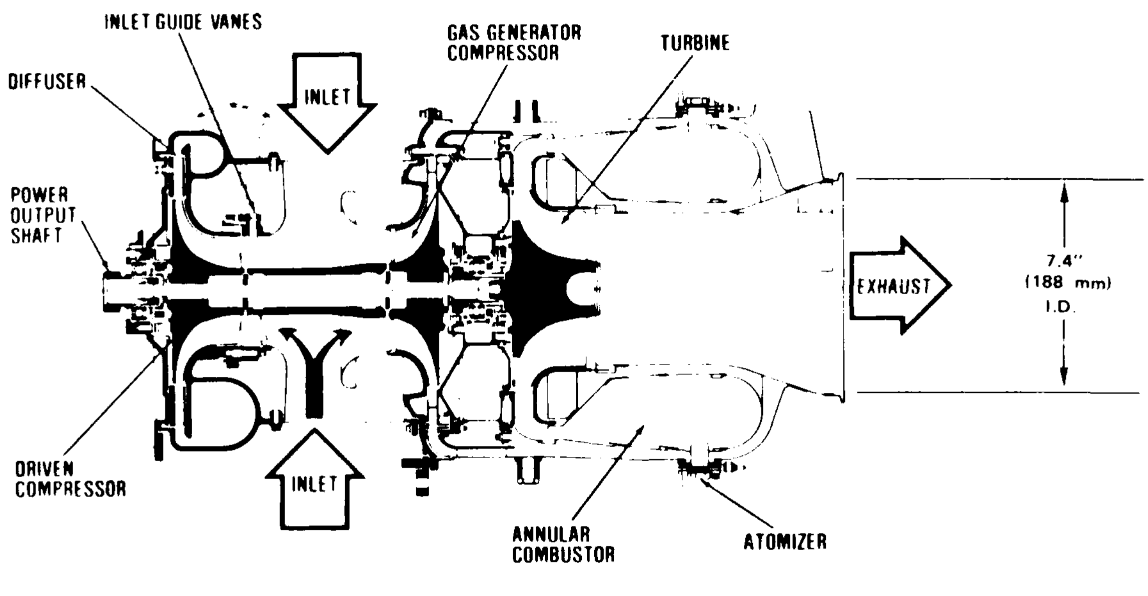


Figure 12. Model GTC3600200 Cross Section.

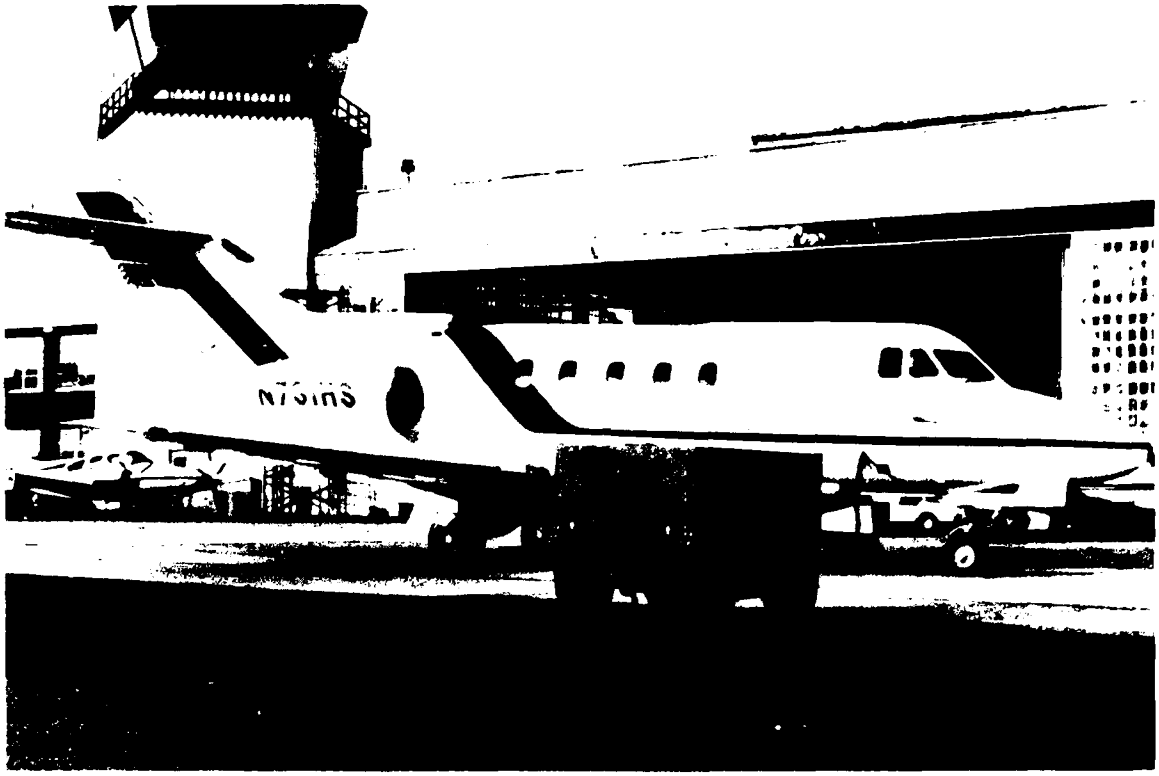


Figure 13. Advanced Ground Power Unit (AGPU).

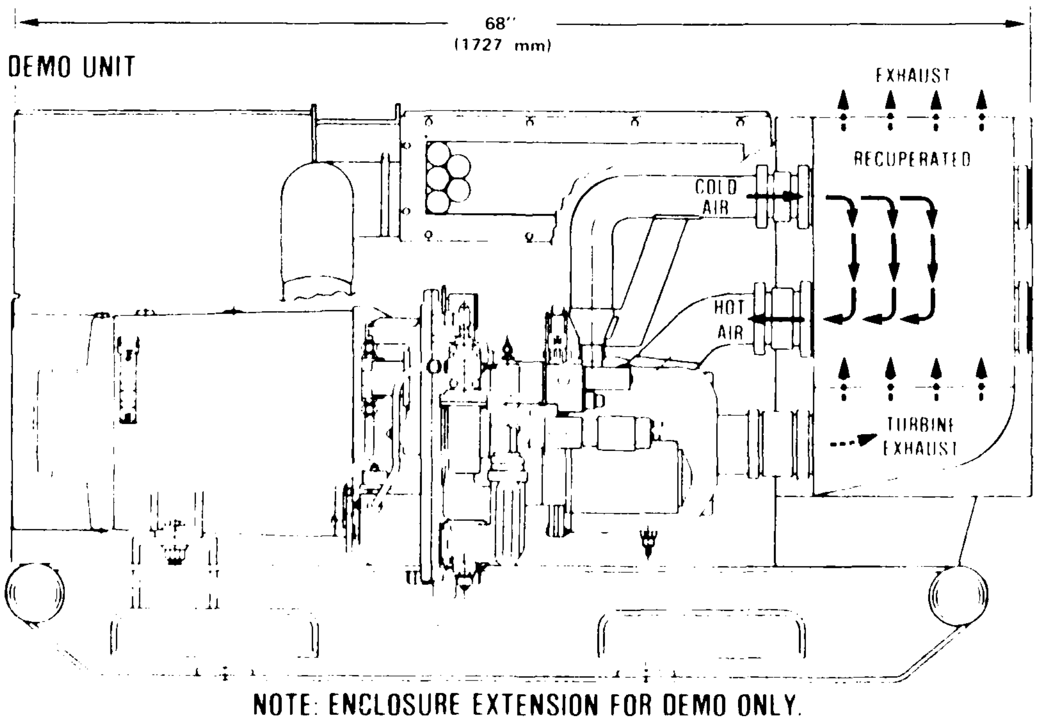


Figure 14. Recuperated 30 kW Generation.

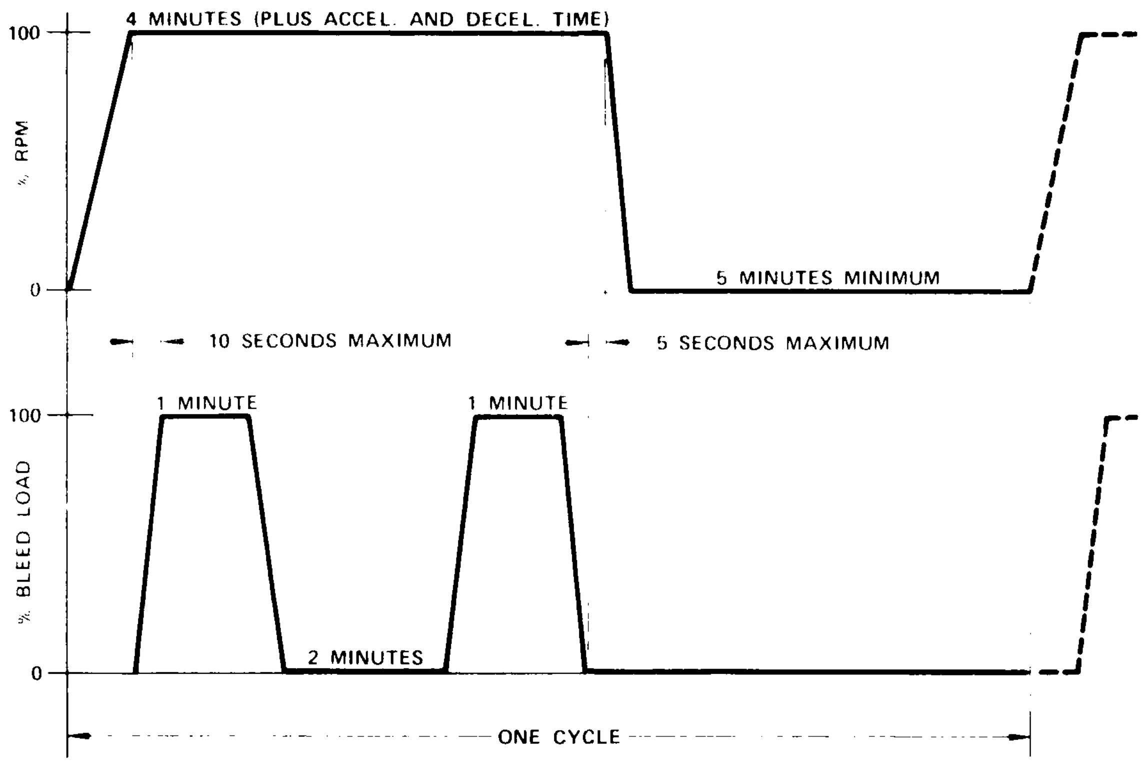


Figure 15. Start/Stop Qualification Test Cycle.

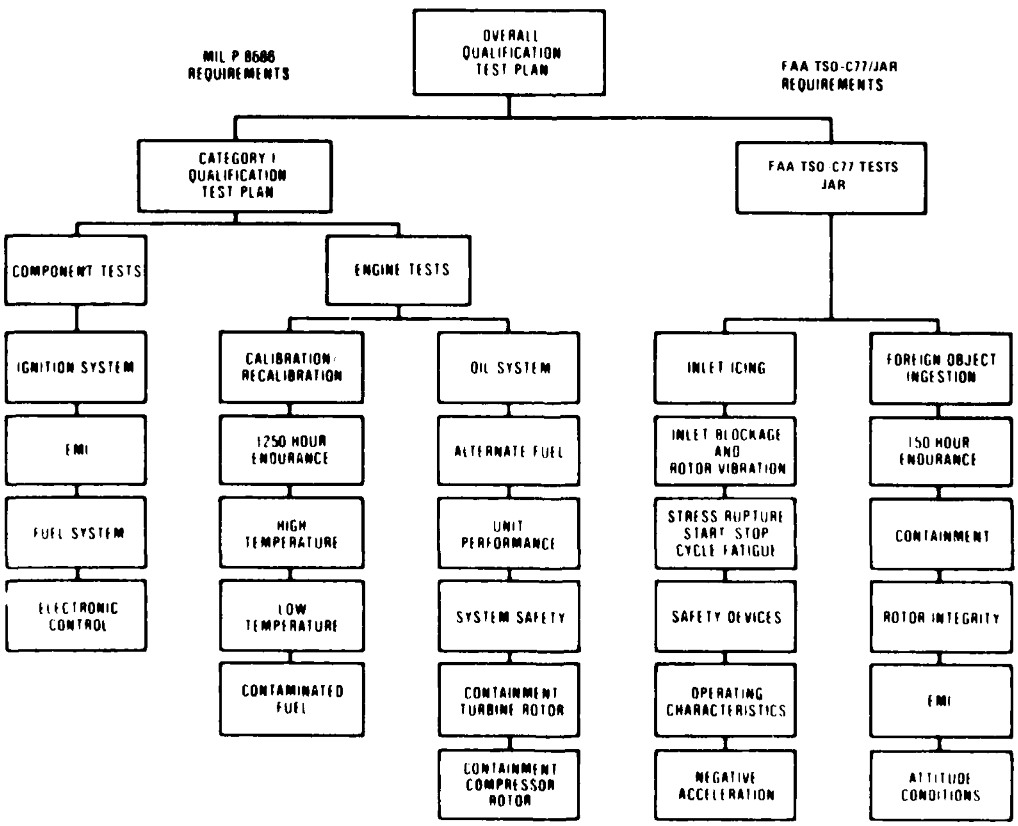


Figure 16. Models GTCP36-50 and GTCP36-100 Qualification Test.

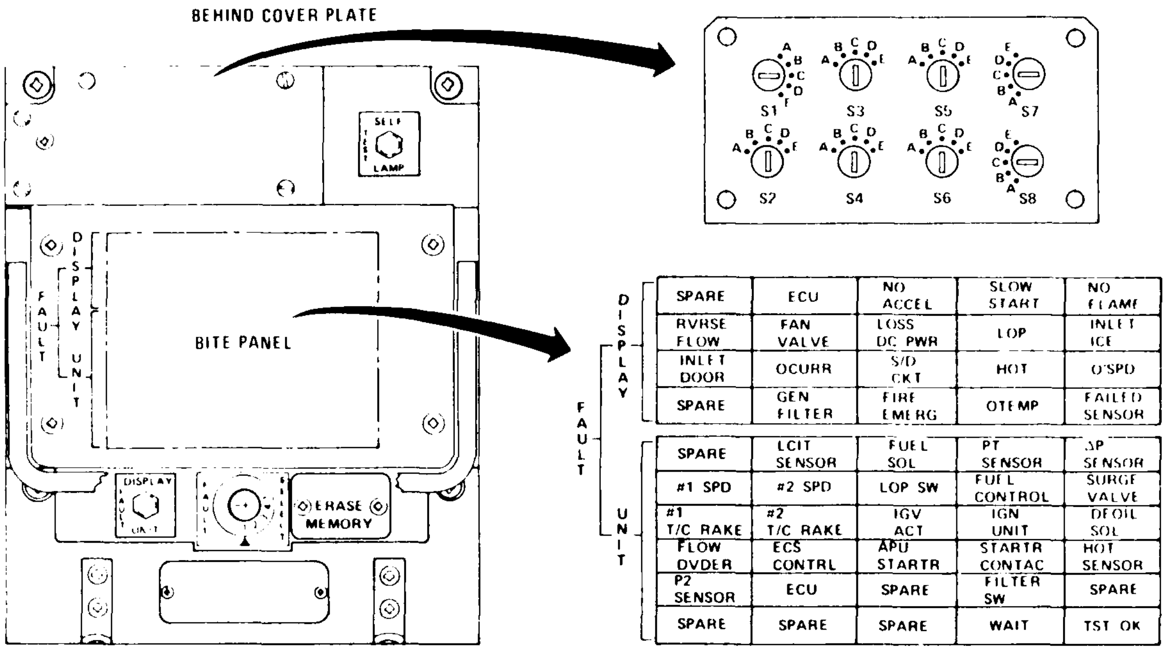


Figure 17. Fault Monitoring and Panel Indication Used With the Model GTCP331-200.

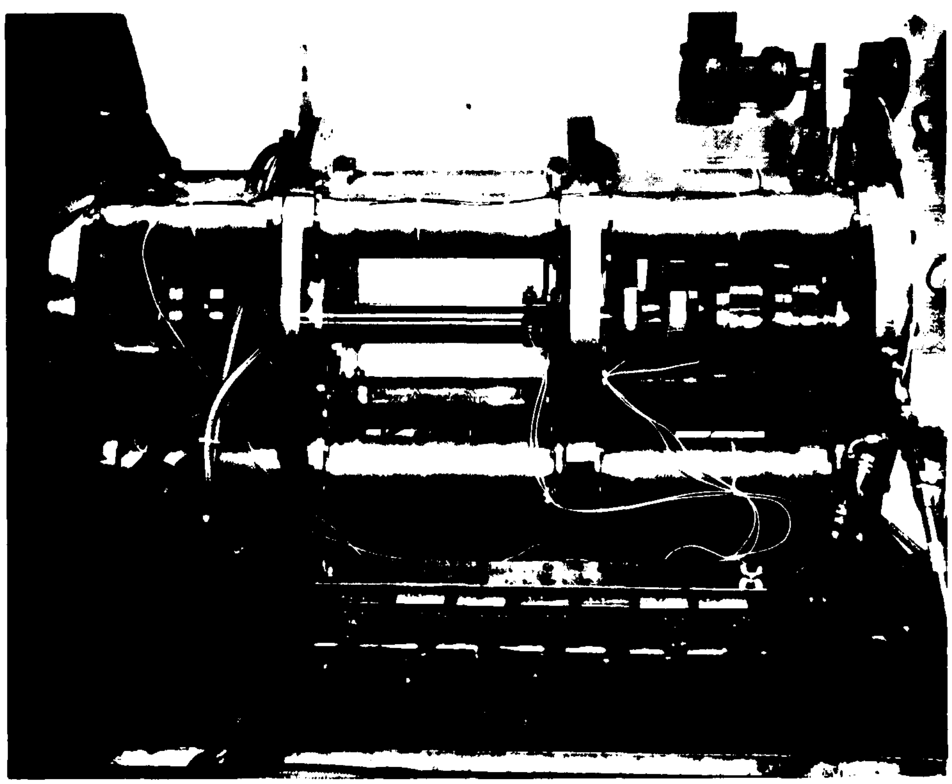


Figure 18. Rotor Dynamics Test Rig.

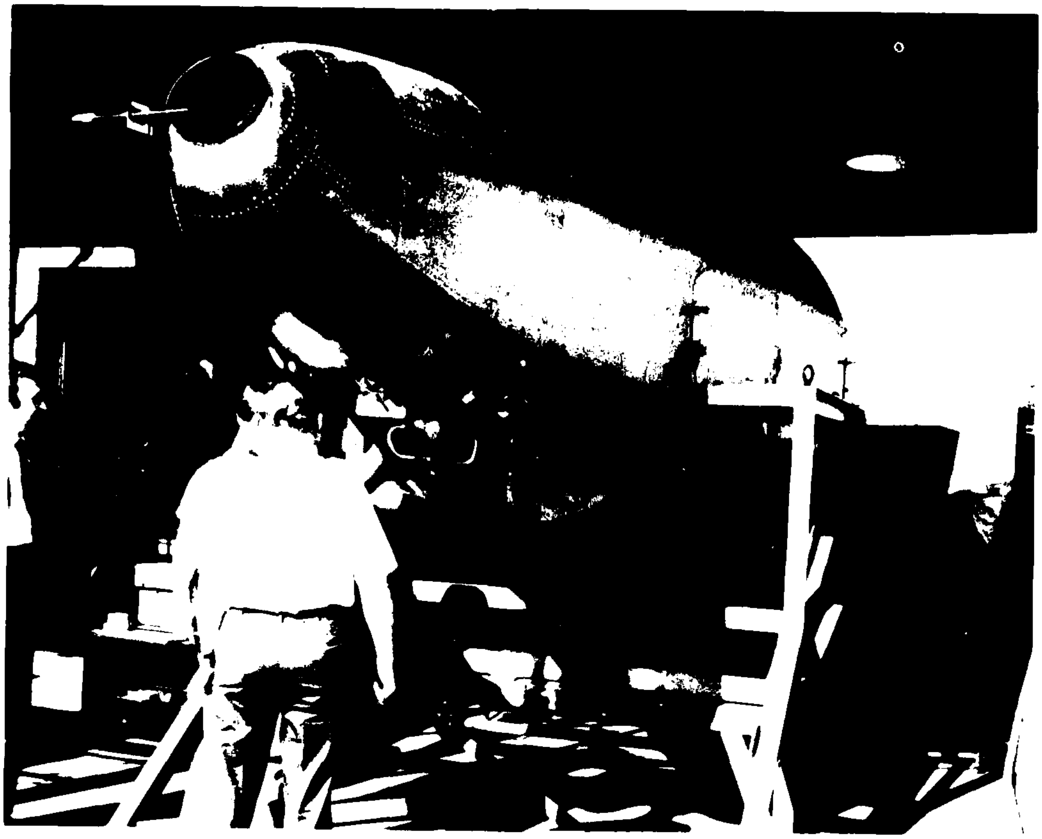


Figure 19. Mercure APU Compatibility Test.

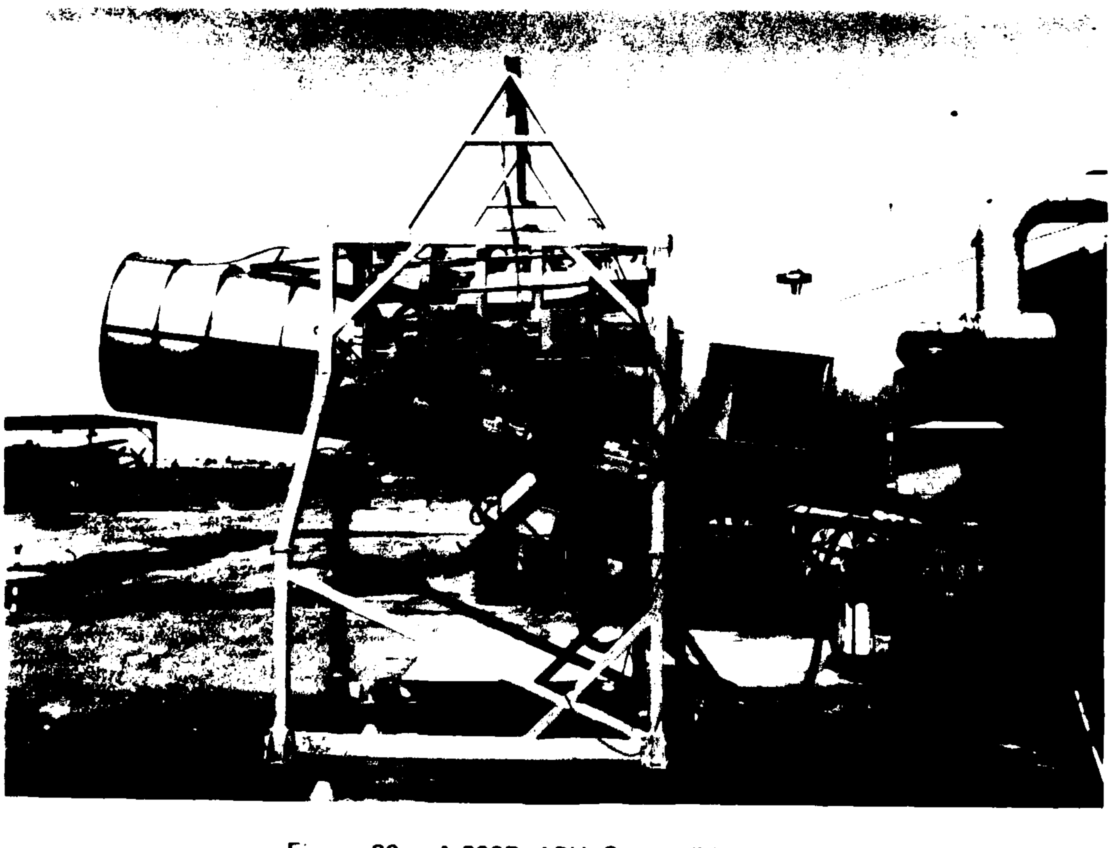
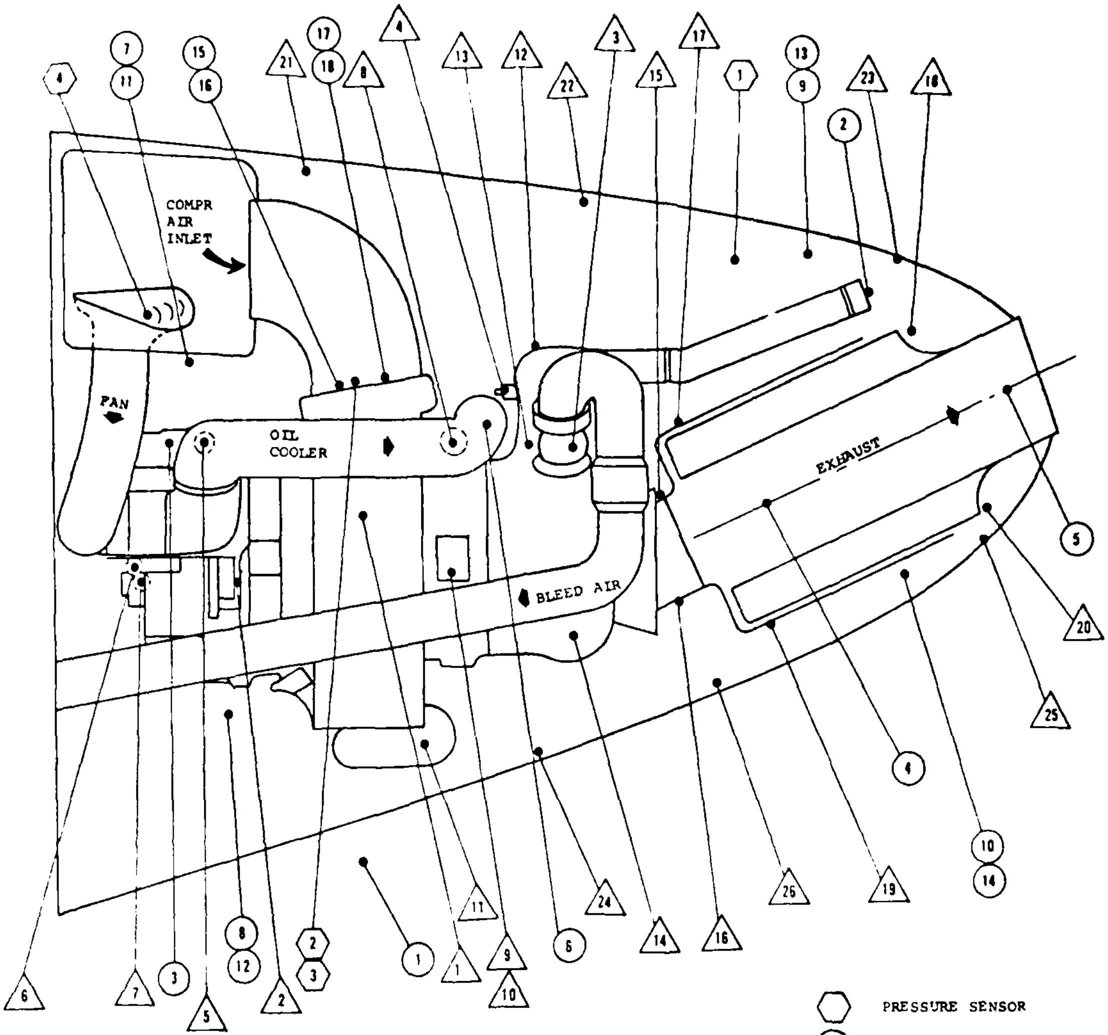


Figure 20. A.300B APU Compatibility Test.



PRESSURE SENSOR

AIR TEMPERATURE SENSOR

SKIN TEMPERATURE SENSOR

PRESSURE SENSING

1 COMPARTMENT AMBIENT
2 COMPRESSOR INLET
3 COMPRESSOR INLET
4 FAN INLET

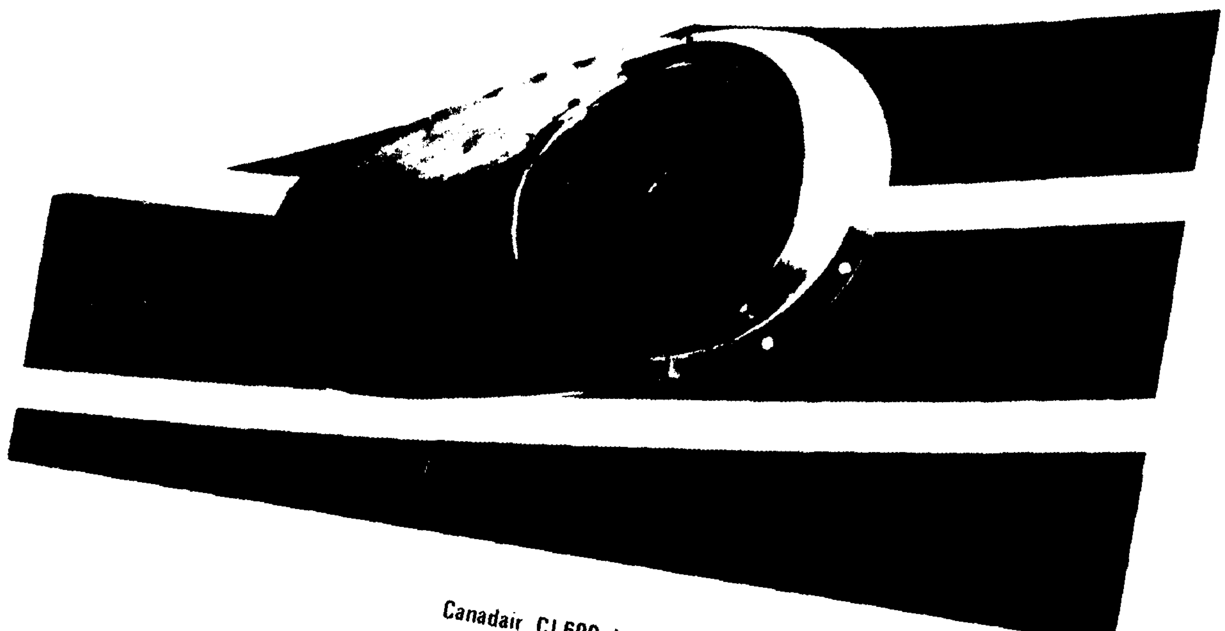
AIR TEMPERATURE SENSING

1 AMBIENT
2 GENERATOR COOLING AIR EXHAUST
3 GENERATOR COOLING AIR INLET
4 TURBINE DISCHARGE, PRIMARY
5 TURBINE DISCHARGE, MIXED
6 OIL COOLER EXHAUST
7 COMPARTMENT, TOP FORWARD, RIGHT
8 COMPARTMENT, BOTTOM FORWARD, RIGHT
9 COMPARTMENT, TOP AFT, RIGHT
10 COMPARTMENT, BOTTOM AFT, RIGHT
11 COMPART, TOP FORWARD, LEFT
12 COMPARTMENT, BOTTOM FORWARD, LEFT
13 COMPARTMENT, TOP AFT, LEFT
14 COMPARTMENT, BOTTOM FORWARD, LEFT
15 COMPRESSOR INLET
16 COMPRESSOR INLET
17 COMPRESSOR INLET
18 COMPRESSOR INLET

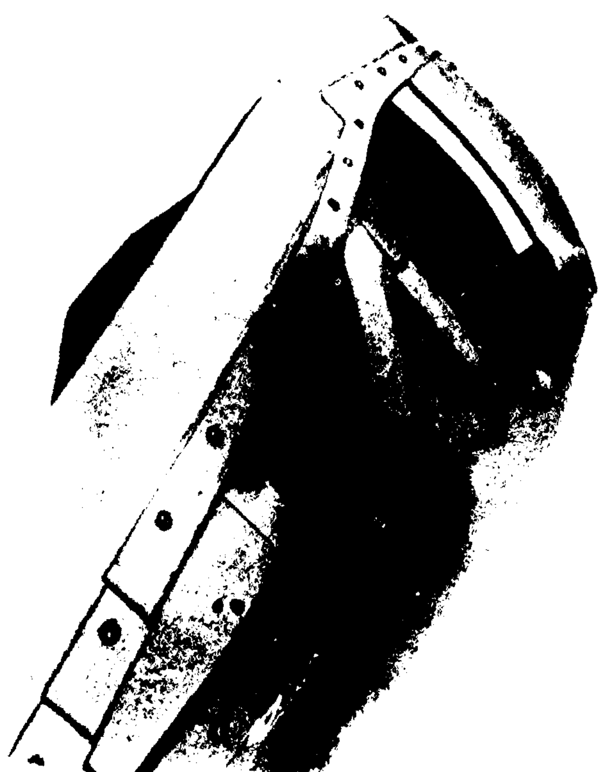
SKIN TEMPERATURE SENSING

1 COMPRESSOR PLENUM
2 GEARBOX
3 BLEED AIR VALVE
4 IGNITOR PLUG
5 GENERATOR HOUSING
6 FUEL CONTROL
7 STARTER
8 TOP MOUNT
9 LEFT MOUNT
10 RIGHT MOUNT
11 DRAIN TANK
12 COMBUSTOR CAP
13 TURBINE PLENUM, TOP
14 TURBINE PLENUM, BOTTOM
15 TURBINE DISCHARGE DUCT, TOP
16 TURBINE DISCHARGE DUCT, BOTTOM
17 TURBINE DISCHARGE MUFFLER, TOP
18 TURBINE DISCHARGE MUFFLER, TOP
19 TURBINE DISCHARGE MUFFLER, BOTTOM
20 TURBINE DISCHARGE MUFFLER, BOTTOM
21 COMPARTMENT SKIN, TOP, FORWARD
22 COMPARTMENT SKIN, TOP, MIDWAY
23 COMPARTMENT SKIN, TOP AFT
24 COMPARTMENT SKIN, BOTTOM, MIDWAY
25 COMPARTMENT SKIN, BOTTOM, AFT
26 COMPARTMENT SKIN, TURBINE DISCHARGE END

Figure 21. Example of APU Compartment Instrumentation.



Canadair CL600 Inlet



737 APU Inlet

Figure 22. Flight Test Modifications.

DISCUSSION

Mouranche, STPA, Fr

Among the problems you mentioned regarding qualifications, I did not find those concerning the air quality for cabin conditioning.

I believe that the existing regulations are very demanding and prescribe a precise analysis of a great deal of factors. What is your opinion on the severity of such regulations?

Author's Reply

The question involves the quality of the delivered air for cabin use. It is a very serious consideration. In APUs we limit the contamination of the delivered air over and above the air ingested at the inlet. Limits for contamination are specified by the aircraft manufacturer. These include the products of burning or breakdown of oil and fuel. This places particular design restraints and requirements on inlet and exit seals particularly on a compressor where the air is extracted, whether it is a load compressor or an integral compressor. Tests are conducted to determine contaminate content of the bleed air.

M.Mihail, Bureau Veritas, Fr

(1) How do you approach, for instance, the problem of the limitation of cycles in the turbine discs, given the demands on the engine during the takeoff period? Is it a matter of calculation, or of experimentation, or does the manufacturer determine a value for the users?

(2) Do you give the users exact values beyond which they should not go, or not?

Author's Reply

(1) The question asked is, how does the manufacturer determine limitations of cycles on turbine discs? Calculation or experimentation?

We do it by test and by analysis. With the application of statistical analysis, a correlation is obtained between the field experience, test lab, and the computer design criteria. In this manner whirlpit testing is being used increasingly to determine the useful life of rotors.

(2) Where required by experience, useful life limits are provided by the manufacturer. Because of the feature of containment of rotors designed into APUs, the disc useful life *may be left* "on condition", to be removed at the discretion of the operator. Usually allowable crack criteria are recommended by the manufacturer.

LES ESSAIS REACTEURS, VUS PAR UNE COMPAGNIE AERIENNE

par

F. Chétail

Chef du Service des Etudes de Propulsion
Air France, Direction du Matériel
Cidex A124, 94396 Orly Aérogare France

RESUME

La nécessité de pratiquer des essais spécifiques au banc d'essais sol sur les réacteurs, se fait sentir non seulement après révision, mais aussi lors de leur avionnage et après utilisation, avant leur mise en réparation en atelier.

Le suivi de la dégradation des performances réacteurs en cours de service apparaît comme complémentaire et très souhaitable et devient possible grâce aux équipements embarqués d'enregistrement semi-continus des paramètres sur avion (AIDS), et aux méthodes avancées d'analyses des performances telles que le GPA qui permettent, sur les derniers types de réacteurs civils, une évaluation, plus fiable que celle fournie par les méthodes classiques, des performances réacteurs au niveau de leurs modules principaux.

1. ESSAIS REACTEURS AVANT AVIONNAGE

La nécessité de pratiquer des essais spécifiques sur les réacteurs après révision et avant avionnage a été reconnue de tout temps, afin de pouvoir vérifier que ceux-ci permettent, une fois avionnés, d'assurer leur fonction de propulsion, avec des niveaux de performance et de fiabilité acceptables, c'est-à-dire satisfaisants aux normes d'acceptation en certification des avions sur lesquels ils sont montés.

1.1. Evolution historique du contenu des essais réacteurs après révision

Parallèlement à l'évolution des politiques d'entretien réacteur, le contenu de ces essais a varié de façon assez caractéristique :

- de 1959 à 1963, alors que les méthodes d'entretien consistaient essentiellement à effectuer des révisions générales à potentiels fixes, les essais au banc après révision comprenaient une liste exhaustive de vérification de performances, qui devaient satisfaire des normes indépendantes de l'âge du réacteur et de l'étendue de réparation ou révision effectuée, et qui s'alignaient étroitement sur les performances minimales (de poussée) et maximales (de température d'éjection des gaz) déterminées lors de la certification du type de réacteur considéré.

- de 1963 à 1969, la distinction fonctionnelle entre parties chaudes et parties froides des réacteurs ayant conduit à la notion de visite intermédiaire et de révision générale, après un premier temps au cours duquel la notion de type unique d'essai avait été conservée, il apparut rapidement nécessaire d'admettre, après visite intermédiaire n'ayant porté que sur l'arrière du réacteur (parties chaudes), que le niveau de performances minimales d'acceptation devait évoluer et tenir compte du vieillissement propre des parties froides. Des normes plus souples furent alors instituées pour les essais en cours de vie réacteur (entre potentiels partiels [illegible]) qui tenaient compte d'une certaine dégradation en service, essentiellement d'ailleurs en matière d'EGT ou température d'éjection des gaz, le niveau de poussée minimum certifié restant bien évidemment conforme à celui déterminé en certification avion.

de 1969 à 197[illegible], l'introduction de la notion d'entretien modulaire élargit encore l'application de ce type d'essais "allégé". Aucun réacteur n'ayant l'ensemble de ses modules révisés en même temps, l'application des normes de "[illegible] maintenance" devint alors générale.

- de 1975 à nos jours s'est fait ressentir progressivement, cependant, la nécessité de procéder, à intervalles de temps de fonctionnement déterminés, à la vérification des "performances de révision générale". Ceci non seulement en matière de température EGT, pour garantir un certain temps d'utilisation sans atteindre la limitation opérationnelle, mais encore et surtout en matière de rétablissement d'une consommation spécifique rétablie à une valeur plus proche de celle du réacteur d'origine, avec laquelle en particulier le rayon d'action et le coût direct d'exploitation de l'avion avaient été établis au départ. Il apparut alors nécessaire de restaurer périodiquement ces performances au plus près de leur niveau d'origine.

En effet, on avait noté en service, surtout pour les réacteurs à fort taux de dilution, JT9D-7 et CF6-50C par exemple, une allure caractéristique des courbes de dégradation de performances en service (fig. 1) (Ref. 1).

Après une période assez courte de détérioration rapide, correspondant très probablement à la "mise en place" des jeux de fonctionnement réglés au plus juste en révision générale de modules, prend place une détérioration progressive à taux relativement constant. Pour le JT9D-7 et le CF6-50 ces taux de détérioration sont d'environ 5° C (EGT), et de 0,5 % de consommation spécifique par 1000 h de fonctionnement, en période stabilisée.

Les crises pétrolières de 1973 et de 1979 n'ont fait qu'accentuer cette nécessité, les révisions générales (ou celles des modules principaux) apparaissant d'autant plus souhaitables que leur coût d'exécution, en valeur désactualisée se stabilise dans le cas de réacteurs ayant atteint leur maturité, alors que le coût du carburant ne fait qu'augmenter et aux deux reprises déjà citées, de façon spectaculaire.

1.2. Suivis statistiques

La disparition de la notion de révision générale à potentiel fixe, la nécessité de maîtriser la détérioration des performances des réacteurs livrés à l'avionnage fait que le suivi statistique des performances réacteurs après révision en atelier prend une importance accrue. Le suivi est effectué après tout type d'intervention, mais prend tout son sens après des interventions de même nature. Parmi celles-ci on distingue maintenant de plus en plus celles correspondant aux programmes de "restauration des performances" du type RAM (refurbishment and modernization) par exemple pour le JT9D, ou "WPG" (workscope planning guide) pour le CF6-50 par exemple.

Les figures 2 et 3 représentent pour les réacteurs CF6-50C révisés par AF pour l'ensemble des partenaires du groupe Atlas, l'évolution de ses performances globales essentielles :

- après révision mineure (fig. 2), en "main base"

- après application du programme de restauration de performance "WPG" (fig.3)

Ce dernier graphique traduit l'effort permanent d'AF pour atteindre un niveau de performances constamment amélioré afin de retrouver les performances d'origine du réacteur par rétablissement des profils et des jeux d'ailettes par exemple, ou même de les améliorer par l'introduction de modifications structurales toujours relativement importantes, telles que le recambrage de certains stators, afin d'optimiser le point de fonctionnement réacteur.

Par ailleurs, l'étude statistique des causes de rejet au banc d'essais permet de surveiller le maintien de la qualité du type d'intervention effectué.

Les figures 4 et 5 donnent un exemple de ce type de suivi pour les réacteurs essayés soit après révision partielle (cas des JT9D-7 à AF), soit après révision plus ou moins importante (cas des réacteurs CF6-50) aux différents bancs d'essais des compagnies faisant partie du groupe Atlas.

1.3. Caractéristiques actuelles des essais au banc des réacteurs

Les considérations historiques et statistiques exposées ci-dessus permettent pensons-nous d'éclairer d'un jour nouveau, les caractéristiques fondamentales recherchées actuellement au banc après révision partielle ou générale des réacteurs, qui apparaissent essentiellement comme les suivants :

- fonctionnement "fiable" du réacteur et de ses accessoires, c'est-à-dire sans fuites extérieures (air, huile, carburant) et à l'intérieur de limites d'utilisation données (EGT, niveau de vibration en particulier) ;

- fonctionnement "performant" du réacteur, c'est-à-dire permettant de projeter une utilisation du réacteur non limitée peu de temps après avionnage (limitation EGT), ou non économique (niveau de consommation spécifique par exemple).

1.4. Limitations relatives à l'interprétation des essais GPA

De façon générale, seules les performances globales du réacteur sont relevées (poussée, consommation carburant, niveau EGT). L'instrumentation actuelle et surtout les méthodes courantes d'évaluation de performances ne permettent pas d'apprécier de façon fiable le niveau de performances modulaires. (Ref. 2 et 3)

Nous citerons a titre d'exemple l'exercice de remise en état progressive effectuée en 1978 par AF sur le réacteur CF6-50C n° 455 169. A chaque étape, un seul module faisait l'objet de remise en état, le réacteur étant ensuite réassemblé et passé au banc d'essais. Le dépouillement des essais par la méthode standard faisait apparaître, par rapport à un réacteur de référence, des variations de rendement et/ou de coefficient de débit qui n'étaient pas en général toujours en rapport avec les changements introduits dans le réacteur (fig. 6). Par contre, le dépouillement de ces mêmes essais par la méthode GPA de HSD* qui tient compte de coefficients de correction variables pour chaque mesure, aboutissait à des résultats beaucoup plus cohérents (fig. 7). Aussi cette méthode est elle maintenant généralisée pour les réacteurs JT9D-7 et CF6-50 du groupe Atlas. (fig. 8)

2. ESSAIS REACTEURS A L'AVIONNAGE

2.1. Nécessité de tels essais

Les essais au banc sol, si utiles qu'ils soient, notamment en matière de réduction des travaux à l'avionnage, ne sont pas toujours exécutés au banc d'essais spécialisé. Dans la mesure où il n'est pas nécessaire de s'assurer que le réacteur après révision partielle, possède un niveau de poussée minimal, il est possible d'effectuer une partie de ces essais sur avion.

Il est en effet absolument nécessaire de s'assurer que chaque réacteur, à son avionnage, indépendamment des travaux qu'il a subi, présente un niveau de fiabilité et de performance satisfaisants.

D'autre part, il faut noter que dans certains cas les réacteurs sont essayés au sol dans des configurations d'habillage -en particulier tuyère (s) d'éjection ou avec certains accessoires différents de ceux qu'ils auront sur avion.

2.2. Caractéristique essentielle des essais réacteur à l'avionnage

Les essais à l'avionnage sont caractérisés par l'assurance qu'ils apportent quant à l'intégrité de l'interface réacteur-avion (problèmes de fuites des différents plans de raccordement carburant et air et problèmes de réglages de commande essentiellement). Ils permettent également de s'assurer de certaines fonctions non vérifiables au banc sol, telle que le déploiement des inverseurs de poussée ou le fonctionnement correct des générations hydraulique et électrique.

Certaines valeurs de fonctionnement dépendent également de l'installation et doivent être notées pour référence ultérieure ; la plus notable de ces valeurs est celle du niveau de vibration, qui pour même valeur relevée au banc sol, peut varier considérablement d'une position réacteur sur avion à une autre, en fonction de la valeur assez aléatoire de la rigidité des mats support réacteur.

2.3. Limitations de tels essais

Ces essais systématiques pour tout avionnage réacteur, représentent une charge de travail importante, même dans le cas où les pré-réglages ont été effectués au banc d'essai sol. Ils sont toutefois essentiels, ainsi que nous l'avons déjà indiqué pour s'assurer de niveaux initiaux satisfaisants en matière de fiabilité de fonctionnement et de performances.

* Gas path analysis de Hamilton Standard

Sur ce dernier point il convient de souligner le caractère tout à fait limité des verifications de performance (sans parler de la difficulté d'appliquer des methodes telles que le GPA). Certes l'avion n'est pas le banc d'essai idéal -bien que permanent- du réacteur. Toutefois, le motoriste ne peut que regretter que l'avionneur, de façon générale, n'introduise pas une instrumentation de précision suffisante pour établir à l'avionnage, et surveiller ensuite de façon efficace, le niveau des performances réacteurs, qui conditionne d'ailleurs largement celles de l'avion.

De façon générale le niveau de précision de l'instrumentation réacteur sur avion est celui nécessaire à l'affichage de la poussée, et à la surveillance de non dépassement des limites de fonctionnement en service.

A ce sujet, il peut être intéressant de comparer -pour une installation récente- les niveaux de précision de lecture des instruments principaux réacteur prévus sur A310 équipé de CF6-80 par exemple (fig. 9).

Les tableaux analogues que l'on pourrait établir pour des installations recentes (A300B, B747) ou plus anciennes (RR Avon sur Caravelle - PWA JT8D-7 sur B727) seraient encore plus éloquents.

Il en résulte que les suivis de performance avion par exemple en matière de consommation présentent un degré important d'incertitude (fig. 10).

3. ESSAIS REACTEURS AVANT REVISION ETAT EN ATELIER

3.1. Nécessité de tels essais

L'application généralisée de la politique d'entretien selon état à tous les réacteurs recents à fort taux de dilution et à performances intrinsèques élevées, tels que le JT9D-7 ou le CF6-50C, a pour corollaire la nécessité de connaître, à leur entrée en atelier, l'état, c'est-à-dire le niveau de performance de chacun des modules principaux réacteurs. En effet, nous avons vue qu'au-delà de la remise en état particulière liée à la révision principale de descente du réacteur -qui peut être liée à un incident très localisé- il s'avère nécessaire de maîtriser de façon permanente la dégradation des performances des réacteurs (fig. 1).

3.2. Limitations et conséquences

De tels essais ne sont pas toujours possibles, en particulier après certains incidents d'origine mécanique qui ne permettent pas de fonctionnement ultérieur, tels que ingestions, ruptures d'ailettes, détérioration de roulements etc.

En outre, il convient de noter qu'ils entraînent un coût et un délai de revision supplémentaire. C'est pourquoi ils ne sont généralement pas effectués.

En conséquence, le seul recours possible est l'évaluation de la dégradation des réacteurs en service à partir des éléments suivants :

. Performances au banc d'essais avant dernier avionnage

. Performances à l'installation sur avion

. Chute de performances évaluées en service continu.

3.3. Suivi des performances sur avion

Le suivi permanent de la dégradation des performances réacteurs sur avion apparaît donc comme une nécessité essentielle, si l'on veut appliquer de la façon la plus judicieuse et la plus economique possible une politique d'entretien selon etat.

- Certes certains points de repere peuvent être disponibles, que l'on ait procédé à des vérifications de performances au point fixe sol de façon planifiée (toutes les 1000 h de fonctionnement par exemple sur turbo propulseur DART) ou sur plainte équipage pour limite opérationnelle atteinte, ou anomalie de fonctionnement (cas de tous les réacteurs de façon générale). Dans ces cas de vérification au point fixe, le relevé des paramètres principaux à des points d'affichage de poussée bien déterminée, permet une comparaison avec les valeurs relevées lors de l'avionnage et donc de déterminer dans une faible mesure, si le réacteur serait encore ou non dans les limites d'acceptation au banc sol et éventuellement de retoucher ses réglages.

[illegible] de poussée, la précision insuffisante de l'instrumentation [illegible] font que ces résultats restent encore plus qualitatifs que quantitatifs. En outre, on ne peut en général reproduire au [illegible] de fonctionnement réel du réacteur au cours du vol.

[illegible] suivi permanent du fonctionnement des réacteurs en opérations [illegible] de façon de plus en plus directe. D'abord orienté vers la détection à plus ou moins court terme des détériorations progressives [illegible] principales du passage des gaz (ailettes de turbine de compresseur, [illegible] turbine etc.), ce suivi prend de plus en plus d'importance compte tenu de la valeur de référence qu'il représente [illegible] paramètres principaux du réacteur : vitesses des [illegible], température d'éjection et débit carburant, rapport de détente [illegible].

[illegible] suivi est effectué de façon assez générale à partir des relevés effectués manuellement au cockpit, en régime de croisière stabilisée, par l'officier mécanicien navigant. La figure 11 illustre, à titre d'exemple, [illegible] traitement [illegible] données pour le réacteur JT9D-7 des B747.

Ainsi que nous l'avons déjà souligné, ces données souffrent d'une grande imprécision. Si elles permettent bien en général de détecter les détériorations progressives, à développement progressif du passage des gaz, elles [illegible] actuellement insuffisantes pour déterminer de façon précise et fiable les détériorations spécifiques des modules principaux des réacteurs.

[illegible].4. Evolution prochaine de ce type de suivi permanent

Compte tenu des objectifs fixés plus haut quant à la connaissance des rendements modulaires à l'entrée d'un réacteur en atelier, des limitations en nombre des paramètres lus, et de leur relative imprécision de lecture, il apparaît maintenant comme essentiel de pouvoir disposer en permanence de valeurs plus précises, non seulement en mode de fonctionnement stabilisé (croisière) mais également lorsque la pleine puissance est requise (décollage).

Il convient de noter que ces renseignements ne sont pas nécessaires à la conduite du vol, ni à la surveillance à court terme des dégradations rapide des paramètres principaux réacteurs, et qu'en conséquence la connaissance ou même l'affichage et le recueil de ces valeurs n'est pas nécessaire au niveau du cockpit.

Au contraire l'acquisition et le stockage de ces paramètres est maintenant envisagée par AIDS. La *figure 12 représente le schéma d'une* telle installation prévue par AF pour les A310, équipés de réacteurs CF6-80.

Les paramètres nécessaires à l'évaluation permanente des rendements modulaires seront relevés par l'intermédiaire de capteurs. Puis cette information, traduite électroniquement en langage approprié au niveau du réacteur, sera [illegible] stockée dans un enregistreur de maintenance. La sortie en clair de ces paramètres sera possible au cockpit sur imprimante, à la demande de l'officier mécanicien navigant, pour transmissions ultérieures au sol et traitement centralisé sur ordinateur de grande capacité à la base principale de maintenance.

Ainsi il deviendra possible, à toute entrée réacteur en atelier, suite à incident ou non, de pouvoir connaître l'état de détermination de performances de chacun des modules et donc d'orienter la définition des travaux de remise en état.

Toutefois, il importe de souligner dès maintenant que cette détermination ne pourra être possible et rentable qu'à deux conditions :

a) la représentativité de chacun des paramètres mesurés devra être assurée. Dans les conditions de mesures retenues (croisière stabilisée, décollage), [illegible] température mesurées localement devront être [illegible] de la moyenne thermodynamique servant ultérieurement aux calculs de rendement ou de coefficients de débits modulaires.

b) une méthode fiable de calcul de ces valeurs moyennes thermodynamiques de rendement et de coefficient de débits modulaires, telles que le GPA, devra être disponible dès mise en service de ce type de réacteur.

[illegible] ne saurait trop insister pour que ces méthodes soient développées en priorité par les constructeurs de réacteurs, de façon à être offertes en même temps que [illegible] de ceux-ci.

En effet les utilisateurs n'ont ni la vocation ni la plupart du temps les moyens de développer eux seuls de telles méthodes d'analyse, qui supposent en particulier la connaissance de certaines caractéristiques fondamentales des réacteurs, connues des seuls constructeurs.

4. CONCLUSION

En résumé, il apparaît que les essais réacteurs, d'abord uniquement effectués au sol après révision générale pour s'assurer de la conformité des performances minimales des réacteurs avec les normes de certification, s'averent de plus en plus nécessaires à l'entrée des réacteurs en atelier apres avionnage, pour organiser de la façon la plus efficace et économique possible la remise en état de leurs modules les plus défaillants. Ne pouvant en général être réalisés à ce stade, notamment apres incident mécanique, cette évaluation est maintenant envisagée de façon permanente en vol, grâce aux AIDS, avec toute la complexité d'acquisition et de traitement que suppose une connaissance fiable des rendements et coefficients de débits modulaires.

Références

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2. URBAN L.A. — Parameter selection for multiple fault diagnostics of gas turbine engines ASME Paper n° 74-GT-62
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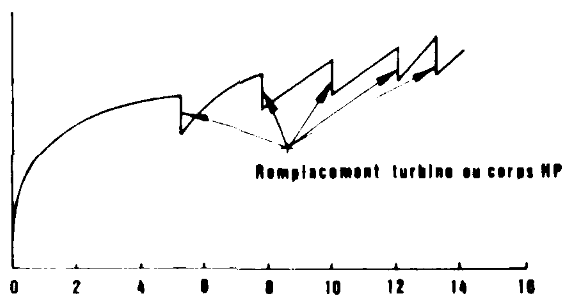
FIG. 1

REACTEURS A FORT TAUX DE DILUTION

ALLURE DE LA DETERIORATION PROGRESSIVE DES PERFORMANCES

%
Δ EGT
ou
%
Δ TSFC

(Augmentation relative par rapport aux valeurs d'origine)



(Heures de fonctionnement $X 10^3$)

FIG.2

ATLAS CF6.50C

RESULTATS STATISTIQUES MOYENS SEMESTRIELS

(reacteurs/ayant subi une intervention mineure)

Marges EGT

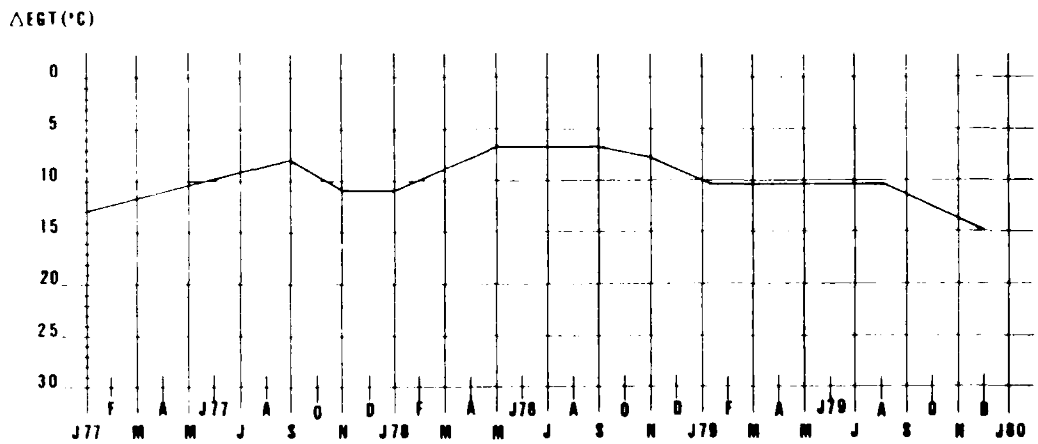


FIG.3

ATLAS CF6.50C

RESULTATS STATISTIQUES MOYENS SEMESTRIELS

(reacteurs/ayant subi le programme WPG)

Marges EGT et consommation specifique

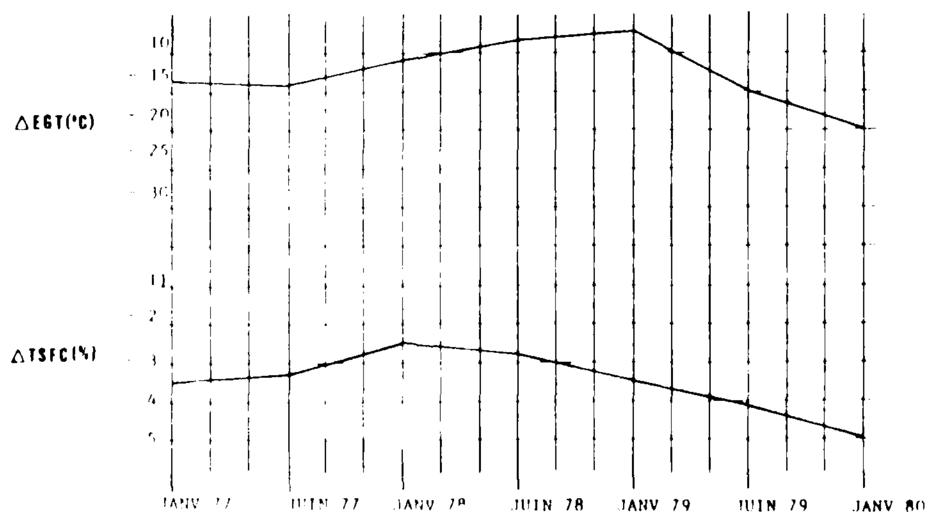


FIG. 4

AIR FRANCE : REACTEURS PWA JT9D.7

RESULTATS D'ESSAIS AU BANC DE ROISSY/CDG

Reacteurs après révision partielle

DESIGNATION \ ANNEES	1977 Qté	1977 %	1978 Qté	1978 %	1979 Qté	1979 %
Nombre total d'essais effectués	49	100 %	54	100 %	76	100 %
Nombre total d'essais avec moteur refusé	13	26,5%	14	25.9%	17	22.4%
Cause de refus : - EGT élevée H.T.	5	10.2%	4	7.4%	4	5.27%
- Temps palier 3 élevée H.T.	3	6.1%	2	3.7%	-	-
- T 6 - H.T.	2	4.08%	-	-	1	1.32%
- Consommation huile - H.T.	1	2.04%	1	1.85%	1	1.32%
- Limaille	1	2.04%	-	-	-	-
- Pt3/Pt2 et Ps4/Pt3 - H.T.	1	2.04%	-	-	-	-
- Vibrations	-	-	6	11.10%	7	9.21%
- Fuites d'huile externes	-	-	1	1.85%	3	3.96%
- Refroidissement turbine	-	-	-	-	1	1.32%

FIG. 5

ATLAS : REACTEURS GE CF6.50C

RESULTATS D'ESSAIS EN AGENCE CENTRALE (CA) ET EN BASES/PRINCIPALES (MB)

REACTEURS EN ESSAIS A :	NOMBRE D'ESSAIS EN 1979	NOMBRE DE REFUS AU BANC	MOTIF DE REFUS AU BANC D'ESSAIS																			
			EGT ELEVEE					VIBRATIONS N1					VIBRATIONS N2					LIMAILLE				
			NBRE PAR TRIMESTRE				TOTAL / TAUX	NBRE PAR TRIMESTRE				TOTAL / TAUX	NBRE PAR TRIMESTRE				TOTAL / TAUX	NBRE PAR TRIMESTRE				TOTAL / TAUX
			1 Q	2 Q	3 Q	4 Q		1 Q	2 Q	3 Q	4 Q		1 Q	2 Q	3 Q	4 Q		1 Q	2 Q	3 Q	4 Q	
AF (CA)	101	18 / 18%	6	-	1	-	7 / 7%	1	-	3	2	6 / 6%	1	-	1	1	3 / 3%	1	1			2 / 2%
AF (MB)	63	11 / 18%	2	2	3	2	9 / 15%								2		2 / 3%					
AZ (MB)	30	3 / 10%	1	-	1	-	2 / 7%							1			1 / 3%					
IB (MB)	13	5 / 38%	2	2	1	-	5 / 38%															
LH (MB)	84	12 / 14%	5	1	4	0	10 / 12%						1	-	1	-	2 / 2%					
SN (MB)	7	0 / 0%																				

FIG.6

CF6-50C 455169

EVALUATION OF MODULE PERFORMANCE ANALYSIS PROGRAM

TEST CASE: K

ACTUAL HARDWARE CHANGES: HPC refurbishment

HSD	GE	HSD gas path analysis (GPA)	GE Engine performance analysis (EPA) BASED ON TSx	GE Engine performance analysis (EPA) BASED ON EGT
Fan				
% Δ flow		0,4		
% Δ η		0,3		
LPC				
% Δ flow	ΔLPS	- 0,7		
% Δ η		- 0,7		
LPT			- 0,6	- 1,5
% Δ η				
% Δ A54		- 0,2		
LPC OLS		0,5		
HPC				
% Δ flow		0,0		
% Δ η	Δ ETAC	0,0 0,9	- 0,4	- 0,4
HPT				
% Δ η	Δ ETAT	- 0,8	- 1,4	- 2,5
% Δ A4		0,0		
HPC OLS	Δ Parasitics		- 0,7	- 0,9
			0,7	2,5

DELTAS BETWEEN STACKED AND MEASURED

$\Delta F\eta = 0,5\%$

Δ EGT = + 5°C

COMMENTS GE : Large change in HPC but also in HPT efficiencies parasitics improved.

HSD : *HPC in the only module of which both airflow and efficiency are significantly affected.*

FIG.7

SUMMARY OF CF6-50 SN 455169 MODULE ANALYSIS RESULTS

1978

TEST Date	TEST Case	MODULE OF HARDWARE Changes Incorporated	Accuracy of analysis HSD program	Accuracy of analysis GE program
17/4	A	Original		
26/4	B	Core replaced	new baseline	
17/5	C	HPC case	correct	correct
16/6	G	LPC 4th stage	correct	uncorrect
10/07	H	HPT NGV repaired	correct	uncorrect
	K	HPC rework	correct	partially correct
	L	repeat of K	correct	uncorrect
	M	T54 Harness	correct	uncorrect
	N	repeat of M	correct	correct
12/9	O	THP rework	partially correct	uncorrect
18/9	Q	LPT	Unclear	correct
10/10	R	New NVG s	partially correct	unclear
17/10	S	New combustor	partially correct	partially correct
	T	VSV closed 1,4°	-	correct

Qualification "correct" "uncorrect" "partially correct" for the diagnosis, means the following :

Correct : Prime module changes identified in direction and magnitude with little or no question regarding which module was effectively altered.

Partially correct : non conclusive, hardware change effect identified but additional changes also detected in other modules enough to "cloud" the diagnosis.

Uncorrect : Analysis failed to isolate or identify the proper hardware changes.

FIG.8

UPGRADE-PG

```
 TYP USE   ENGINE NO DATE   TIME    TT2   PT2  HUM   A2B    AB BEL   PS2  PID
 OPA TEST  455705-50 100380 TO    9.000 14.39   33 CDGF  CDGC  CDG 12.37  200

            N1   N10      O/T  O/P   VIBF VIBA  WETB DRYB                 LHV
          3740 3777.                                                    10640

PARAMETER    W20     N2   T2C  PS2C     T3   P33    WF   TS4   PS4    FN  VB   VSV
RAW DATA          10105 97.00 26.77    529 400.0  8357   823 91.02 44998  RD  -.30
AERO CORR    1402 10105 103.0 1.859   542. 20.41 19191 840.0 6.325 45757  VB   1.7
CORR DATA    1423 10230 108.6 1.861   558. 20.03 19730 852.1 6.421 51427  NC 8870.
BASELINE     1446 10185 106.0 1.825   558. 20.44 19484 847.4 6.425 50700  EV  -.30
TOTAL DEL   -1.50 -.441 -1.26 -.150  -1.91  1.37 -2.73 -1.00 -1.40  10.6  WV  1.38
DEL AN      0.000 0.000 0.000 0.000  0.000 0.000 0.000 0.000 0.000 0.000  EH  .109
GROSS DEL   -1.61  .444  .469  2.01   .044 -1.44  1.26  .424 -.076  1.04  WH -2.38
SENS DEL     .533  .256  .040  .289  -.054 -1.03  .264 -.202 -.390  .463  N  8716.
DELTA ADJ   -1.76  .293  .212  .649   .080 -.759  .572 -.114 -.125  .716  NE -.423
NET DELTA    .147  .151  .250  1.36  -.036 -.684  .691  .539  .049  .320

MODULE       FAN     LPC     HPC     NGV     HPT     LPT      CALC NET DELTAS  SENS  DEL
TYPE  S/N   C       C       C       C       C       C            THRUST    .3  TT2 -.170
DELTA EFF     .20    -.20    -.19            -.35                TSFC      .3  PT2 -.379
DEL WO/AT     .33    -.14   -1.00     .70     --     .21         VSV     -2.0  N1  -.034
TIT EFFECT    2.0     1.5     3.1    -1.3    2.3     2.3         TIT C    6.0
OPL SHIFT     .11     .91     .01      MESSAGES ON PAGE 2        EGT HD 983.0     NEG YES

PERFORMANCE AT RATED N10 OF 3703. RPM:
N2R = 10206. RPM
WFR = 19717. LBS/HR
EGTR =  855.1 DEG C
FNR = 51200. LBS
EPR = 6.453 RATIO
TSFC =  .385 RATIO
CORE SPEED MARGIN  136. RPM BELOW ACCEPTANCE LIMIT
FUEL FLOW WITHIN EXPECTED BAND
EGT MARGIN   7.5 DEG C BELOW ACCEPTANCE LIMIT
THRUST MARGIN 1332. LBS ABOVE ACCEPTANCE LIMIT
EPR WITHIN EXPECTED BAND
TSFC  .020 RATIOS BELOW ACCEPTANCE LIMIT
  7.0 DEG C EGT HOT DAY MARGIN
HPC FLOW DEVIANT
```

FIG.9

PARAMETRES PRINCIPAUX REACTEUR

Precision et Fiabilite requises sur banc d'essais sol et sur avions (Exemple CF6.80/A310)

Source.GE avril 1980

PARAMETRE	PRECISION		FIABILITE	
	BANC SOL	SUR AVION	BANC SOL	SUR AVION
N_1 ou N_2	± 0,1 %	± 0,5 %	± 0,1 %	± 0,25 %
EGT	± 3° F	± 3°F	± 1,67° C	± 3° C
Wf Débit carburant	± 0,5 %	± 0,5 %	± 0,4 %	± 0,1 %
P_t 5,4 et Ps2c	± 0,25 %	± 1,0 %	Inf. à 0,5 %	± 0,25 %
Ps 3	± 1,5 PSI	± 1,0 %	Inf. à 3 PSI	± 0,25 %
T_3	± 1,67° C	± 3,0 ° C	Inf. à 3°C	
T_{2c}	± 1° C	± 1,0°C	Inf. à 2°C	± 0,5°C
VSV	± 0,75 degré	± 1,5 degré	Inf.à 1,5 degré	± 0,5 %
OAT ou TAT	± 0,5°C	± 2°C	Inf. à 1°C	± 1°C
PT2	± 0,2"H_2O	± 1 %	Inf.à 0,4"H_2O	± 0,25 %
Po	± 0,007 PSI	± 50 pieds	Inf.0,015 OSI	± 10 pieds

FIG.10

SUIVI DE PERFORMANCES AVION B747/CF6.50E
ECARTS AVEC REFERENCE
DES DEBITS CARBURANTS CROISIERE
MOYENS MENSUELS

1°SEMESTRE 1979

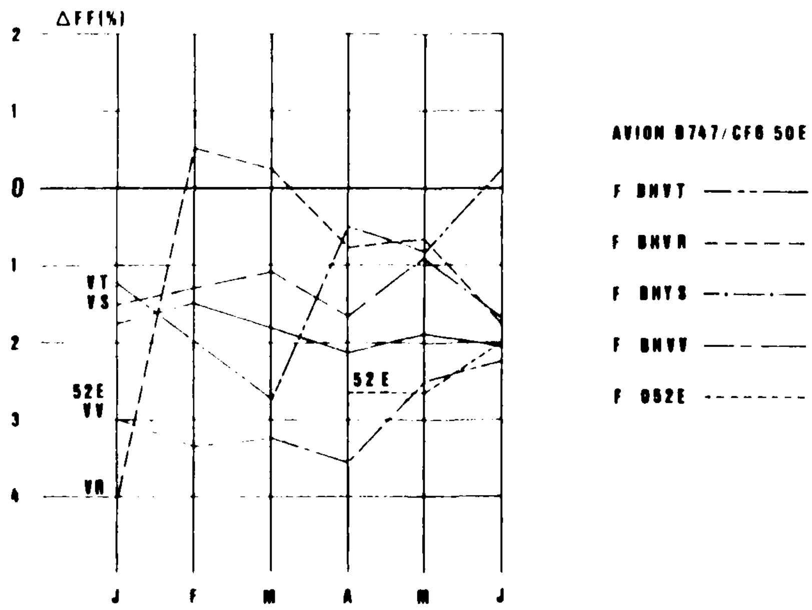
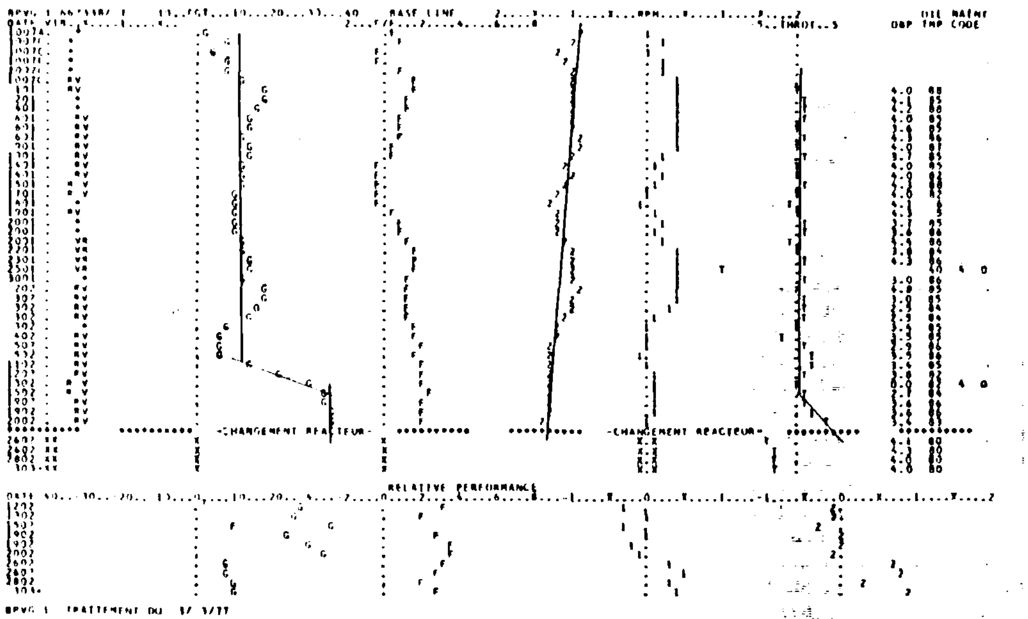


FIG 11

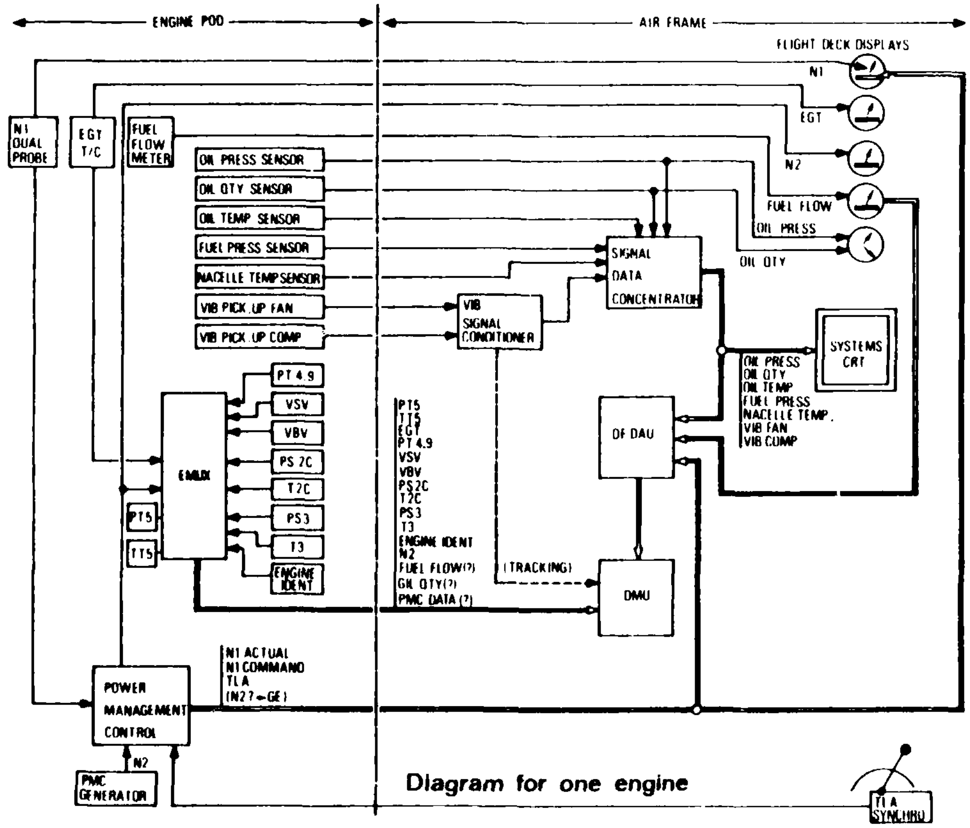
B 747 JT9D.7 ENGINE CRUISE MONITORING PLOT



COMBUSTION CHAMBER INNER LINER DISTRESS

FIG 12

A 310 CF6.80 A1. EXPANDED AIDS DIAGRAM WITH EMUX



[illegible]

AUTHORS: [illegible]

SUMMARY

[illegible] in-flight data [illegible] Maintenance [illegible].

[illegible] military engine lifing. [illegible] summary [illegible] of the results achieved.

1. [illegible]

[illegible]

[illegible] present the [illegible] military aircraft during the last two years. [illegible] work in this area [illegible] and it is the intention to [illegible] factors which have influenced the work in this subject since that time. [illegible] to develop equipment and techniques which are [illegible] to provide an integrated and flexible approach to the accepted UK Services policy of aero-engine On Condition Maintenance (OCM). The way in which this is being achieved and exploited further is detailed.

It has been recognised that in order to achieve OCM, management of individual engines by monitoring their usage [illegible] offers the best approach to a comprehensive maintenance package. Details of a [illegible] trial to establish the cost effectiveness of this approach is covered in the paper. [illegible] trial has assumed even greater importance by having complete service usage [illegible] limited components. These data will be used for in-depth assessment of life prediction [illegible].

This review explores future possible concepts for engine in-flight data collection and analysis to meet the requirements of advanced military aero engine technology.

2. [illegible]

2.1 The [illegible] Usage Monitoring System (EUMS). The following [illegible] in a variety of different applications.

2.1.1 EUMS Mk 1 and In-Service Low Cycle Fatigue (LCF). The description [illegible] of EUMS Mk 1 was detailed in Ref 1 and experience with the system has [illegible] over the past two years. [illegible] initially to the results of the low cycle fatigue programme, [illegible] accumulation of engine ground and flight hours [illegible] provide a more meaningful sample from which individual fleet exchange rates [illegible] refined.

The EUMS Harrier has joined the Hawk, Gnat, and Tornado, completing a list of [illegible] different aircraft types fitted with the system. A further [illegible] different types namely Sea Harrier, VC10 Tanker, Chinook, Puma, Jetstream and Sea King are planned for [illegible] fitment.

Statistical techniques which take account of variations in engines, pilot performance, type of sortie flown and other random factors have been developed to provide an equivalent [illegible] hourly exchange rate, based on the general equation $y = A + Ct$

Where y = LCF total consumption for a particular sortie y

A = LCF consumption due to take-off and landing (TOL)

C = LCF consumption per hour for the particular sortie pattern [illegible]

t = Total flight time.

Assuming that the amount of random errors is proportional to t, the average TOL free rate for LCF consumption per hour for a particular sortie is $C = \frac{(y - A)}{t}$

[illegible]

Mission type [illegible]	[illegible] average [illegible]	
[illegible]	[illegible]	[illegible]
[illegible] of aircraft	[illegible]	[illegible]
[illegible] sortie pattern [illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]

[illegible] RAF [illegible] fleet [illegible] exchange rate [illegible] programme at RAF Strike Command [illegible] EMU [illegible] flying patterns [illegible] Service aircraft may [illegible] exchange rate [illegible]. Whatever the [illegible], EMU will provide the basis for future direction.

[illegible] fleet exchange rate for [illegible] Harrier, [illegible] EMU [illegible] remaining time expired [illegible] significantly [illegible]. Similarly, EMU data from RAF [illegible] as a basis for life [illegible]. An average increase [illegible] has been agreed.

The RAF Red Arrows display team [illegible] consumption rate [illegible] dependent [illegible] aircraft [illegible] formation [illegible] vary by a factor of [illegible]. EMU [illegible] life consumption of the team.

[illegible] EMU Installed Engine Performance Investigation. [illegible] EMU has recently [illegible] part of a more comprehensive [illegible] steady state performance characteristics of the [illegible] This exercise was of particular significance in that performance was [illegible] aircraft installation [illegible] test bed and compared with the in-flight characteristics using the same data acquisition system [illegible].

[illegible] EMU was [illegible] the [illegible] performance parameters [illegible], fuel flow, [illegible] additional [illegible] equipment [illegible] pressures and temperatures [illegible] EMU were [illegible] computer compatible tape for further detailed performance analysis by Rolls Royce. The results achieved [illegible] that EMU has the capability to undertake comprehensive steady state performance [illegible]. More significantly [illegible] potential [illegible] the EMU Mk [illegible] an in-service engine condition monitoring [illegible] programmed to provide engine performance criteria, particularly [illegible] in helicopter power index measurements.

[illegible] EMU Mk [illegible] New Engine Specifications. A knowledge of engine throttle movements, and the resultant component cyclic life consumption on existing aircraft may be utilised

[illegible] future aircraft at the preliminary [illegible] projected aircraft [illegible] aircraft [illegible] similar [illegible]. This [illegible] more [illegible] projected aircraft are similar [illegible] from several types of aircraft [illegible] projected aircraft's mission. This is because [illegible] existing types [illegible].

[illegible] performance [illegible] aircraft [illegible] in the late [illegible] simultaneously defining [illegible] between MOD and Industry [illegible] aircraft [illegible] time, and [illegible] more [illegible] frequency [illegible] Thus, the [illegible] from certain [illegible] aircraft [illegible] aircraft [illegible] data [illegible] systematically [illegible] components of the [illegible] aircraft. In particular, the [illegible] life [illegible] influenced by the data [illegible].

[illegible] potential area for further [illegible] current [illegible] that [illegible] material [illegible] benefits [illegible] prerequisite [illegible].

[illegible] components are withdrawn from service [illegible] depending upon the component [illegible], having been monitored at [illegible] (NDI) techniques. This inspection period is [illegible] the accuracy of [illegible] available, maximum [illegible] characteristics but also (just as [illegible]) [illegible] during the inspection intervals.

[illegible] demonstrated the variations in [illegible] rates for [illegible] the extremes in aircraft flying different sorties [illegible] gain the maximum benefit from [illegible] the use of a 'nominal' [illegible] of the worst flying profile would be waste[illegible] therefore, to derive the considerable advantages offered [illegible] engine usage monitoring is undertaken.

[illegible] In addition to the aircraft installations outlined [illegible] engine test bed to record endurance [illegible] existing test bed more comprehensive [illegible].

[illegible] recording on the test bed is to gain experience [illegible] accumulated during [illegible] hour Type Test, [illegible] including the evaluation of new [illegible] changes to the engine [illegible] have [illegible] engine strip with the engine running [illegible] validation of lifing calculations as well as [illegible].

[illegible] two way exchange. Data acquired from service missions are [illegible] realistic test schedules can be compiled. [illegible] successfully produced test bed failure [illegible] of the effectiveness of optical [illegible] blade temperatures directly is being [illegible] the intention to use the system in the aircraft [illegible] year in RAF normal flying roles.

[illegible] Flight Data Recording Trial (Air Staff Target 603). See Section [illegible].

[illegible] Plans are in hand to expand the main ground data [illegible] to keep pace with the additional aircraft installations coming onto [illegible] the new [illegible] equipment (See Section [illegible]). To support the forthcoming engine [illegible] at RAF Valley [illegible] the station is to provide back-up in the event of [illegible] additional computer programme development capability as necessary. It is also anticipated that with the enhanced computer power available more in-depth analysis will be carried out.

[illegible] The Low Cycle Fatigue Counter (LCFC)

[illegible] Background. The LCFC manufactured by SMITHS INDUSTRIES to MOD specifications is a microprocessor based device which computes the fatigue life consumed by 4 specific features located in the engine rotating components. To do this it calculates the stress ranges seen by these features during rpm excursions and converts these into reference stress cycles. It is being evaluated by the RAF on [illegible] types of aircraft which are also fitted with [illegible]. This enables the LCFC readings to be checked against the results obtained by processing the [illegible] data through the same algorithm in a ground based computer. A full analysis can also be carried out to include thermal effects (not accounted for at present in the counter) and thus the errors associated with their omission to be determined.

The specification for the LCFC requires that the accumulated count should be within [illegible] of the value calculated by the ground based computer using the same algorithm for all [illegible] patterns.

[illegible] Experience. The LCFC is currently fitted to [illegible] aircraft and the majority of [illegible] have been accumulated on the Adour engine in the Hawk and Jaguar. The experience gained [illegible] years of service use has enabled the cyclic exchange rate to be reduced from [illegible] [illegible] cycles/hour mainly on the basis of [illegible] information, which also indicates that with an [illegible] on each aircraft the figure would come down to around [illegible]. The Hawk entered service [illegible] years ago and is also used by the RAF aerobatic team, the Red Arrows. The [illegible] cyclic exchange rate is generally [illegible] cycles/hour but the Red Arrows rates are estimated to vary between [illegible] and [illegible] cycles/hour depending on the formation position, making them a priority candidate for the LCFC.

The experience has revealed several problems which have been overcome by changes to hardware and software. In particular, power supplies on older aircraft from a pre-digital era were found to have characteristics which were outside current electrical specifications. Problems were also caused by the microprocessor reset timing and by [illegible] [illegible] generated [illegible] on the input speed signal. All these have now been overcome [illegible] that the LCFC has been proven to operate within specification on the Hawk. A cumulative accuracy of [illegible] is being achieved. The evaluation will be continued on the Jaguar, Harrier and the Hunter to give a range of engines and environments.

The validation exercise has been carried out at the National Gas Turbine Establishment [illegible] with [illegible] tapes and LCFC readings. Further analysis has also been carried out by [illegible] at MOD request to determine the effect of neglecting thermal gradients. The results have been very encouraging and give support to the assumption that thermal effects may be ignored for some components.

[illegible] Read Across. Ideally, each life limited component on the engine would be monitored by a separate channel on a multi channel counter, but the requirements of low cost, simplicity and high reliability make this prohibitive. An alternative method is to monitor only one or two components on each rotating shaft and to derive the fatigue damage accumulated on unmonitored components from the figures relating to the monitored components. This technique is called 'Read-Across'. Work done at NGTE has shown that this method can achieve a high degree of accuracy when using a linear relationship between the damage accumulated by components on the same shaft. The read across equation is derived by analysing flight traces of spool rpms to obtain arithmetic mean values of rpm turning points, these are then used in an equation which has the form $Y = AX+B$

Where Y = fatigue per flight of unmonitored component

X = fatigue per flight of monitored component

A and B = constants

The line described by this equation does not extend below the value of 1 reference cycle because it is assumed that at least 1 reference cycle will be consumed on take-off and values below this level constitute ground runs. The other assumption made in the derivation of the equation is that both components experience the same number of stress time turning points, that is to say, they are on the same shaft and are not significantly affected by fluctuating thermal stresses.

[illegible] Thermal Stresses. In areas where thermal gradients or thermal fatigue are significant, or where the life limiting mode is not centrifugally induced low cycle fatigue, the [illegible] [illegible]. These problems are dealt with by additional equations in the LCFC or by alternative equipments. However on a number of RAF in-service engines, group A components [illegible] lives are largely independent of thermal effects. In cases where the thermal effects cannot be ignored, three alternative courses are available, all of which offer advantages over [illegible] components in flying hours.

[illegible] of all, the LCFC is being developed to take account of turbine blade thermal fatigue [illegible] [illegible].

[illegible], as the LCFC algorithm uses a value of [illegible], if a temperature value were available it would be a simple matter to adjust the [illegible] to take account of changes in material temperatures and thus compensate for different operating conditions.

[illegible] parameter [illegible] thermal gradients [illegible] relevant [illegible] respectively to refine an initial [illegible] life. [illegible] techniques of EUMS Mk [illegible] used in conjunction with [illegible] in the first [illegible] categories, would provide the best solution.

[illegible] EUMS

[illegible] Engine Usage Monitoring System (EUMS) Mk 2

[illegible] The Requirement. The existing EUMS Mk 1 being used in service has proved to be an adequate and reliable equipment. Initially suffering from a slow build up of data the total system is now very firmly established with the Royal Air Force and entering into [illegible] Navy applications. Winkler and Nelson in their report (Ref [illegible]) highlighted the differences between the US and UK approaches to engine monitoring. The EUMS Mk 1 concentrates on engine lifing and usage, exploiting the range of design and test appli[illegible] [illegible] Section [illegible]. The UK considered it essential to have experience in this area before attempting to define a complete engine maintenance package.

[illegible] Philosophy. The initial concept of a low cost Data Acquisition Unit (DAU) and Quick Access Recorder (QAR) has proved to be the right decision and has enabled the system to be introduced into densely packed aircraft as a retrospective fit. Experience in the field with the EUMS Mk 1 focussed our thought on improving some of the features [illegible] [illegible] capabilities but provide additional features consistent with the [illegible] engine life monitoring requirements. The EUMS Mk 2 was therefore developed to Ministry specification requirements by Plessey Electronic Systems at Havant.

[illegible] Specification. It is a microprocessor based system which has been designed to monitor continuously and analyse in real time the performance of aero gas turbine engines. The system includes a display suitable for either cockpit or equipment bay mounting, which gives parameter rating exceedance warnings and immediate access to engine life data, without reference to ground replay equipment. In addition, the system retains the facility for retrospective analysis of the flight data by recording this information on an airborne cassette loaded recorder.

The airborne system comprises three units: the Engine Usage Life Monitoring Unit, the [illegible] Display Data Unit and the Quick Access Recorder (QAR).

The system has been designed to provide versatility of operation, allowing it to be easily programmed to function in different operational modes, from in-depth investigation of problems arising in the operation of engines, to the routine gathering and recording of engine data to derive performance and life information. To achieve this versatility, the system has an efficient and flexible data storage capacity, [illegible] with an easily programmed processor capable of handling both a wide range of [illegible] signals and data rates.

[illegible] Characteristics. The system can accept some [illegible] analogue parameters of the following types at a sampling rate programmable from once to [illegible] times per second: [illegible], AC Voltage Ratio, DC Voltage Ratio, DC Voltage Absolute, Potentiometer and Variable Resistance. In addition, the system accepts [illegible] speed inputs and [illegible] signals. The data rate is programmable from [illegible] to [illegible] words per second [illegible] the QAR.

[illegible] parameters can be checked and analysed for limit exceedances. The analysis may be carried out against simple fixed limits, or against limits computed from one or more variables and is available for immediate display to the flight crew or ground maintenance personnel. [illegible] consumption rates per flight and accumulative values are available for a number of critical engine components. Memory space is allocated to enable additional computations to be incorporated for other lifing procedures.

Data can be stored in either solid state memory or Quick Access Recorder for retrospective recording. The solid state memory is intended for use when the operational procedure [illegible] to extract data from the system for detailed analysis after each flight. For retrospective recording, the QAR has a typical storage capacity of [illegible] hours and is intended for retaining a history of engine performance over a number of flights. In the [illegible] recording mode, durations for the QAR are from [illegible] to [illegible] hours depending upon input data rate and type of cassette used.

[illegible] Experience. Considerable experience has already been gained with the EUMS Mk [illegible] fitted to an RAF Jaguar in operational service. Although the particular application was an investigation of airframe fatigue, the opportunity was taken to include principal engine lifing parameters.

Further development of the system on Barrier flying test bed has taken place at [illegible] [illegible]. A photograph of the installation is included at Fig [illegible].

Recovery of the data from the cassette is effected in the same manner as for the original EUMS Mk 1 system at the main data processing facility. Overall management of the EUMS Mk [illegible] is carried out using the procedures and experience gained with the Mk 1.

5. [illegible] Monitoring

5.1 Background. There is a service need for a meaningful onboard indicator or [illegible] monitor based on proven analytical precepts. In attempting to develop a procedure for predicting the life of turbine engine components validation of the design process initially used to specify the components is required. Design of an engine part is a well established reiterative process based on aircraft requirements, material characteristics and structural analysis verified by a suitable test programme. The UK is currently looking to the [illegible] programme to provide the data necessary to confirm life predictions and justify the appropriate procedures. A disciplined approach is suggested for examination of selected in-service components having known usage histories based on EUMS recordings.

Creep and thermal fatigue algorithms have been developed by Rolls Royce and NGTE and validation of these will be undertaken using measurement of blade temperatures and shaft speeds coupled with the service history of the component under review. By establishing a programme based on phased removal of components up to the failed condition, it will be possible to validate blade life predictions from metallurgical examination, determination of deformation (or failure mode) and by carrying out additional tests on unfailed components to determine remaining life.

5.2 Requirements. Based on the work already carried out to define methods of computing turbine blade creep and thermal fatigue from EUMS tapes, it was logical to extend these procedures to onboard calculations using the spare capacity which had been built into the current [illegible]. The essential parameters needed to define creep and thermal fatigue are, compressor outlet temperature (turbine blade cooling air), turbine inlet temperature and shaft speed.

On a [illegible] typical engine this meant that a minimum of 5 engine parameters would be required: low pressure shaft speed, high pressure shaft speed, total intake temperature, total intake pressure and turbine gas temperature.

From these parameters blade bulk temperatures and average section stresses can be calculated. This information is then applied to the Graham-Walles creep equations:

$$D_{creep} = K1 \sum \frac{\sigma^{K_2} \, dt}{(K3 - T)^{20}}$$

Where D_{creep} = cumulative creep damage

σ = blade average section stress due to centrifugal loads

dt = time interval between samples

T = blade transient bulk temperature

$K_1, K_2, K3$ = constants

Further calculation will give the blade instantaneous peak temperatures and therefore the temperature differences across the blade section and hence the thermal strain. This can be calculated continuously and added to the centrifugal strain; strain turning points can be detected and a rainflow technique used to determine strain cycles and their corresponding stress ranges. From this point the fatigue damage accumulated by that section of the blade can be estimated.

5.3 Progress. [illegible]

[illegible]

6. [illegible]

6.1 [illegible]

[illegible] staff [illegible]

[illegible] we [illegible] the following [illegible]

[illegible] the parameters [illegible]

[illegible] aircraft [illegible] the [illegible]

[illegible] equipment [illegible]

[illegible] have been [illegible] and quantify them.

[illegible] aircraft [illegible] availability and flight safety have been [illegible] and quantify [illegible].

[illegible] transfer of information and data between the RAF, Ministry of [illegible] and Industry [illegible]

[illegible] RCM [illegible] maximum improvement [illegible].

[illegible] what extent the Hawk RCM trial can be related to other aircraft [illegible] which additional analysis will be required for individual [illegible].

[illegible] management structure which would be necessary to integrate [illegible] system [illegible].

[illegible] the Trial.

[illegible] We intend to centralise most of the activities at the operating station [illegible] communication lines and to ensure that the engineering [illegible] as quickly as possible. The Hawk was [illegible] training aircraft, it has a modular engine and much of [illegible] aircraft. [illegible] the influence [illegible] which will [illegible] the initial [illegible] aircraft will be fitted with [illegible] equipment and [illegible] there will [illegible] will be fitted with [illegible] it will be necessary to [illegible] reliability, they will [illegible] will be [illegible] to provide engineering [illegible] maintenance [illegible]

[illegible] operating [illegible]

[illegible] maintenance [illegible] RAF [illegible] management [illegible]

[illegible]

[illegible] demonstrate that the feasibility of using in-flight parameter monitoring to indicate engine [illegible], the recorded data will be used to indicate the serviceability of an [illegible] for its next flight and for planning future maintenance. [illegible] engine [illegible] trend analysis [illegible] show the changes of condition which will enable the serviceable [illegible] unserviceable state of any particular engine [illegible] the normal variation of condition monitored in a fleet of apparently identical serviceable engines.

[illegible] system will therefore [illegible] the data [illegible] serviceability during the previous flight and [illegible] performance data [illegible] update the engine file, [illegible] trend and give [illegible] in-flight engine [illegible]. It will also provide [illegible] data [illegible] failures, [illegible] or any other engine [illegible] which may [illegible].

[illegible] will determine the method of data presentation [illegible] serviceability [illegible].

[illegible] data [illegible] small and efficient [illegible] equipment [illegible] computer [illegible] data [illegible] time [illegible] data [illegible] aircraft flight line.

[illegible] software. [illegible] are preparing the software which [illegible] major influence [illegible] programme. We have all learned [illegible] in the past from underestimating [illegible] software which give the required results. [illegible] over from hardware [illegible] project [illegible]. Particular [illegible] area in the development period. We are [illegible] software will [illegible] software documentation [illegible] Service [illegible].

[illegible] software [illegible] flight data transcription and analysis and [illegible] maintenance [illegible] information.

[illegible] minimum manual intervention. The main programme [illegible] information [illegible] data from [illegible].

[illegible] computer [illegible]

[illegible]

[illegible] the analysis programs for flight record verification [illegible]. The [illegible] against [illegible] components. [illegible] replacement parts [illegible] Maintenance File [illegible] data [illegible] the maintenance [illegible] from aircraft. It [illegible] divided into two [illegible] the aircraft. [illegible] trial aircraft and engines, [illegible] repairs [illegible]. The [illegible] will be [illegible] the data [illegible] the master file [illegible] trial [illegible].

[illegible]

[illegible] this paper [illegible]

[illegible] operational [illegible] performance [illegible] performance data from this facility and from the [illegible] rather than [illegible] data from [illegible]

[illegible] Engine Health Monitoring. The [illegible] analysis, [illegible] inspection, filter debris, magnetic [illegible] spectrometric oil analysis, ferrography and [illegible] will be used [illegible] and the trial will [illegible] from flight data recording.

[illegible] With [illegible] of the actual [illegible] we will [illegible] the EHM techniques are cost-effective. [illegible]

[illegible] difficult to handle and to be [illegible]. The [illegible] As the diagram shows, [illegible]

[illegible]

[illegible]

[illegible]

[illegible]

[illegible]

[illegible] parameters [illegible] for more detailed investigation [illegible] required.

[illegible] data analysis [illegible] formats [illegible] [illegible] from [illegible]

[illegible] system [illegible] with [illegible] terminal [illegible] for detailed analysis [illegible]

[illegible] Data Analysis [illegible] programme [illegible] a variety of [illegible] performance [illegible] parameter [illegible] time in flight [illegible] terminal will [illegible] future [illegible] the [illegible]

[illegible]

[illegible] engine [illegible] monitoring programme [illegible] the definition of a comprehensive [illegible] policy.

[illegible] will be used in a number of applications [illegible] outlined in the paper, the main [illegible] which [illegible] correlation and validation of [illegible]

[illegible] requirement for a flexible data acquisition and analysis system capable of [illegible] variety of applications. Electronic technology developments are sufficiently [illegible] provide a system at a reasonable cost and timescale. Current developments are [illegible]

[illegible] will continue to be improved by providing additional [illegible] the basic on-board unit for the more comprehensive system suggested [illegible]

[illegible] engine flight data recording trial being carried out by the Service will [illegible] direction for future [illegible] EHM requirements and determine the need for an engine [illegible] monitoring package which must be cost effective and capable of integration into the [illegible] maintenance policy of the Royal Air Force.

ACKNOWLEDGEMENT

[illegible] wish to [illegible] the co-operation of associates in Industry and also to their [illegible] in the Ministry of Defence for assistance in the preparation of this paper.

[illegible] expressed in the paper are those of the authors and do not necessarily reflect Ministry of Defence [illegible].

[illegible]

1. [illegible], [illegible], Engine Monitoring Development [illegible] UK, [illegible] Paper [illegible]

2. [illegible], Aircraft [illegible] Engine Monitoring experience [illegible] Evaluation [illegible], [illegible] report [illegible], April 19[illegible].

[illegible] © Controller HMSO [illegible] 1[illegible]

ENGINE PERFORMANCE RESULTS USING EUMS EQUIPMENT

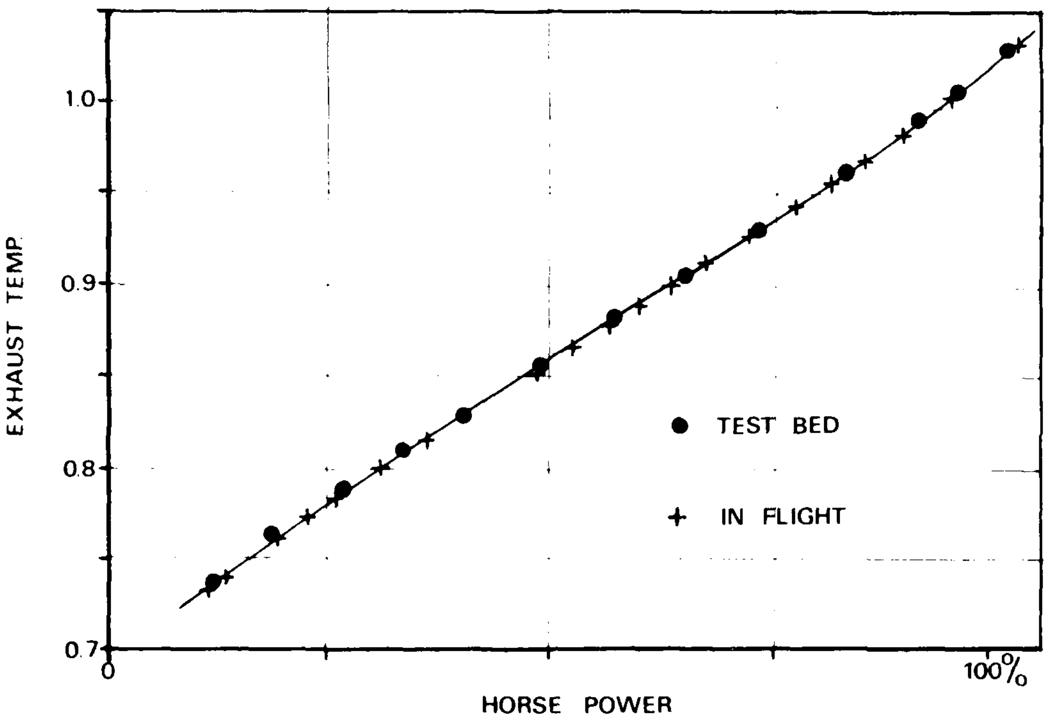
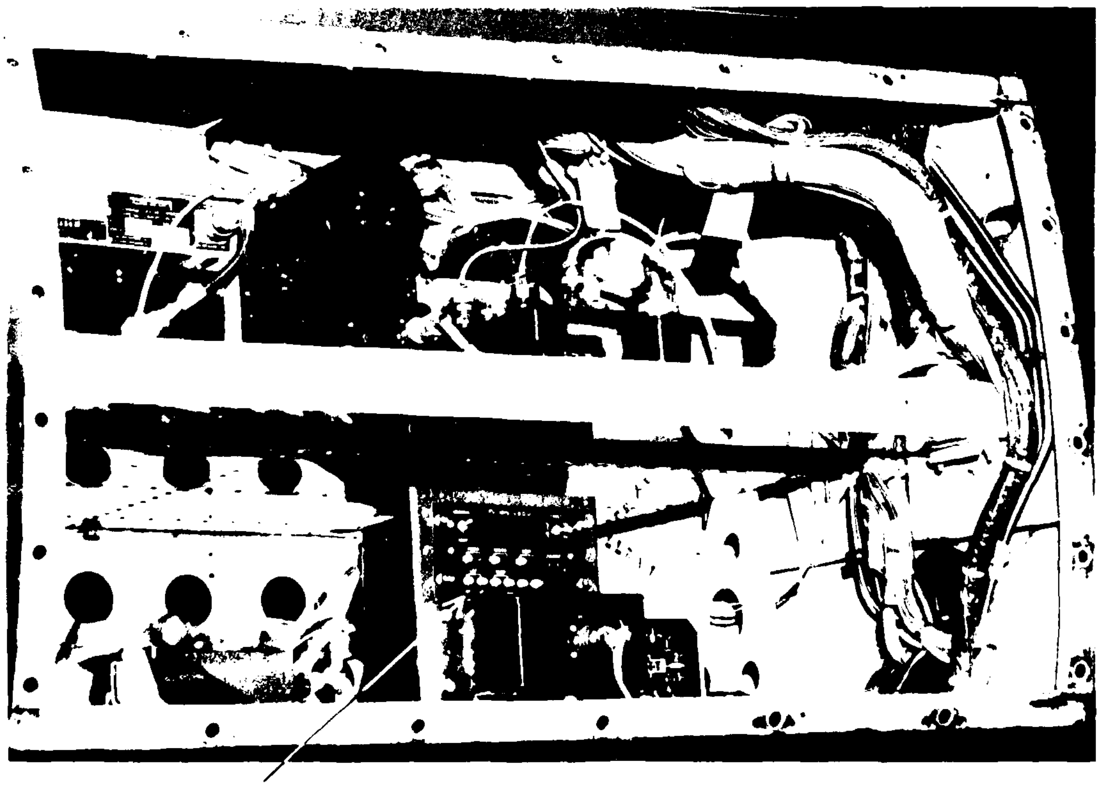


FIG 1



EUMS MK 2 IN HARRIER – VISUAL DISPLAY DATA UNIT

FIG 2

AST 603 GROUND DATA PROCESSOR UNIT

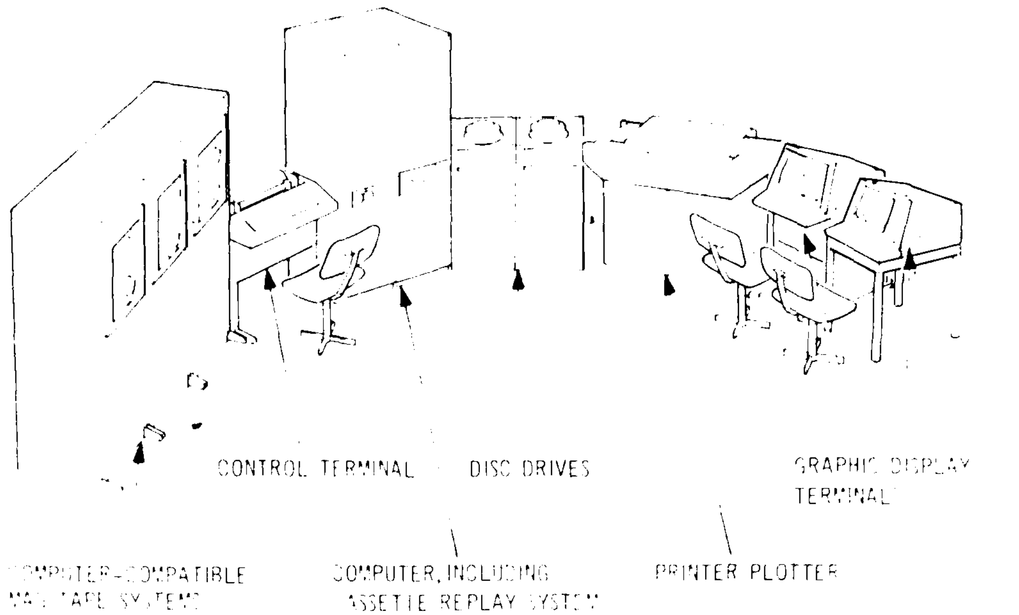
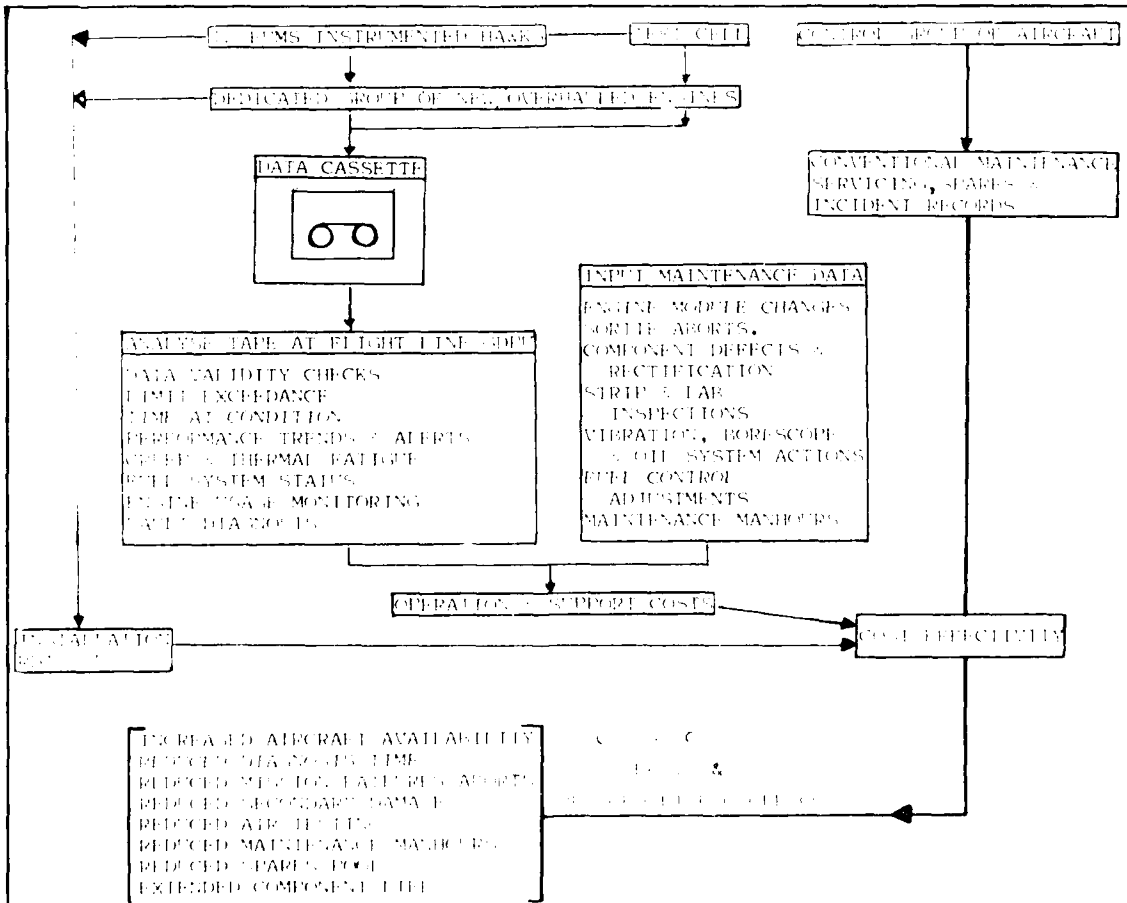
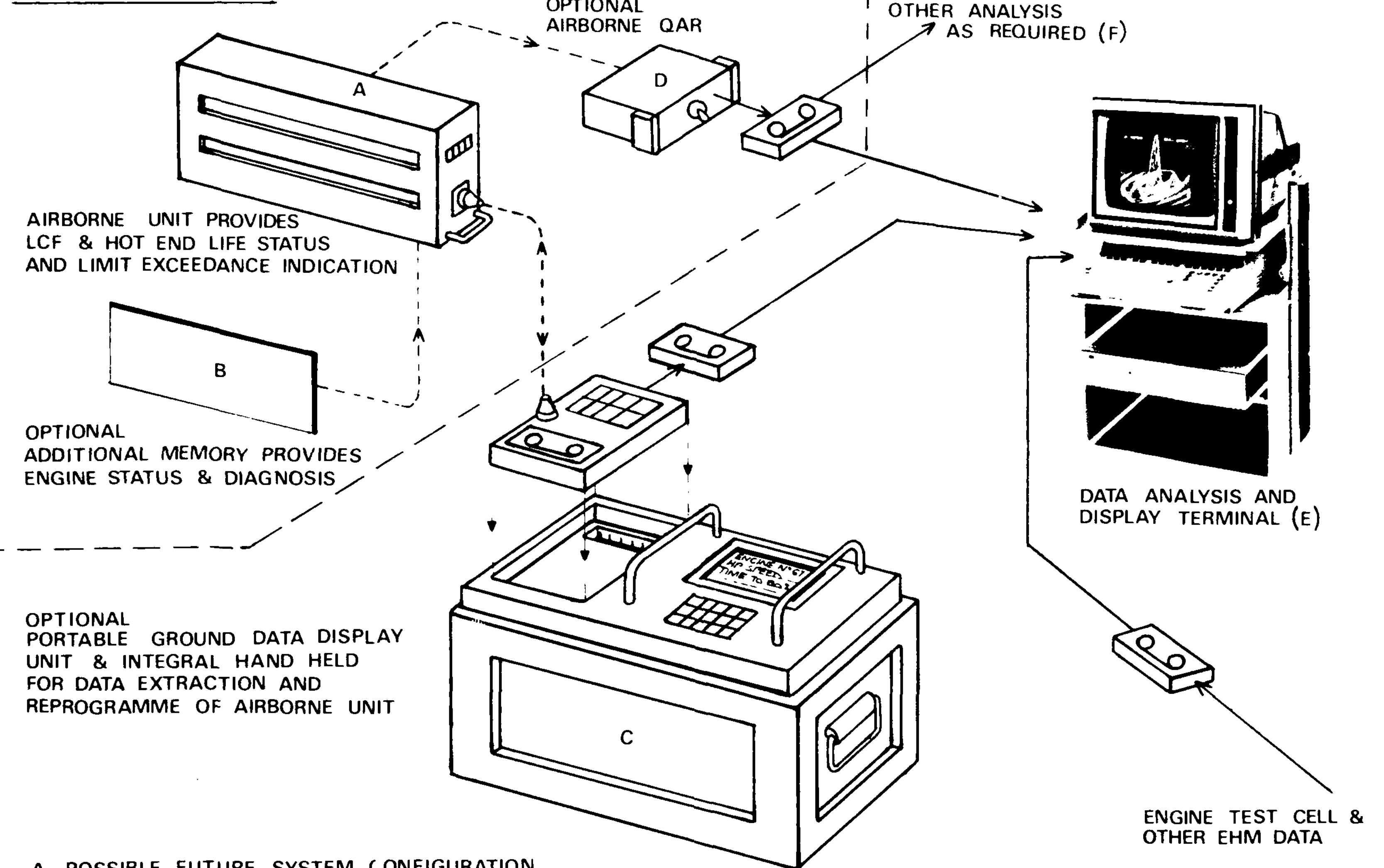


FIG. 3



DATA PROCESS FLOW

FIG. 4



A POSSIBLE FUTURE SYSTEM CONFIGURATION

FIG.5

Investigation of Performance Deterioration of the CF6/JT9D High-Bypass Ratio Turbofan Engines

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SUMMARY

The extent and magnitude of performance deterioration of the Pratt and Whitney JT9D and the General Electric CF6 engine models is presented. Overall engine and contributing module performance deterioration with respect to flight cycles and/or time has been analyzed. The overall engine performance deterioration analyses were based on data obtained from historical records, special engine tests, and tests for specific effects. Hardware inspection data from overhaul shops and special module tests were the basis for the modular performance deterioration data used in the analyses. Various damage mechanisms such as seal rubs, erosion, surface roughness and thermal distortion, and how they contribute to performance deterioration were included in the modular analyses.

Results indicate that early performance deterioration (also known as short-term deterioration) occurring within the first few flights of these engines is less than 1 percent in cruise specific fuel consumption (SFC), that it is event-oriented, and that it is the result of increased blade tip clearances. This performance deterioration gradually increases to about 2.5 to 3.0 percent (including the initial short-term deterioration) after 2500 to 3000 flights where increased blade tip clearances, airfoil quality degradation and thermal distortion are the contributing causes.

INTRODUCTION

Efficient air transportation is an international concern since commercial aircraft constitute a primary segment of public transportation and as such has a significant influence on the world's commerce. The viability of air transportation is threatened by rapidly escalating fuel prices. Figure 1, taken from U.S. Civil Aeronautics Board (CAB) data (ref. 1) illustrates the recent trend in U.S. airline fuel prices. In late 1973 and, recently in 1979, fuel prices have dramatically increased. These latest fuel prices, if converted, approach 60 percent of the airlines' Direct Operating Costs (ref. 2). In response to the resulting need for greater fuel efficiency, the National Aeronautics and Space Administration (NASA) initiated the Aircraft Energy Efficiency (ACEE) program in 1975. Engine Diagnostics, managed by the Lewis Research Center, is an element of the ACEE program in which performance deterioration studies were conducted by Pratt and Whitney for the JT9D and by General Electric for the CF6 turbofan engines. The basic objectives of the Engine Diagnostics program were: (1) to identify and quantify the causes of the engine performance deterioration that increase fuel consumption and (2) to develop the data necessary to minimize performance deterioration of current and future engines, (ref. 3).

In investigating the mechanisms that contribute to performance deterioration, the general approach was:

1. Gather historical (existing) data from airline in-flight recordings and from test cells at both airline and engine overhaul shops. Also, conduct inspections on selected used parts that contribute to performance deterioration.

2. Augment this information with special engine tests and inspections in order to evaluate the effects of deteriorated components, and subsequent refurbishment, on both overall performance and engine module performance.

3. Using these data, establish statistical trends and analytical models.

4. Isolate the performance deterioration to specific events and/or modules and conduct tests for specific effects to quantify in more detail the causes of the performance deterioration and the sensitivity of performance to these causes.

The investigative results are not complete, but sufficient results are available to identify the causes and sources and to establish the magnitudes of performance deterioration of the JT9D and CF6 high bypass ratio turbofan engines.

ENGINE DESCRIPTION

The Pratt & Whitney JT9D-3A/7/20 and General Electric CF6-6D engines, Figures 2 and 3, are used to power the Boeing 747 and Douglas DC-10 airplanes. Both engines are assembled and maintained via a modularized concept. Typical module nomenclature for the CF6 engine is shown in Figure 4. The JT9D nomenclature is similar, except the quarter stage is replaced by a low pressure compressor module. Table 1 describes some of the characteristics of each engine.

It should be noted that the JT9D and CF6 engines are maintained using an "on-condition maintenance" concept. Under this concept the engines are repaired, as required, based on engine inspection data rather than at a fixed time interval between overhauls. This "on-condition maintenance" concept requires the engine be assembled from modules or subassemblies that are completely interchangeable so the engine can be easily disassembled. This permits maintenance of individual modules rather than entire engines and, moreover, permits repair of only those parts that are defective. Thus, in airline practice, when an engine enters the shop for repairs, typically the modules are separated, repaired if required, and dispersed to inventory. A "new" engine is then assembled from modules from inventory and returned to the fleet for service.

RESULTS AND DISCUSSIONS

Historical Data

Performance deterioration trends were determined by both Pratt and Whitney (JT9D) and General Electric (CF6) from analyses of historical information from contributing airlines (representing approximately one-third of the world's fleet of airplanes), aircraft companies, overhaul/repair organizations, and engine manufacturers. Figure 5 shows the sources of these data. Included were (1) data from engine testing (e.g., production acceptance, aircraft acceptance, and pre- and post-repair), (2) normal flight data and (3) observations and documented records of used parts condition and replacement rates.

The major contributors to performance deterioration are shown in Figures 6 and 7. In the cold section of the engine (fan and compressor), airfoil quality (foreign object damage (FOD), erosion, and surface roughness) is significant. In the hot section of the engine, thermal distortion is one of the predominant deterioration mechanisms, causing - for example - warpage or distortion of vanes. Clearance increases, resulting in efficiency losses, occur throughout the entire engine as a result of blades rubbing their outer shrouds.

The performance degradation mechanisms - such as clearance increases, erosion and airfoil roughness, and thermal distortion - affect the cruise specific fuel consumption (SFC) as shown in Figure 8, for the JT9D-3A/7/20 engine modules (refs. 4 and 5) and Figure 9, for the CF6-6D engine modules (ref. 6). In each of these figures, the degradation mechanisms reflect the module condition at three points in an engine's life cycle:

1. First flight of an airplane

2. Multiple flights - typically representing the point at which the engine comes in for first repair

3. Multiple flights after first repair/overhaul

Assumptions used in the data presented (Figs. 8 and 9) were (1) the original modules remained together and were not separated, and (2) only the high pressure turbine module was replaced or repaired between the second and third point in the engine life cycle. This technique was employed to illustrate the contribution of each module to total engine performance deterioration over a number of flights. Summing up all the modular deterioration at a selected number of flights then gives the deterioration for an "average" engine. The combustion system is omitted in Figures 8 and 9 because the direct effect of combustor deterioration on performance is insignificant. It should be acknowledged, however, that the indirect effects on performance can be significant. In addition to reducing turbine life, changes in radial and circumferential temperature patterns affect clearances and cause other mechanical changes in the turbine. The NASA Engine Diagnostics program did not address these indirect effects.

The data represents an average value of performance deterioration based on available samples of historical data. In some instances the data sample was limited (e.g., pre-repair test results) and the data scatter was large. To quantify the variability of the historical data, statistical methods were employed by both Pratt and Whitney and General Electric. Table 2 shows the average values of performance deterioration along with the statistical deviations.

An examination of the historical data reveals, for both engines, an early deterioration - referred to as short-term deterioration - occurs within the first few flights of an airplane. For the studies, it was decided to use the first flight of the airplane as the base for quantifying short-term effects. The subsequent deterioration, referred to as long-term deterioration, is more gradual and increases with flight cycles and/or time until the engine is removed because of mechanical problems or exceedance of established exhaust gas temperature (EGT) limits.

The values for the short-term deterioration for the JT9D and CF6 engines were, in general, determined from both test cell and cruise performance recordings, supplemented by hardware inspections. The assessments from all these data sources indicated the average short-term performance deterioration of the JT9D-3A/7/20 is a 0.7 percent cruise SFC increase while the CF6 value is 0.8 to 0.9 percent. The 0.8 percent SFC value for the CF6-6D is based upon hardware observations and documented records of used parts condition while performance data results, which substantiated the hardware results within the range of data scatter, indicated the value to be slightly higher (0.9 percent). For this paper, the hardware/used part value is used. Hardware/used part examinations showed

that clearance increases are the predominant cause of short term deterioration after the first flight (Figs. 8 and 9) and are a result of rubs of stationary and rotating parts. These rubs in the JT9D are believed to be associated with deflections produced by aircraft induced flight loads that occur during take off rotation, flight maneuvers, landing, and thrust reversal. Engine power transients during these flight events may also contribute to rotor case interferences that produce increases in blade tip clearances.

Examination of the CF6-6D data (Fig. 9) shows that high pressure turbine clearance increases contribute over 90 percent of the total short-term performance deterioration. The cause of this performance deterioration is believed associated with an event which produces rotor case interferences that result in increased blade tip clearances. This event is termed "hot rotor reburst" and, as illustrated in Figure 10, is a type of thermal transient response which can result in high pressure turbine blade rubs due to different thermal growth rates between rotating and stationary structures. This event, while not a normal operation, may occur during aircraft acceptance tests or waveoff.

Long-term performance deterioration data included test cell recordings (pre-repair), cruise performance recordings and hardware inspection records. The assessments from these data sources indicated the average performance deterioration for the JT9D-3A/7/20 at 1500 flights was 2.0 percent in cruise SFC, and this value grew to 3.0 percent at 3000 flights. These values are a summation of the modular contributions shown in Figure 8. For the CF6-6D, (Fig. 9), the performance deterioration values, based on hardware inspection results, at 1650 and 2500 flights was 2.3 percent and 2.4 percent in cruise SFC, respectively.

A closer examination of the data from both engines reveals that the causes of long term performance deterioration are additional blade tip clearance increases in all engine modules along with fan and compressor airfoil erosion/roughness. High and low pressure turbine distortion is also a contributing source to the long term performance deterioration.

As mentioned earlier, the assumptions used in these analyses were that only the HPT was repaired at every repair cycle and furthermore that all other modules remained intact for subsequent engine rebuilds after repair. Re-examining Figures 8 and 9 reveals that for the JT9D-3A/7/20 a potential of about 0.8 percent cruise SFC could be obtained with a HPT repair while for the CF6-6D the value is about 0.9 percent. These values for HPT repair are generally not realized in practice, however, because the data obtained from references 4, 5, and 6 indicates that not all performance deterioration is restored, there being a residual of approximately 0.2 percent cruise SFC deterioration remaining after each repair.

Special Engine Tests

Several tests of JT9D and CF6 engines were conducted to expand the understanding of performance deterioration and more precisely assess modular contribution to the overall SFC loss (Table 3). Pratt & Whitney acquired pre- and post-repair performance data, as well as parts condition information from 32 JT9D-7A engines in Pan American World Airway's fleet of Boeing 747 SP aircraft. In addition, four of these engines were specially instrumented and periodically subjected to on-the-wing ground calibrations during their first 1000 cycles of operation (ref. 5). One of these engines (serial number (S/N) 743) was removed from the aircraft and subjected to extensive performance tests and hardware inspections (ref. 7). Analysis of the data from these efforts corroborated the historical data results which indicated that the JT9D exhibits a cruise SFC loss of about 0.7 percent during its early flight cycles. The performance deterioration in the long-term (1000 cycles) was about 2.0 percent. Of the short-term deterioration that occurs during the early flight cycles, all was attributed to clearance increases throughout the engine. At 1000 flight cycles, about 50 percent of the performance deterioration is attributed to clearance increases while the remainder is caused by thermal distortion in the turbines and increased airfoil surface roughness in the fan and low pressure compressor.

Verification of historical CF6-6D short-term performance deterioration results was accomplished by a series of tests and inspections conducted with an engine that was removed from a DC-10-10 aircraft prior to delivery to American Airlines (ref. 8). This aircraft engine had undergone the normal Douglas Aircraft Company acceptance test flights but had not been introduced into revenue service. The tests following removal of the engine (S/N 507) from the aircraft indicated an increase in cruise SFC of 0.9 percent over the level measured during engine production acceptance tests at General Electric, Table 3. Subsequent engine disassembly and detailed inspection revealed the short-term deterioration to be primarily a result of blade tip-to-shroud rubs causing increased clearance in the high pressure turbine module, again corroborating the historical data results.

Long-term performance deterioration of the CF6-6D engine, as shown in Table 3, was investigated through test and parts inspection of two engines: the first engine after approximately 4000 hours of operation (1910 flights) prior to its first refurbishment (S/N 479) and the second engine (S/N 380), prior to its third refurbishment, after 12000 hours of operation (3740 flights) (refs. 9 and 10). The primary result of this investigation was the identification of the deterioration mechanisms that contribute to long-term deterioration (increased blade tip clearances, increased airfoil surface roughness and erosion, and distortion of parts). The value of overall engine performance deterioration, although less conclusive than for the JT9D because of the small data sample, did fall within the statistical data band, Table 2.

Additional special testing of the CF6 was done to determine the contribution of individual modules and components (Table 3) to the overall increase in engine specific fuel

consumption. Back-to-back tests of CF6 fans and low pressure turbine (LPT) modules established the amount of deterioration attributable to those components. The sequence in which the back-to-back tests were accomplished was: (1) test the engine in its as-received condition, (2) remove one specific module (fan or LPT), (3) repair the removed module or replace it with a new or refurbished one, (4) reassemble the module into the engine, and (5) retest the engine. Separate tests of two fans, in which the fan blades were cleaned and the leading edge recontoured, produced a 0.4 percent average reduction in cruise SFC, from the pre-repair value. Six LPT modules with various operating times (ref. 11) were tested back-to-back with new and refurbished modules, and the average change in cruise SFC contributed by LPT deterioration was also 0.4 percent. Inspection of the LPT modules disclosed the primary cause of the performance deterioration was clearance increases resulting from blade tip-to-shroud rubbing.

Testing for Specific Effects

Examination of the historical data and data from special engine tests and inspections corroborated the fact that a primary cause of engine performance deterioration was increased blade tip clearances throughout the engine. In both the short and long term, the JT9D and CF6 engines exhibit a significant amount of performance deterioration resulting from increased running clearances. To this effect, investigations were initiated to better understand the cause and effect of increased clearances for both engines.

A major cause of increased clearances in the JT9D engine was believed to be flight loads (aerodynamic and inertial) transmitted to the engine during normal aircraft operations (take-off, landing, etc.) as shown in Figure 11. An integrated NASA Structural Analysis (NASTRAN) model of the JT9D/747 installation, Figure 12, developed jointly by the Boeing Commercial Airplane Company (BCAC) and Pratt & Whitney, was used to predict engine structural deflections and fuel consumption increases resulting from various aircraft/engine flight profiles during quasi-steady engine operation (ref. 12). These profiles included representations of the aircraft flight acceptance test and normal revenue service operations. Estimates of the flight load magnitudes that might be expected during the flight profiles were provided by BCAC. In addition, the NASTRAN model was used to account for dynamic effects resulting from aircraft operation during gust encounters and hard landings (ref. 13).

The output of the NASTRAN analysis is in the form of structural deflections of the engine rotors and cases resulting from flight loads on the engine. The process by which these structural deflections were translated to performance losses - as documented in references 12, 13, and 14 - required the establishment of baseline or "hot running" clearances for the particular flight condition being analyzed. These baseline clearances took into account the effect of centrifugal forces and internal and external pressures and temperatures. The next step was to add to these clearances the contribution from manufacturing offset grinds of the seals along with any rub damage produced during the previous flight conditions. The resulting clearances are those that are available to accommodate structural deflections due to thrust and flight loads. Asymmetric rotor/stator deflections were then introduced from the NASTRAN analysis and when the relative closures exceeded the available gap, the extent of rub damage was recorded as circumferential uniform wear of the blade tips, and local wear of the rub strip. The trade-off between blade-tip/rub-strip damage was obtained by using empirically derived abradability factors. The resulting rub damage was then converted to an average clearance increase for each stage of the engine. The final step involved the conversion of these permanent clearance changes to increases in SFC by the use of influence coefficients unique to the JT9D engine. These coefficients relate blade tip clearance increases to performance loss (SFC).

Results of these NASTRAN analyses are shown in Table 4. As indicated, the nacelle aerodynamic (pressure) loads account for 87 percent of the total short-term engine performance deterioration (0.7 percent cruise SFC increase). Inertia loads, which affect the HPT and fan primarily, cause approximately 13 percent of the deterioration. The dynamic loads did not cause any significant changes in the steady loads analysis.

To acquire a better understanding of the effect of flight loads on engine running clearances and the associated performance deterioration, a sequence of analyses (theoretical) and tests (empirical) was developed, (Fig. 13). For the simulated aerodynamic loads test, a specially prepared JT9D engine was instrumented to measure performance, clearances and case thermal gradients and subsequently installed in a high-energy X-ray facility (Fig. 14). This facility was modified with a specially designed loading device that used "belly-bands" around the engine nacelle, connected to cables that were used to apply simulated flight loads to the engine.

Extensive instrumentation (Fig. 15) provided the measurements necessary to assess performance deterioration on a modular basis. X-rays and laser proximity probes were used to measure blade tip and seal clearances. X-rays, both top and bottom, were taken at seven axial positions along the engine, while laser proximity probes were available for nine stages (four circumferential locations per stage). In addition, 400 thermocouples and pressure taps were installed to measure engine case, flange, and cavity air temperature and pressure gradients. These temperature measurements were required to separate thermally induced clearance changes from those caused by externally applied structural loads.

The simulated aerodynamic load test program consisted of three sequences which were preceded and followed by performance calibrations and cold clearance measurements. The first test sequence involved the determination of changes in engine running clearances due to thermal and thrust loads. In the second test sequence, changes in engine static (cold) clearances were established as a function of simulated inlet aerodynamic loads. The third

sequence involved operation of the engine at power while applying simulated inlet aerodynamic loads to determine the combined effect of thrust, thermal, and inlet loads on clearances. Following the tests the engine was disassembled and inspected to determine changes in build measurements so that measured performance changes could be correlated with hardware condition.

Preliminary results from the simulated aerodynamic load test indicate the amount of performance deterioration was approximately 0.9 percent in cruise SFC. This is slightly higher than results from other parts of the Engine Diagnostics Program have indicated. Modular analysis is now in process to better understand this level of deterioration.

For these static tests in the X-ray facility, loads were applied to the engine during the tests based on estimates from a limited amount of flight test data. Thus, the final step in this effort to more fully understand the effects of flight loads on JT9D performance deterioration is a flight test using instrumented engines on a 747 aircraft (Fig. 13). The feasibility of several technical options for a flight test program were considered, as reported in reference 15. For these tests an inboard and outboard engine will be instrumented as shown in Figure 16. The inboard engine instrumentation includes expanded performance measurements, clearance measurements with laser proximity probes in the fan and high-pressure turbine (HPT), and HPT case thermocouples. The outboard engine instrumentation consists of laser proximity probes in the fan in addition to normally installed engine performance measurements. Additional engine preparation for the flight tests includes an analytical build (i.e., documented condition and clearances) of the HPT and restored clearances in the fan at the inboard location and restored fan clearances at the outboard location.

The 747 to be used for the flight tests is the BCAC research aircraft RA001. The primary objective of this test is to measure the actual flight loads (aerodynamic and inertial) encountered during aircraft acceptance tests and normal revenue service operations. To this effect, in-flight aerodynamic loads on the inlets and nacelles will be measured by a series of strategically located pressure probes (252 at the inboard location and 45 at the outboard location). Gravitational and gyroscopic forces on the airplane center-of-gravity, the wing strut intersection, and the engine will be measured using accelerometers and rate gyros. All data will be continuously monitored and recorded during the flight tests and time synchronized for correlation of clearance changes with load conditions.

The testing sequence begins with a performance calibration of the inboard engine in a Pratt & Whitney test cell. Following installation on the aircraft, an installed engine [illegible] calibration of both inboard and outboard engines is to be performed to obtain a baseline to reflect performance changes associated with the various test segments. After this calibration, the flight tests are to begin with a duplication of that portion of the normal aircraft acceptance flight profile that contributes to engine performance deterioration. During the flight test, loads and engine clearance changes will be measured simultaneously. Another installed ground calibration test will be conducted after the acceptance flight to establish the level of engine performance deterioration resulting from the acceptance flight. In the final segment of flight tests, airplane gross weight will be varied [illegible] the effects on engine performance deterioration of take-offs and [illegible] "g" (wind-up turns) maneuvers which might be encountered during airline revenue [illegible] operations. A final installed engine calibration will then be conducted to determine [illegible] engine performance change due to the flight load tests. Following this [illegible] calibration, the inboard test engine will again be calibrated in a Pratt & Whitney [illegible] and then undergo an analytical teardown/inspection to measure clearance [illegible] the correlation of measured flight loads and clearance/performance [illegible] testing is scheduled to occur during October 1980. Results from the flight test and [illegible] aerodynamic load tests in the X-ray facility will be used to refine the NASTRAN structural model. This model should then assist in the development of design criteria for minimization of engine performance deterioration.

HPC and HPT Clearance Investigations

The effects of clearance changes on CF6 performance is being investigated in two areas: the high-pressure turbine (HPT) and the high-pressure compressor (HPC). The predominant mode of CF6 short-term performance deterioration is clearance increases caused by rubs in the HPT, while in the long-term approximately one-half of the performance losses in the HPT and HPC are attributable to increased clearances.

The HPT clearance investigation will be run on a CF6 engine. The engine instrumentation will include pressure and temperature measurements to determine performance, and laser clearanceometer probes will be employed to measure blade tip-to-shroud clearances in the 1st stage of the HPT. The tests will be conducted to produce increasing turbine tip clearances by incurring blade-tip shroud rubs through progressively more severe transient operations. These engine operations include idle, steady state operations at various power levels, accelerations and decelerations to and from takeoff power, rebursts to takeoff power, and a fuel shutoff at takeoff power. These tests will allow the effect of varying HPT clearances on performance to be accurately determined. The results will enable updating of analytical methods and assumptions to enhance future refurbishment or design criteria.

The sensitivity of HPC performance to clearance changes will be determined during tests of a CF6 core engine. Clearance variations, measured by capacitance-type clearanceometers, will be accomplished by flowing cooling air through the compressor rotor internal cavity to adjust thermal growth under steady-state operating conditions. In addition, the

transient clearance behavior of the compressor will be investigated during engine acceleration and deceleration. The test results from this program will provide a better understanding of the degree of performance deterioration associated with tip clearance variations, and will influence future refurbishment and design criteria.

These HPT and HPC clearance investigations are scheduled to occur during the summer of 1980 and results are not available.

PROGRAM RESULTS

The results of the program to date are documented in Figure 17 as a composite curve and labeled "NASA Program Results - Average data (JT9D and CF6)". The assumption used to develop the composite curve was that the rate of deterioration between the data points was linear between repair cycles. This linear assumption was made because the exact shape could not be precisely defined in the NASA engine diagnostic studies (refs. 4, 5, and 6). The upper curve represents the original industry estimates, references 16 and 17 (circa 1974), prior to the NASA Program. Comparing the two, reveals that the original estimate of performance deterioration was high by at least a factor of two. As a part of the NASA Engine Diagnostics program, cost effective feasibility studies were also conducted which - when extrapolated to today's fuel price (approximately 25 cents per liter) - reveal that 80 percent of the [illegible] cruise SFC currently unrestored, after each engine repair/overhaul, is cost-effective to restore. In addition, remedial hardware modifications have been identified which will minimize the short and long term performance deterioration.

CONCLUDING REMARKS

The NASA Engine Diagnostics Program results revealed wide variations in the performance deterioration rates for individual engines in both the JT9D and CF6 engine families. The prime contributing factor to this variation is believed to be the "on-condition maintenance" concept which permits selective repair and interchange of engine modules during engine repair/overhaul. The results presented in this paper are a reasonable representation of the average deterioration characteristics of the JT9D-3A/7/20 and CF6-6D engines. Testing for specific effects is continuing and, therefore, the results are not final. Analysis of these carefully documented engine tests may suggest revisions to some of the findings.

In summary, some of the most important results to date reveal the following:

- o Short term performance deterioration is less than one percent cruise SFC. The causative factor is either flight loads (JT9D) or thermal mismatches (CF6) which result in rubs between blade tips and stationary shrouds.

- o Long term performance deterioration occurs gradually and is about 2.5 to 3.0 percent cruise SFC (including initial short-term deterioration) after 2500 to 3000 flights. The long term losses are associated with more severe rubs, airfoil quality, and distortion.

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TABLE 1. - ENGINE CHARACTERISTICS

Engine	JT9D-7	CF6-6D
Max Power at Sea Level (Dry) Total Airflow Overall Pressure Ratio By-Pass Ratio	202,829N 696 kg/s 22.5 5.1	177,920 N 591 kg/s 24.4 5.72
No. of Compression Stages Fan Low High	 1 3 11	 1 1* 16
No. of Turbine Stages High Low	 2 4	 2 5
No. of Combustion Stages	1	1
Application	Boeing 747, 747SP, 747SR	DC-10-10

*Designated as a Quarter Stage

TABLE 2. - PERFORMANCE DETERIORATION AVERAGE VALUES/STATISTICAL VARIATIONS

Δ% CRUISE SFC

Period	JT9D-3A/7/20			CF6-6D		
	Flights	Avg.	SEE	Flights	Avg.	SEE
Short-Term	1	0.7	±0.3	1	0.8	±0.6
Long-Term	1000	2.0	±0.8	1650	2.3	±1.1
Long-Term	3000	3.0	±0.8	2500	2.4	±1.1

SEE-Standard Error of Estimate (Root-Mean-Square of Deviations About a Fitted Curve)

TABLE [illegible] - SPECIAL ENGINE TEST RESULTS

Period or Module	Engine	Serial No	Flights	Avg Δ Cruise SFC,%	Primary Cause of Deterioration
Short-Term	JT9D CF6	743 507	141 4	0.7 0.9	● Clearance Increase
Long-Term	JT9D CF6 CF6	(1) 380(2) 479	1000 1270 1910	2.0 2.2 3.1	● Clearance Increase ● Airfoil Quality Distortion
Fan (2 Tests)	CF6		1910 3740	0.4 (Avg)	● Airfoil Quality (Leading Edge Bluntness & Dirt)
LPT [illegible]	[illegible]		2180- 7444	0.4 (Avg)	● Clearance Increases

[illegible] (2) Flights Since Last Repair

TABLE 4. - JT9D/747 Propulsion System Structural Analysis

Steady State Results		
Loads	Engine Components Affected	% of Total SFC Loss
Nacelle Aerodynamic	All	87
Inertia		
"G"	HP Turbine	8
Gyro	Fan	5
Dynamic Results		
Wind Gust Encounters	No Significant Change From Steady State	
Revenue Service Landing		

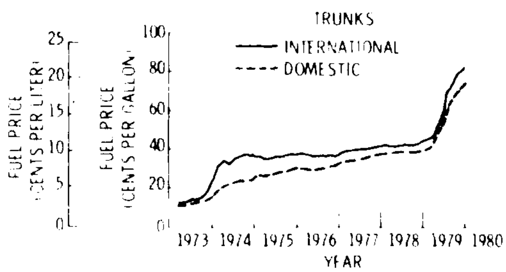

Figure 1. - Fuel cost history. U. S. Airline jet fuel price monthly averages. CAB data.

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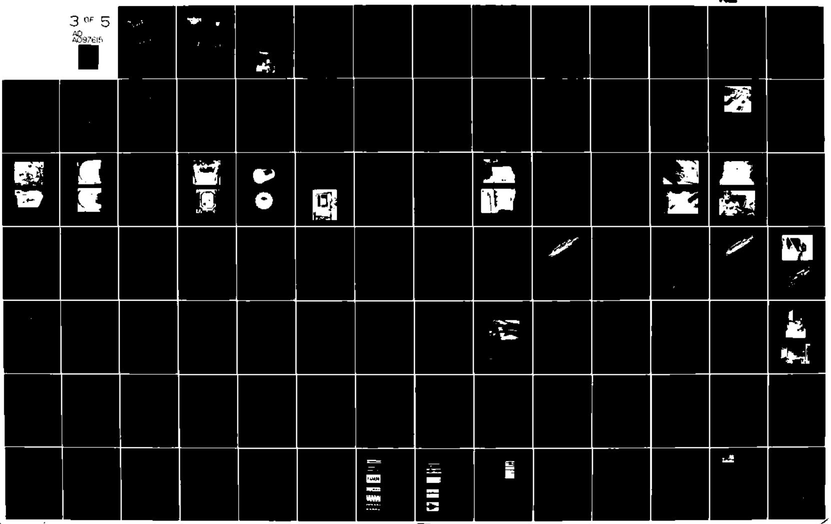

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Figure 2. - The JT9D propulsion package.

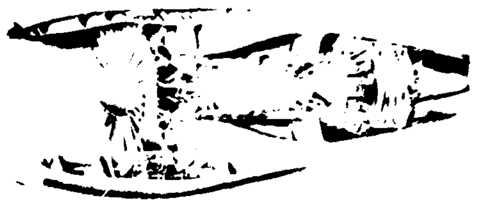

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Figure 3. - The CF6-6 propulsion package.

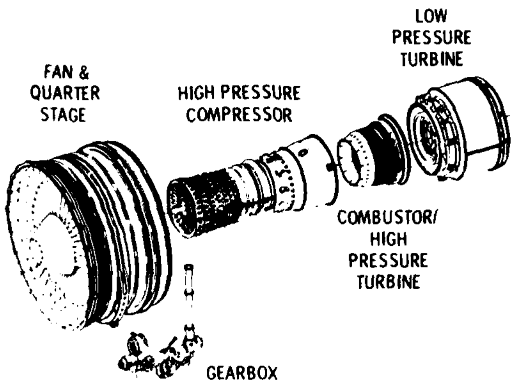


Figure 4. - CF6 engine modular design.

GENERAL ELECTRIC	PRATT & WHITNEY
GE/EVENDALE	P&W ENGINEERING
GE/ONTARIO	P&W SERVICE CENTER
COOPER AIRMOTIVE	BOEING
DOUGLAS AIRCRAFT	DOUGLAS AIRCRAFT
AMERICAN AIRLINES	AMERICAN AIRLINES
CONTINENTAL AIR LINES	NORTHWEST AIRLINES
NATIONAL AIRLINES	PAN AMERICAN WORLD AIRWAYS
UNITED AIR LINES	TRANS WORLD AIRLINES
WESTERN AIR LINES	UNITED AIR LINES
	PACIFIC AIRMOTIVE

Figure 5. - Data sources.

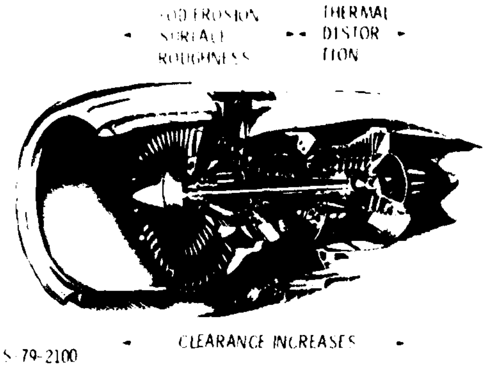


Figure 6. - Contributors to engine performance deterioration.

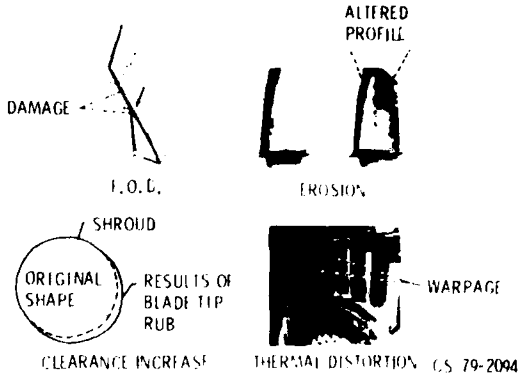


Figure 7. - Examples of engine performance deterioration.

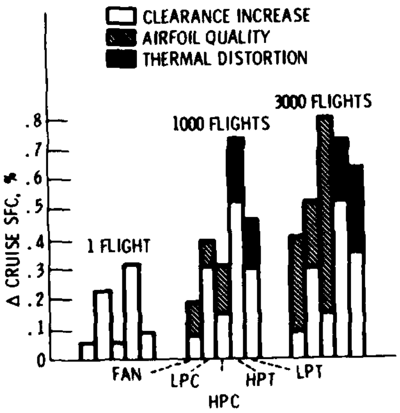


Figure 8. - JT9D-3A/7/20 performance deterioration. Modular contribution.

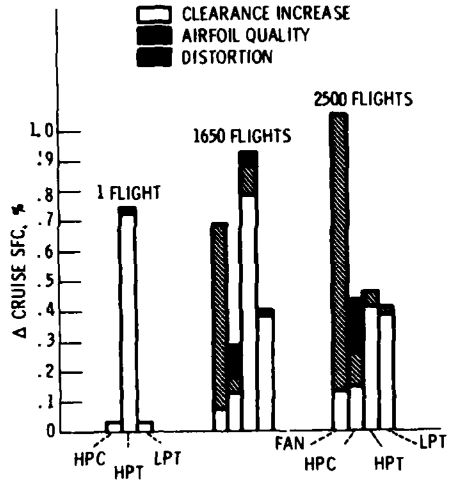


Figure 9. - CF6-6D performance deterioration. Modular contribution.

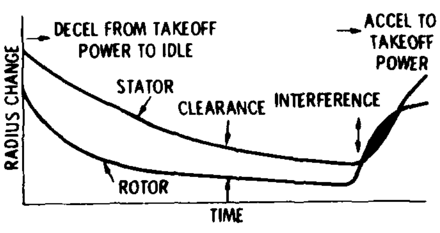


Figure 10. - Hot rotor reburst. HPT clearance.

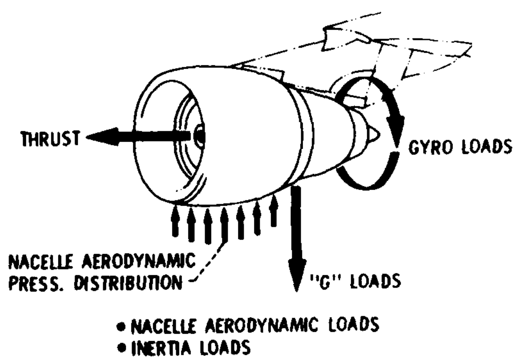


Figure 11. - JT9D external applied loads and reactions.

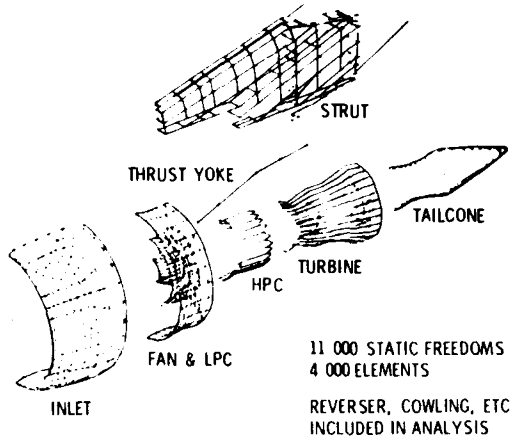


Figure 12. - JT9D/747 propulsion system structural model.

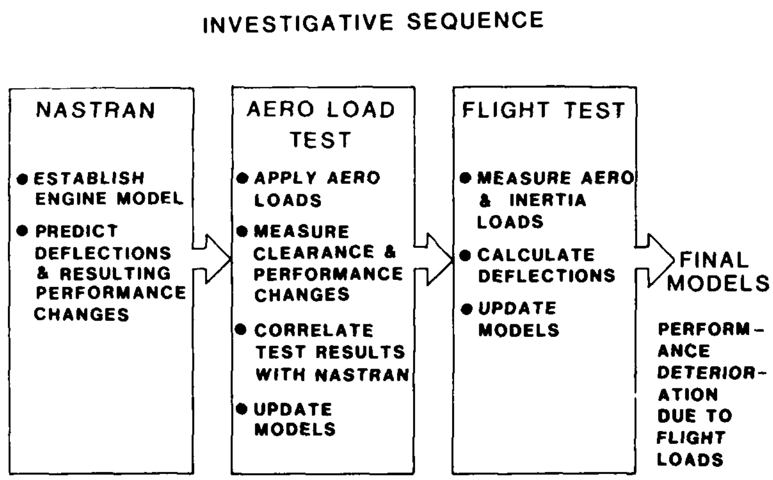


Figure 13. - Effects of flight loads JT9D/747 propulsion system

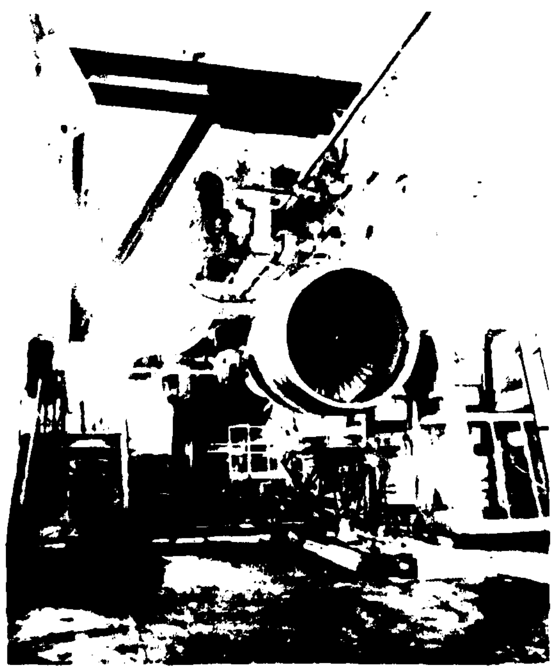


Figure 14. - JT9D engine installed in X-ray facility.

INSTRUMENTATION

CLEARANCE MEASUREMENTS

- X-RAY SYSTEM
- LASER PROXIMITY PROBES - 9 STAGES, 4 PER STAGE FAN, LPC (4TH STAGE) HPC (5, 6, 9, 10, 11 AND 14 STAGES); HPT (1ST STAGE)

PERFORMANCE MEASUREMENTS

- ALL STATIONS T_T, P_T, AND P_S
- N_1, N_2, F_N, W_F

ENGINE CASE THERMALS

- HPC, DIFFUSER, HPT, LPT, TURBINE EXHAUST CASES
- 400 CASE, FLANGE, AIR CAVITY THERMOCOUPLES AND PRESSURES

Figure 15. - JT9D aerodynamic load test.

INSTRUMENTATION

	ENGINE (OUTBOARD)	ENGINE (INBOARD)
BOEING		
INLET (INTERNAL AND EXTERNAL) PRESSURE PROBES	45	252
ACCELEROMETERS	12	12
RATE GYRO	2	2
PRATT & WHITNEY		
EXPANDED ENGINE PERFORMANCE	NO	YES
FAN CLEARANCE PROBES	4	4
HPT CLEARANCE PROBES	--	4
HPT CASE TEMPERATURE T/C	--	20

Figure 16. - JT9D propulsion system flight test.

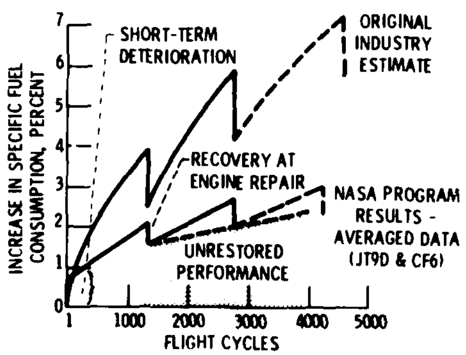


Figure 17. - SFC performance deterioration trend. Typical engine.

DISCUSSION

M.Mihail, Bureau Veritas, Fr

In your lecture you insisted on the tests before installing. Why do you consider those tests so important? Is it a matter of instrumentation, or what?

Author's Reply

What we would like to do in this program is to determine performance deterioration as the engine progresses through its life cycle. We try to get the performance before it was installed on the wing, and as it goes through its life cycle.

Paul Chetail, Air France, Fr

(1) To what extent are the results of the tests on CF6-6 and JT9D applicable to the CF6-50 and the JT9-D-7?

(2) What does the author think of the incidence of the wear of the fans' shrouds due to the rotation of the plane upon take-off on the problem of the initial deterioration of performances?

Author's Reply

(1) The paper presented included the results of our investigation of the JT9D-3A/7/20 engines (Pratt & Whitney). The General Electric engine studied was the CF6-6. General Electric has also conducted a similar study on the CF6-50 engine for NASA. The results of this study should be available in a report to be published in the next few months. Preliminary results indicate the CF-6-50 performance deterioration values are less than the values associated with the CF6-6.

(2) The analyses conducted by Pratt & Whitney on the JT9D and General Electric on the CF6-6 did not indicate any significant fan deterioration except for air foil quality. The flight test of the Boeing 747 research airplane with four (4) laser proximity probes in both the outboard and inboard engines should indicate if any rub strip damage occurs in the fan. This data should be available in about 6 months. The data will indicate in which quadrant of the fan rub strip damage occurred.

CONCEPTION, DEVELOPPEMENT ET CERTIFICATION DU MOTEUR CFM56

Résistance à l'ingestion des corps étrangers

Monsieur R.J. ARIZZI
Ingénieur
SNECMA

77550 MOISSY-CRAMAYEL
FRANCE

SOMMAIRE

L'exposé décrit l'approche théorique, corrélée par des résultats expérimentaux, utilisée pour dimensionner les aubages de tête d'une turbomachine civile, dans le but d'aboutir à un moteur résistant à l'ingestion des corps étrangers. L'objectif poursuivi est double : d'une part satisfaire les règlements de certification internationaux (FAR, JAR-E) et, d'autre part, constituer des marges de tenue mécanique confortables afin de minimiser les coûts de remise en état, donc d'alléger les frais de maintenance des compagnies utilisatrices.

Après une brève description du moteur, des règlements de certification et de l'approche théorique, le reste de l'exposé décrit abondamment l'ensemble des essais réalisés, qui ont été continuellement utilisés pour affiner les résultats théoriques, et qui ont abouti à la définition finale du moteur CFM56.

1 - GENERALITES

Le moteur CFM56 a été étudié et développé par la SNECMA en France et GENERAL ELECTRIC aux USA qui ont travaillé ensemble sous la coordination de CFM INTERNATIONAL. C'est une société appartenant conjointement à la SNECMA et à GE et créée pour la certification, la vente et le support après-vente du moteur auprès de la clientèle.

Le CFM56 est un moteur à fort taux de dilution (6), appartenant à la classe des machines de 20'000 à 27'500 livres de poussée (9200 à 12700 daN). Il est, de ce fait représentatif des moteurs de taille moyenne de la nouvelle génération.

Lorsque ce projet a démarré, au début des années 70, plusieurs objectifs principaux ont été fixés aux deux groupes techniques :

- excellente fiabilité et maintenabilité
- faible consommation de carburant
- faible niveau de bruit et de pollution
- grande robustesse.

Le choix d'une modularité intelligente, d'un cycle thermodynamique approprié, ajouté à un raffinement technologique très poussé a permis d'obtenir un rapport poussée-masse élevé, un faible bruit, une faible pollution et une basse consommation spécifique ainsi qu'une maintenabilité conforme aux désirs des compagnies futures utilisatrices.

La robustesse du moteur a fait l'objet d'essais et de recherches intensifs dans l'optique non seulement de satisfaire les exigences de certification mais également d'obtenir une très grande fiabilité d'exploitation, des faibles coûts de remise en état après incident, ainsi que de faibles durées d'immobilisation.

La robustesse particulièrement élevée, dont doit faire preuve la partie frontale du moteur, pour résister aux impacts de corps étrangers, est un des atouts majeurs du CFM56 dans le domaine de la sécurité des vols.

Le partage de la responsabilité entre GE et SNECMA, dans le développement du moteur, a assigné à cette dernière société la tâche de développer le corps basse pression du CFM56.

2 - DESCRIPTION SOMMAIRE DU MOTEUR

Le moteur CFM56 est une turbo-soufflante dont le taux de dilution est de 6. Le rotor basse pression est formé par un étage de soufflante et trois étages de compresseur basse pression entraînés par une turbine à 4 étages. Le rotor haute pression possède 9 étages de pression (avec stators variables) entraînés par un seul étage de turbine refroidie par air. Les deux rotors sont supportés par 5 paliers, dont un roulement différentiel, appuyés sur deux structures qui sont respectivement : le carter intermédiaire, supportant 3 roulements, et le carter d'échappement qui porte le roulement arrière.

La poussée du moteur est reprise à hauteur du carter intermédiaire et le moment autour de l'axe moteur peut être repris soit par l'une ou l'autre des deux structures.

Les aubes de la soufflante en titane ont été étudiées pour obtenir, comme dit plus haut, des hautes performances et une bonne marge au pompage, alliées à une grande robustesse.

Ce difficile compromis a été atteint en choisissant une technologie particulière qui est, à notre connaissance, adoptée pour la première fois au monde sur un moteur civil, à savoir les talons extérieurs ou périphériques.

Le carter intermédiaire, également étudié par la SNECMA, est la principale structure du moteur qui assure une rigidité suffisante pour conserver le contrôle des jeux rotor/stator sous des charges élevées, telles que : effort de poussée, charges aérodynamiques, etc. Cette structure, ainsi d'ailleurs que le rotor BP, a été conçue pour résister aux charges extrêmes dues à la perte de 2,5 aubes de soufflante et ou à un blocage du rotor BP.

Le compresseur HP, qui comporte 4 étages de stator à géométrie variable, est composé d'un rotor à tambour dont les éléments sont soudés entr'eux. Le carter du compresseur est constitué de demi coquilles, qui peuvent être démontées séparément, afin de faciliter l'accès aux organes internes.

La chambre de combustion est très courte, de forme annulaire et à injection à basse pression. Cette chambre, qui a favorisé la réalisation d'un moteur très compact, a un degré de pollution nettement inférieur aux niveaux limites des règlements en vigueur. La turbine HP, à un seul étage, a les aubes mobiles et fixes refroidies par air. Les jeux entre rotor et stator sont maitrisés par des écoulements d'air à température variable.

La turbine BP, à 4 étages, comporte des disques sans aucun perçage dans les sections travaillantes et un carter monobloc refroidi par de l'air grâce à des rampes circulaires, ce qui permet également de maitriser les jeux entre rotor et stator.

Comme dernière information, nous signalons que parmi les moteurs à haut taux de dilution actuellement en exploitation, le CFM56 est celui qui comporte le moins de pièces, par exemple 34% de moins que le CF6-50.

3 - RAPPEL DES REGLEMENTS DE CERTIFICATION

Depuis le démarrage du projet, il avait été décidé par CFM INTERNATIONAL que les deux certificats de type, américain et français, devaient être obtenus.

Cela a abouti à l'addition de deux règlements de certification : la FAR 33 pour la FAA et le JAR-E pour la DGAC ; tout le programme de développement fut donc conçu pour satisfaire les deux règlements en même temps, y compris l'ingestion des corps étrangers.

Le programme d'ingestion a été partagé en trois parties, qui étaient fonction de l'exigence attachée à chacune d'elles :

1) Un moteur affecté
 - Exigence : pas de risque pour l'avion, extinction du moteur admise
 - Ingestion des corps suivants :
   - une aube de soufflante
   - un oiseau de 4 lb (1,8 kg)
   - une chape de pneumatique

2) Plus d'un moteur affecté
 - Exigence : le moteur doit être capable de fournir une poussée résiduelle égale à 75% de la poussée du décollage
 - Ingestion des corps suivants :
   - sept oiseaux moyens (1,5 lb = 0,68 kg) en moins d'une seconde (le JAR-E n'en exige que cinq)
   - 4,8 oz (136,1 g) de gravier
   - 33,2 oz (941,2) de sable

3) Plus d'un moteur affecté
 - Exigence : aucune perte de puissance
 - Ingestion des corps suivants :
   - 25 grêlons de 2 in (51mm) de diamètre
   - 25 grêlons de 1 in (25,4 mm) de diamètre
     la volée de 50 grêlons en moins de 5 secondes
   - Glace provenant de l'entrée : pas de dégivrage pendant 30 secondes
   - Eau : 4% en masse du débit d'air.

Ainsi qu'on peut en juger, la quantité des corps étrangers devant être ingérés était très grande, à tel point que dans le passé, jamais aucun moteur de cette taille n'en avait avalé autant, avant d'obtenir sa certification.

Le risque d'ingestion encouru par un moteur en vol est néanmoins très réel et ne peut être ignoré. C'est pour cette raison que CFM INTERNATIONAL, et la SNECMA en particulier, décidèrent d'étendre le programme d'ingestion en y ajoutant des essais de super sévérité, tels que : l'ingestion de barreaux de glace, de plusieurs oiseaux de 4 lb, et d'autres corps étrangers.

L'objectif final était d'apporter la preuve d'une marge extrêmement confortable vis-à-vis des dommages primaires et secondaires, afin de faire non seulement la démonstration légale de la certification mais aussi, en limitant les dégats aux pièces impactées, d'une grande fiabilité, d'un faible coût de remise en état, d'une immobilisation réduite et enfin d'une très grande sécurité d'exploitation.

4 - APPROCHE MATHEMATIQUE

Le problème consistait à simuler numériquement, à l'aide du programme d'analyse dynamique tridimensionnel non linéaire PAM NEP-D, l'effet d'un impact d'un corps étranger sur une aube de compresseur en fonctionnement.

L'examen des exigences de certification à satisfaire, ainsi qu'une campagne préliminaire de tirs d'oiseaux sur des secteurs d'aubes statiques, ont permis de déterminer que le cas le plus sévère était représenté par l'ingestion de l'oiseau moyen de 1,5 lb (0,68 kg); il y a une exigence de "survie" de 75% de la poussée du moteur ; de plus, pour des raisons commerciales évidentes CFM INTERNATIONAL avait fixé un objectif bien plus ambitieux : pas de fragmentation des aubes. C'est donc pour ce cas particulier que l'approche mathématique a été utilisée.

Trois zones d'impact ont été choisies à priori : le pied de l'aube, le milieu de la hauteur et le sommet, au voisinage du talon extérieur.

Les conditions de fonctionnement du moteur lors de l'ingestion sont déterminées par la vitesse de décollage : le régime de rotation de la soufflante et la vitesse d'ingestion maximaux sont donc connus.

Ces valeurs ont été maintenues constantes pendant toute la campagne d'essais et de calculs.

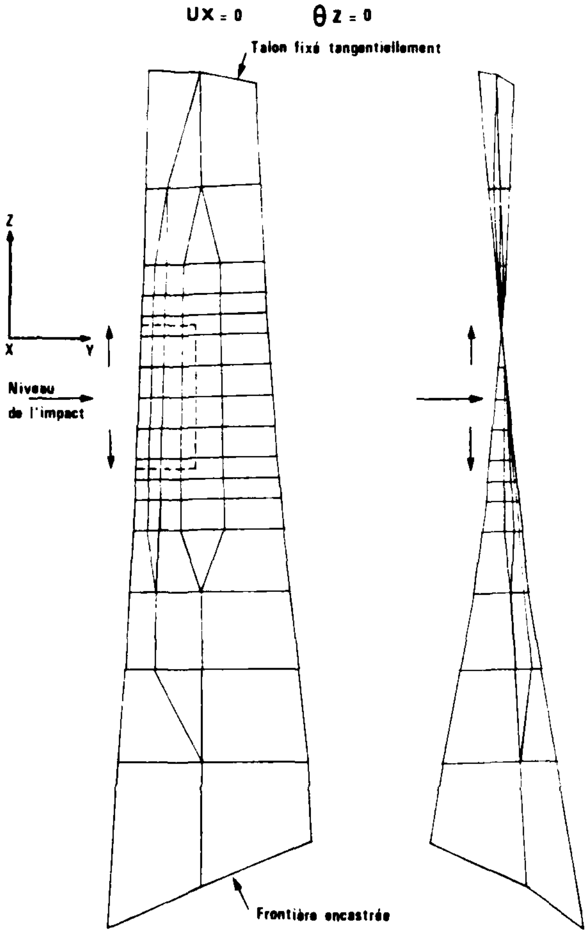
Sont également fixées à l'avance : les conditions aux limites de l'aube : encastrée au pied, appuyée au sommet et les caractéristiques mécaniques de la matière : limite élastique, allongement élastique, coefficient de Poisson, module élastique ou de Young.

Les propriétés plastiques de la matière ont été déduites d'une courbe de traction classique.

Le programme de calcul permettait la prise en compte des charges extérieures : champ centrifuge, choc de l'oiseau et des charges induites par les déformations (efforts d'inertie).

Le calcul du comportement dynamique de l'aube a été fait par éléments de plaque mince dont le maillage (ou discretisation)est représenté sur la planche 1, pour le cas particulier d'un impact à mi-hauteur de la pale. Naturellement la finesse du maillage est augmentée dans la zone de choc et est donc variable en fonction de la hauteur.

Planche 1 _ GEOMETRIE DE L'AUBE ET MAILLAGE.



Le calcul des efforts a été réalisé par un programme manuel simple décrit ci-dessous :

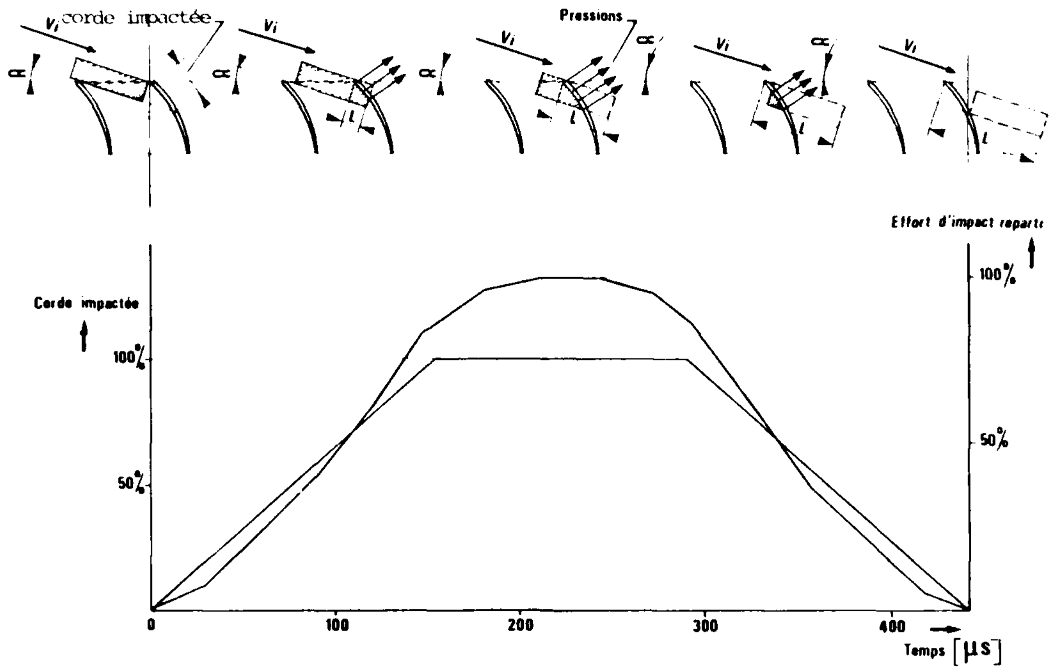
. Détermination de la tranche élémentaire d'oiseau découpée par un canal inter-aubes et de son vecteur vitesse (direction, intensité)

. Etablissement de la caractéristique de l'effort d'impact en fonction du temps par étude du cheminement de la tranche dans le canal inter-aubes. (voir planche 2)

. Cet effort est réparti sur la surface projetée de la tranche d'oiseau, au lieu de le considérer concentré au centre de l'impact, comme certains auteurs le font lors des études préliminaires. L'augmentation de l'effort due à l'augmentation d'incidence résultant de la déformation du bord d'attaque n'a pas été considérée.

. La vitesse de la tranche d'oiseau est considérée constante pendant la trajectoire dans le canal.

Planche 2 _ VALEUR DES EFFORTS D'IMPACT REPARTIS ET DE LA CORDE IMPACTEE EN FONCTION DU TEMPS_



. La détermination de la caractéristique de l'effort est faite par points dont le pas est de 2% du temps total mis par la tranche pour quitter l'aube. A titre d'information, le temps total de l'exemple donné en planche 2 est de 0,45 ms, le temps pour atteindre l'effort maximum 0,23 ms.

Lorsque la caractéristique des efforts en fonction du temps est établie, le programme PAM NEP-D effectue le calcul par éléments de plaque mince sur l'ensemble de la pale et détermine la déformée globale à des temps choisis.

Comme dit plus haut, le calcul est fait dans le domaine élastoplastique ; chaque point nodal des éléments a 6 degrés de liberté : 3 translations et 3 rotations. Il est ainsi possible de connaître les zones à fortes contraintes particulièrement en traction et flexion.

De même, un tracé automatique des zones à iso allongement permet de connaitre les zones où les déchirures vont s'initier.

La planche 3 montre le tracé de la déformée de l'aube à différents temps du phénomène. Il y apparait nettement comment les déformations et les ébranlements s'initient et se propagent le long de la pale :

. au temps t = 0,25 ms qui correspond au temps d'établissement de l'effort maximum seule une poche est formée au point d'impact
. au temps t = 0,45 ms, temps total d'impact, l'ébranlement se propage vers le haut et vers le bas
. au temps t = 0,6 ms, le sommet de l'aube amorce une rotation autour de l'axe X, qui est encore plus grand au temps t = 0,76 ms. Cette rotation favorise naturellement les chevauchements des talons supérieurs.

Dès les premiers calculs, il avait été déterminé que, dans la zone du pied, la limite de rupture du matériaux n'était jamais atteinte (calcul pied encastré).

Cela fut confirmé pendant la phase expérimentale où aucune déchirure, amorce de crique ou fragmentation n'a été rencontrée.

Lorsqu'on aura précisé que le programme peut calculer les contraintes sur le profil, intrados et extrados, on aura une idée globale de la puissance de calcul de notre modèle mathématique.

La portée d'un tel outil est néanmoins limitée si il ne repose pas sur des bases expérimentales statistiques très solides. C'est pourquoi, en parallèle, une activité d'essai sans précédent a été décidée par CFM INTERNATIONAL dans le cadre du programme de développement du CFM56.

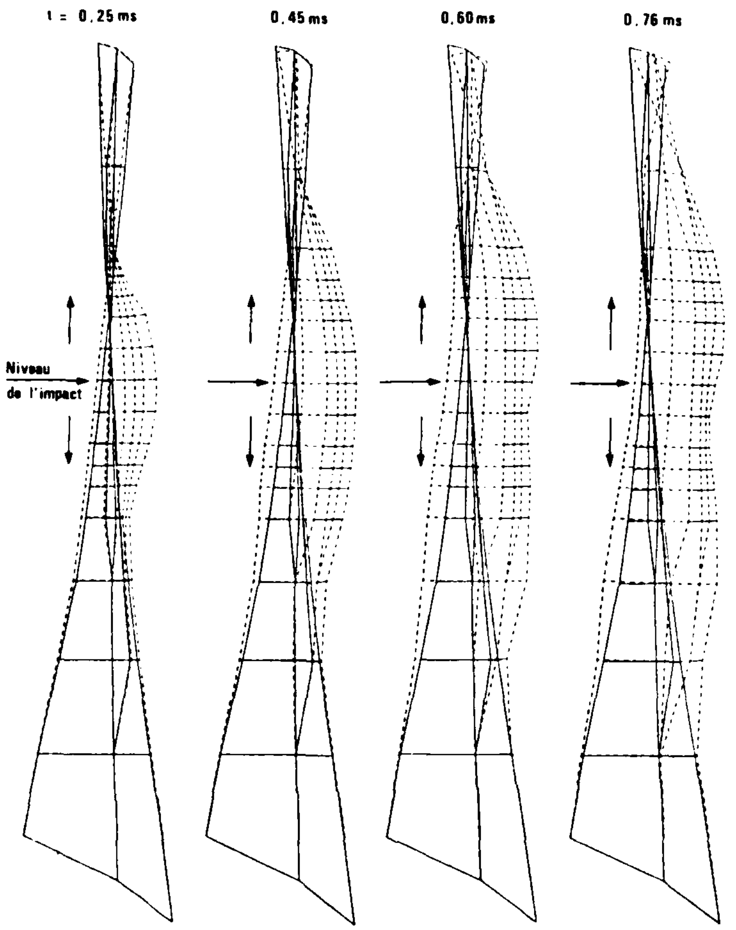
5 - APPROCHE EXPERIMENTALE

5-1 Objectifs - Historique

L'activité expérimentale d'ingestion des corps étrangers a été définie et appliquée par la SNECMA au Centre d'Essais de Villaroche, près de Paris.

Deux objectifs principaux étaient assignés à cette activité :
. obtenir des résultats d'essais nombreux et exploitables, lors des ingestions d'oiseaux moyens, pour alimenter et affiner en permanence l'approche analytique.

Planche 3 _ AUBE DEFORMEE



. Effectuer un grand nombre d'ingestions d'autres corps étrangers afin d'éprouver d'une façon réaliste les versions successives des aubes de soufflante, définies par le calcul, en vue d'obtenir la certification finale du moteur CFM56..

Une partie des essais de développement a été réalisée sur des montages partiels, au lieu d'utiliser des moteurs complets.

Les bancs d'essais partiels utilisés sont des fosses à vide qui ont l'avantage de ne demander qu'une puissance minime ainsi qu'un personnel très réduit pour leur mise en oeuvre.

L'expérience a montré d'ailleurs que les efforts aérodynamiques, qui ne sont pas simulés dans le vide, ne changent en rien les résultats ; ceci est dû principalement au caractère très rapide des phénomènes étudiés : temps très brefs, énergies considérables.

La période d'essai s'est étendue sur 4 années (de 1975 à 1978) ; environ 350 aubes de soufflante de différentes versions ont été détruites (soit l'équivalent de 8 aubages); ainsi qu'un compresseur BP complet.

Tous les essais ont été réalisés avec des poulets achetés chez un fermier des environs du centre d'essais: Ils étaient nourris de blé, de maïs et d'orge (pas d'hormones) et vivaient en liberté dans les champs.

Ils ont été tués par asphyxie avec du gaz carbonique (CO_2) de façon à conserver dans leur organisme toutes les matières, y compris le sang.

Il est donc considéré que la dureté de leurs os ainsi que la densité moyenne ont représenté d'une manière réaliste les caractéristiques des oiseaux réels.

5-2 Essais partiels

Les débuts de l'approche théorique avaient permis de déterminer les conditions dans lesquelles un impact d'oiseau pouvait se produire. Pour vérifier les premières conclusions, un ensemble de tirs, sur des aubes de soufflante immobiles, a été réalisé en reproduisant les conditions d'impact : vitesse de choc relative à l'aube, angle d'impact par rapport à la corde, conditions aux limites (pied et talon extérieur).

L'absence de champ centrifuge était évidemment importante, mais le but prioritaire était d'apporter des données expérimentales au calcul et de faire des prévisions de tenue en fonctionnement.

Les essais suivants ont été réalisés :

. 6 tirs d'oiseaux moyens (1,5 lb) à la vitesse simulée de décollage
. 7 tirs de barreaux de glace (1 x 4 x 6 inch) à la vitesse simulée d'aspiration dans la manche
. 3 tirs d'oiseaux lourds (4 lb) à la vitesse simulée de croisière.

Le programme de tirs sur soufflante en rotation dans les fosses a été ensuite mis en route. Le calcul, dans le champ centrifuge, avait prévu des fragmentations sur la même version d'aubes que celle utilisée pour les essais statiques ; c'est ce qui a été effectivement confirmé. C'est alors qu'une intense activité d'itération a été programmée entre le calcul et les essais afin d'aboutir à la configuration définitive de l'aubage capable de résister, sans fragmenter, à l'ingestion des oiseaux de 1,5 lb.

La plus grande partie de la campagne d'essai a été menée en utilisant uniquement l'étage d'aubes de soufflante, car elles sont considérées comme étant les pièces moteur qui doivent dissiper toute l'énergie de l'impact.

Quelques essais d'ingestion ont été effectués en utilisant la soufflante et le compresseur BP mais, comme cela était attendu, les modifications des aubages de ce dernier ont été très limitées.

L'ensemble de l'activité ingestion en fosse a résulté en un nombre impressionnant de tirs de projectiles de toutes tailles et sortes ; ceci est résumé dans le tableau ci-dessous :

Type de projectile	Dimensions Masse lb/kg	Vitesse d'ingestion m/s	Nombre de tirs
Oiseaux moyens	1,5/0,68	Décollage	41
Oiseaux lourds	4/1,8	Croisière	20
Barreaux de glace	1x4x6 in. 1x3x16 in.	Aspirés	7 1
Grêlons	Ø 1 in et 2 in	Croisière	7
Chape de pneumatique	10x10x0,5 in 4/1,8	Aspiré	7
Aube de soufflante (titane)	2,5 kg	libérée à la vitesse maximale	4

Les tirs, en particulier ceux d'oiseaux moyens, ont balayé toute la hauteur des aubes de soufflante, depuis le pied jusqu'au sommet, sous le talon extérieur.

Les poulets utilisés étaient empaquetés dans un support en plastique de façon à obtenir l'effet d'ingestion le plus sévère et aussi pour uniformiser au maximum les conditions d'impact.

Comme il a été dit auparavant, 550 aubes de soufflante ont été utilisées pour mener à bien la campagne d'essai.

Il a été ainsi possible de déterminer que les dégradations résultant des impacts étaient très reproductibles et directement liées aux paramètres des tirs : masse et vitesse des oiseaux, hauteur d'impact, géométrie des profils d'aubes, etc. ; les lois mécaniques ont pu ainsi être définies et utilisées dans le modèle mathématique, d'une façon continue.

Le résultat technologique final a été en définitive très satisfaisant.

5-3 Essais sur moteur

Les essais sur moteur complet ont été réalisés dans un banc à ciel ouvert qui a été spécialement construit au CEPr de Saclay en France, pour les tirs d'ingestion et de rétention de l'aube de soufflante.

Comme conséquence logique de l'importante campagne d'essais partiels dans les fosses à vide, les essais sur moteur ont confirmé que la définition technologique choisie pour le moteur CFM56 était capable de satisfaire malgré leur sévérité, les exigences de certification et celles des futures Compagnies utilisatrices.

C'est pour cette raison que seuls les essais de certification ont été effectués sur moteur complet :
- oiseaux moyens : après la volée de 7 oiseaux, le moteur a retrouvé 98% de sa poussée initiale du décollage pendant une minute ; puis une endurance de 20 minutes à 75% de la poussée a été accomplie ; pas de chevauchement des talons, une aube légèrement endommagée au bord d'attaque.
- 50 grêlons en une volée de 2,5 secondes : pas de dommage, pas de perte de puissance.
- oiseau lourd dans le flux primaire (donc au pied de l'aube de soufflante) : pas de fragmentation, pas d'incendie.
- perte d'une aube de soufflante (ingestion et rétention simultanées) : balourd légèrement supérieur à celui d'une aube, les dommages sur les autres aubes de l'étage ont été limités à des entailles et des déchirures locales ; pas d'incendie et arrêt moteur normal.

Les certificats de type FAA et DGAC ont été délivrés à CFM INTERNATIONAL le 8 Novembre 1979 à PARIS.

6 - CONCLUSION

Il apparait de ce qui précède qu'à CFM INTERNATIONAL et à la SNECMA en particulier, nous avons été toujours conscients des risques encourus par les moteurs modernes du fait de l'ingestion de corps étrangers.

Les Autorités de Certification ont aussi pris conscience de ces risques et ont imposé aux constructeurs des règles très sévères.

Nous avons donc considéré qu'il était nécessaire de satisfaire à ces règles, pour des raisons de sécurité des vols, et d'aller au delà pour satisfaire les besoins économiques des Compagnies exploitantes.

Les essais et les études qui ont été nécessaires pour atteindre ces objectifs nous ont coûté beaucoup d'argent et de temps, mais nous pouvons maintenant affirmer que le moteur CFM56 a été étudié et essayé sans aucune sorte de bienveillance et plus sévèrement qu'aucun autre moteur civil précédent.

On peut ainsi constater que le CFM56 a atteint son dernier objectif : la robustesse, qui est synonyme de sécurité, de fiabilité, de maintenance rapide, de faible coût, de régularité d'exploitation.

Le CFM56 est maintenant une réalité et il est prêt pour une longue et sûre carrière.

DERIVATION AND CORRELATION OF ACCELERATED MISSION ENDURANCE TESTING

By

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SUMMARY

The Accelerated Mission Test has undergone continuous refinement in its application to the F100 engine program. The details of the test cycle and the configuration of the engine must be rigorously matched to detect service oriented durability problems ahead of the field. The testing can further be tailored to a specific distress mode or substantiation of a specific package of engineering changes. But certain service problems cannot be adequately detected by mission testing. Methodical analytical teardown inspections of "Lead the Force" service engines are an integral part of developing successful maintenance techniques in managing a military engine program as it grows to maturity.

ABSTRACT

An approach to defining the Accelerated Mission Test (AMT) program as an integral part of the overall engine development process will be discussed. This includes not only the initial cycle derivation but also the necessary revisions to the AMT cycles in order to more accurately predict and detect certain failure modes.

The validity of the test results must also be established. A comparison of AMT engine hardware, and operational engine hardware having equivalent cyclic history, is used for this purpose. As an illustration of the benefits of AMT engine testing, data on several gas turbine engines are presented. Also included in the discussion are the various types of failure modes that are currently not detectable in the AMT type program.

DISCUSSION

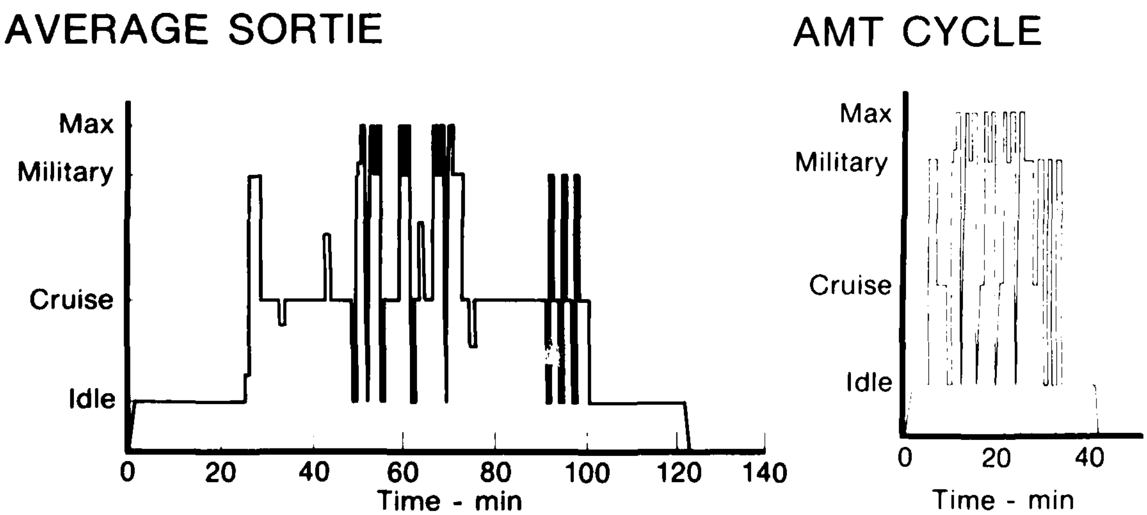
During the typical development phases of a military engine the configuration is evolving to meet performance and operability objectives. The statistical base is small and uncertainties as to how the engine will be used in service exist. With initial production there is a transition to hard tooling and the beginning of a statistical base. As the squadron becomes operational, engine usage evolves with tactics, interface characteristics with the aircraft; and the maintenance shop tooling and manuals are better defined. The overhaul cycle introduces repairs, and worn parts bring new resonances into the running range. The now large statistical base allows the improbable combination of events to occur.

Each of the above phases of an engine's life drives specific types of durability problems. High cycle fatigue and stress rupture modes typically can be detected early in the development phases. The hot section must be tuned to the evolving usage of the engine in service. Infant low cycle fatigue modes and wear associated with variable geometry occur as the lead engines approach their first trip to depot. Wear oriented modes, low cycle fatigue and the statistically remote problems typically occur as the engine maintenance characteristics approach their steady state value.

Accelerated mission testing can be used to detect many of the early service oriented durability problems in advance of the field, but this technique must be expanded to include "Lead the Force" field engines and data from the depot if the high time events are to be controlled with a practical, cost effective maintenance concept tailored to the engine's specific durability characteristics.

DEVELOPING ACCELERATED MISSION TEST CYCLES

The first ingredient in deriving a mission test cycle is to ensure that it contains all the damaging events in service that drive the particular failure modes to be detected. Those relatively nondamaging events are dropped out, providing the acceleration factor.



The AMT cycle is obviously only as good as the quantification of the "average sortie" in service. This step is described in more detail in the references but involves pilot interviews and the automatic recording of engine parameters over a statistically significant number of sorties at each operational base. In practice it takes more than one AMT cycle to simulate adequately the sortie mix of a typical military engine.

The idle time at the end of the AMT cycle must be sufficient to cool the disk bores to simulate an overnight shutdown. The engine is then pulled to cut off, and motored to cool the flow path down to ambient — ensuring the cold engine full thermal strain range on restart.

A facility will have to be used to simulate inlet pressure and temperature if the flight points in the average sortie are significantly different in (a) turbine inlet temperature, or (b) engine inlet temperature/pressure than can be simulated in an ambient test stand. But the test becomes most complex as the airflow transients of modern military engines are a good deal faster than most facilities can handle. It is these very transients that drive the thermal fatigue problems that the AMT approach is trying to detect.

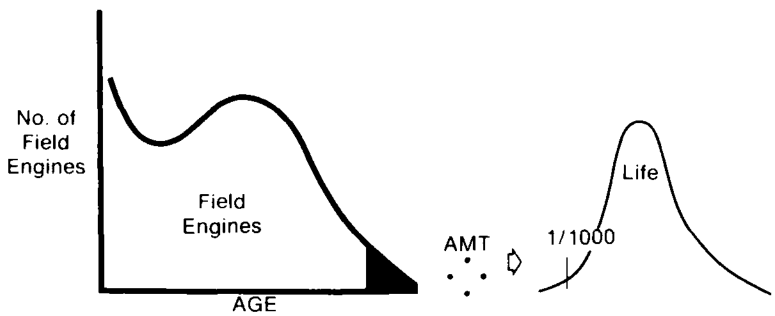
The acceleration rates from idle to military power and above drive the thermal gradients on the cooled parts in the turbine. To ensure a damage rate comparable to the field, not only the number of these full throttle transients but the acceleration rate must be matched to service usage. The hold times at idle establish the bore-to-rim thermal gradients that affect LCF life of the hot disks in the engine, and this also must be carefully patterned after the field.

If the purpose of a particular AMT is *only* to determine the hot section characteristics of a particular configuration or usage, the idle bore cooldown and the longer idle dwells can be dropped from the test as well as replacing the augmented time with military power. As a cooled turbine vane maximum thermal gradient usually occurs above idle, even the shutdown and motor-to-cool on the starters can be eliminated in the event a given vane configuration (only) is to be evaluated.

Although the above flexibility can sometimes be useful, the majority of the AMT engines will have a broader objective involving both Engineering Change substantiation and the driving of prime reliable parts out to high time for detecting service problems.

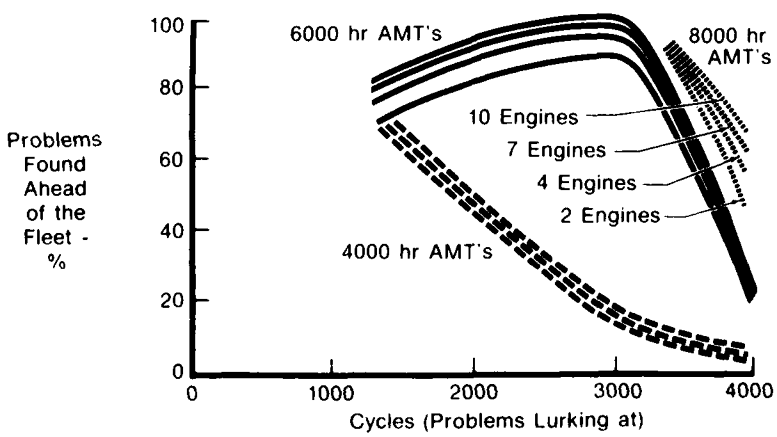
A last ingredient is to purposefully unbalance the rotors to the field limit and intersperse rpm stairstep cycles throughout the testing. The unbalance acts like a hammer on minor resonances in the static parts of the engine to accelerate any normal wear that might occur over a longer time in the field. A final stairstep cycle at 2000 hr accumulates 10^7 cycles (3E on low rotor) such that if wear has allowed a minor resonance to enter the running range the HCF cracks can be detected on teardown.

Finally the AMT engine must be run far enough ahead of the field to completely traverse the probable failure mode distributions.



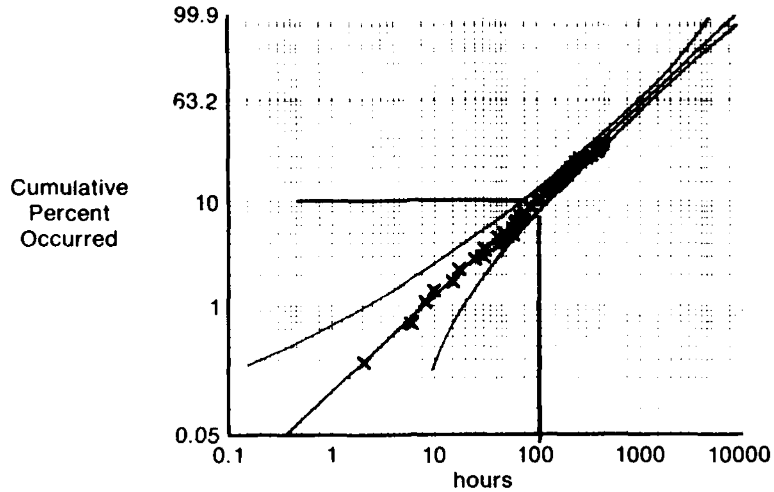
AMT engines must penetrate deeply into the failure distribution to catch the 1/1000 failures modes before the top 10% fleet gets there

With its larger statistical base the service engines will encounter the distress as they penetrate the leading edge of these failure distributions. Two AMT engines must be run to 10,000 cycles each to confidently clear the field for any chronic distress modes up to 3000 cycles.

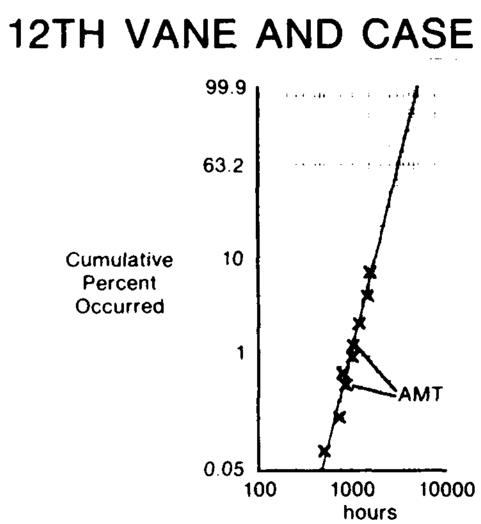


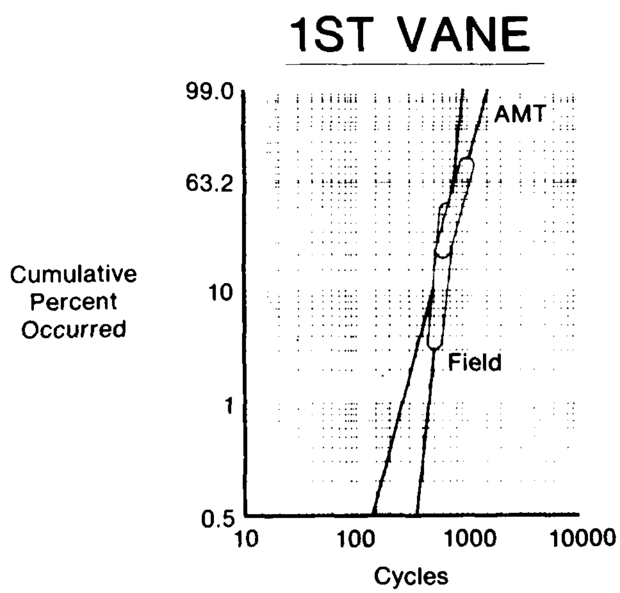
CORRELATING AMT DURABILITY CHARACTERISTICS WITH THE FIELD

The Weibull analysis is a statistical technique using a few incidents and an existing population of successes to predict the shape and location of an upcoming failure distribution. Bearing manufacturers have long used this technique to project the life for a 10% failure rate.



If data from two or more different failure modes are plotted together the result is nonlinear. In this manner AMT durability characteristics can be compared to field data as it becomes available.



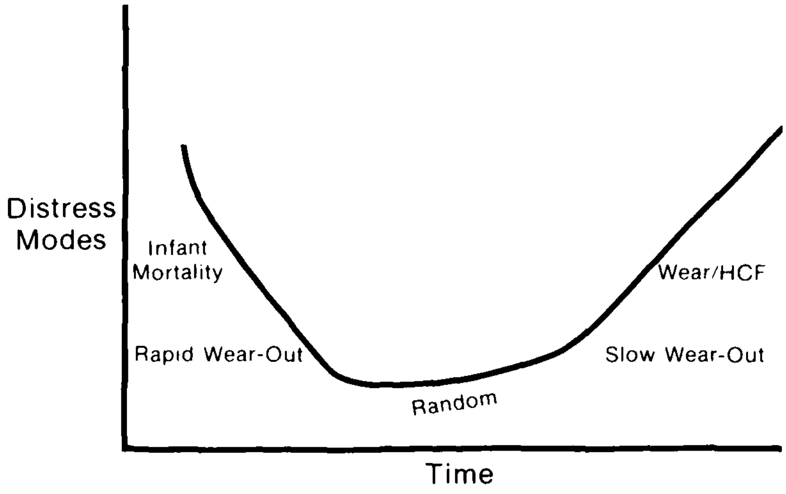


Typically AMT hot section distress, infant LCF, and variable geometry wear data correlate with statistical significance against the service experience. *Those wear modes driven synthetically by unbalanced rotors on* the AMT yield data displaced to the left (lower life than the field). Conversely the exhaust nozzle wear modes influenced by the aircraft's flow field loads appear at lower hours in the field data. Some of these modes (buffeting, axisymmetrical loads) never appear on AMT engines. Gearbox wear modes cannot be confidently detected during AMT even though the horsepower extraction for aircraft accessories is simulated. Some, but not all of the control malfunctions can be detected by running AMTs with the hot fuel ramps that occur in flight.

HIGH TIME DISTRES^ MODES

As the higher time service engines wear and are repaired and refurbished through the depot, the classic far side of the bathtub curve is reached.

The character of the engine itself begins to change. Continuing to run AMT engines out in cycles and maintaining those engines in a development shop loses its credibility.



Two approaches have been taken to attempt to penetrate this region realistically. A high time "Lead the Force" engine was brought through depot for refurbishment and documented extensively. It came in to run as an AMT for a second maintenance interval, returning in two months to depot having been artificially "aged" five years. A second analytical documentation took place before the engine returned to service as a "super" lead the force engine.

A second approach was to take another high time service engine into the program for repair substantiation. This engine is run on mission cycles — duplicating the cruise and idle time as well as typical sorties in the field. Although this engine does not move out as fast as our AMT engines, it still ages far faster than does the field. As each module comes up on its next maintenance interval it is sent to depot for the work, picking up new repairs as well as the necessary inspections and refurbishments. In this fashion, the engine can approximate the characteristics of the natural aging process that will occur later in the service engine. This approach also develops the depot manuals.

SUMMARY

1. Derivation of the details of the AMT test cycles must be done rigorously and updated as service engine usage evolves.

2. Certain important failure modes cannot be addressed adequately in accelerated mission testing — this becomes more critical as the service engines age.

3. Useful durability information can still be captured by applying AMT techniques to high time service engines.

REFERENCES

McDonnell, Bernard J., "Accelerated Mission Test — A Vital Reliability Tool," AGARD Propulsion and Energetics Panel, The Hague, Netherlands, 31 March 1977

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ADD to page 19-4

DISCUSSION

J.Fresco, Turbomeca, Fr

(1) How long does the AMT cycle last at maximum power?

(2) About the counter: What are the threshold in power or spur with regard to idle and maximum?

Author's Reply

(1) Typical AMT matched to F15 aircraft usage accumulates 40 hours time at maximum augmentation in a 1,000 Engine Flight Hour test. The F16 version of this test would accumulate about half this exposure.

(2) The gates in the counter are just above idle, just below military power. Gas generator turbine temperature is held at military setting as power is increased into augmentation to maximum power. Thus these gates miss the smaller transients of the throttle . . . . a compromise to allow the practical recording of this data into the scheduled maintenance of this engine.

FREE-JET TESTING OF POWERPLANTS FOR AIRCRAFT AND MISSILES

- by -

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SUMMARY

The paper describes the free-jet test facilities available at NGTE for testing complete aircraft propulsion systems, that is the air intake, engine and exhaust system, at conditions reproducing those encountered during flight at altitude. Supersonic and subsonic flight conditions can be simulated, both steady state and transient, the latter aspect including the effects of time-variant changes in aircraft flight speed, attitude and engine power.

The scope of free-jet testing is reviewed and compared with what can be achieved using direct-connect facilities.

The paper concludes with a description of subsonic free-jet tests made under the extreme conditions encountered in an icing cloud to determine the effectiveness of intake and engine anti-icing equipment and the ability of the powerplant to operate satisfactorily following the shedding of ice that may have accreted on the inlet duct surfaces.

1. INTRODUCTION

The three main elements of a jet aircraft propulsion system - the intake, the engine and the propelling nozzle - interact closely with each other so that the overall behaviour of the system can only be assessed when all the interacting elements are present. This is especially so in the case of supersonic powerplants where correct matching of the intake, engine and nozzle is vital for reliable operation and optimum performance.

Free-jet testing enables intake/engine interactions and compatibility to be studied over a range of conditions including flight speed, altitude and engine speed. The effects of aircraft manoeuvre transients can also be examined. Methods have been developed at NGTE to enable tests to be made at subsonic, transonic or supersonic flight conditions and various aerodynamic techniques such as spill air diffusion, transonic free-jet testing using slotted blowing nozzles and airframe boundary layer simulation have been developed to extend test plant capacity and make the test representative of free flight.

A specialized form of free-jet test which has assumed considerable importance in recent years is the full-scale icing test undertaken to examine the performance of intake icing protection systems. Tests have been made on supersonic powerplants to complement conventional icing tests on isolated engines using connected test facilities and on helicopters to examine the likelihood of ice shed from the fuselage being ingested by the engine. These techniques are described in References 1 and 2.

The NGTE Engine Test Facility, shown in Figure 1, has five altitude test cells, three of which are normally used for free-jet testing. Together they have covered an extremely wide spectrum of tests, from low altitude subsonic tests on a small pulsejet through transonic launch tests of a ramjet missile engine to high altitude supersonic tests of the Concorde powerplant. These latter tests are described in References 3 and 4.

2. FREE-JET AND CONNECTED TESTING

The test arrangements for these two basic types of test are shown in Figure 2. Each type enables a different aspect of powerplant performance to be studied. The free-jet test enables intake/engine interactions to be explored whereas in a connected test the performance of the basic engine and its exhaust system is under examination. A comparison of the test capabilities of free-jet and connected facilities is given in Table 1.

TABLE I

CAPABILITIES OF FREE-JET AND CONNECTED TEST FACILITIES

Type of test	Ability to test	
	Free-jet	Connected
1. Engine steady state performance	Limited	Yes
2. Engine transient performance (handling)	Yes	Yes With limited simulation of inlet flow distortion
3. Intake/engine compatibility	Yes	Not applicable
4. Definition of intake control laws	Yes	Not applicable
5. Interaction between intake and engine controls during flight transients	Yes	Not applicable
6. Examination of off-design and failure cases (Mach overspeed, cold day, yaw, etc)	Yes	Yes Basic engine only
7. Examination of engine light-up sequence during launch (missiles)	Yes	Yes Limited tests on basic engine
8. Icing tests	Yes	Yes Basic engine only

Evaluation of engine steady state performance requires very precise measurements of engine airflow and thrust and such tests are made using connected rather than free-jet facilities. Connected tests are normally made with the cell set to the correct altitude pressure so that the pressure ratio across the engine propelling nozzle is exactly simulated. These considerations need not apply to free-jet tests and normal practice is to run with the cell pressure greater than that corresponding to the test altitude. This reduces the power required to extract the exhaust from the cell without influencing the internal gas dynamics of the engine, provided the propelling nozzle is "well choked". This method of operation requires the achievement of a good spill diffuser pressure recovery, an aspect which is discussed in Section 5.2.

Three further points concerning testing technique are worth noting:

a. It is necessary to run all engine tests, whether connected or free-jet, at the correct free flight air total temperature. This is to ensure that the engine operates at its correct non-dimensional speed and that the intake/engine matching is correct.

b. Free-jet tests have to be run using dry air to avoid condensation effects which degrade the flow quality in the jet.

c. The size of the free-jet is seldom large enough to allow extensive representation of parts of the aircraft structure adjacent to the air intake, but some simulation is usually possible with flat plates.

3. SCOPE OF FREE-JET TESTS

Free-jet tests can be run with either a dummy engine (ie plug nozzle) or a real engine installed behind the intake to investigate such factors as:

a. Intake pressure recovery and engine face pressure distortion (both steady state and time-variant). Tests can be at full-scale and are often made at full-scale Reynolds number.

b. The management of variable geometry intakes in terms of variables such as ramp angle, dump door angle, bleed flow, etc to establish intake control laws which will ensure acceptable engine face pressure distortion.

c. The effectiveness of intake anti-icing systems for both supersonic and subsonic power-plants, including those used in helicopters.

d. The suitability of the intake control laws with an engine running behind the intake. These can be assessed in terms of surge-free engine operation over a range of Mach No, pitch angle and yaw angle. By operating the engine off its normal control line the extent of the surge margin can be determined.

e. The combined effect on the engine surge characteristic of intake distortion and power off-take from multi-spool turbojets and turbofans.

f. The operation and effectiveness of the fuel control system during the launch and light-up phase of ramjet and turbojet engines for missiles.

4. DESCRIPTION OF FREE-JET TEST FACILITIES AT NGTE

4.1 Cell 1

Cell 1, which first ran in 1957, was designed for supersonic free-jet tests of ramjet engines. Fixed circular blowing nozzles are employed to cover a Mach number range from 1.8 to 3.6 and these are used in conjunction with an axisymmetric spill diffuser system. A selection of slotted blowing nozzles is also available for transonic tests. The blowing nozzle is mounted in a mechanism which allows the nozzle angle to be altered while the test is in progress between 0° and +25° in the pitch plane. The test envelope of Cell 1 when used in the high supersonic mode is shown in Figure 3, from which it will be seen that the correct air inlet temperature can be achieved only up to Mach 2.45 since the capacity of the air heater is limited. This cell can also be used for subsonic free-jet tests of small powerplants, but the test envelope in this mode of operation is rather limited since Cell 1 has no cold air supply. Two typical Cell 1 installations are shown in Figures 4A and 4B.

4.2 Cell 3 West

Cell 3 West, which became operational in 1969, was designed for connected tests of large fan engines, and as such is capable of passing airflows up to 550 kg/s. It was soon appreciated that the large airflow combined with the low inlet temperature capability made the cell ideal for free-jet icing tests of large installations. The cell has been used extensively for icing tests on the Concorde and Tornado powerplants (Reference 2) and for forward fuselage and air intake icing tests on Sea King and Lynx helicopters (Reference 5). When used as a free-jet test cell the blowing nozzle is fixed, but a limited range of pitch and yaw can be achieved by altering the attitude of the test installation. Some typical Cell 3 West icing installations are shown in Figures 5A and 5B. A description of the Cell is given in Reference 6.

4.3 Cell 4

Cell 4 was designed specifically as a supersonic free-jet cell capable of testing medium-sized powerplants at high altitude conditions over a Mach number range from 1.5 to 3.5. It was later 'stretched' to test the Concorde powerplant over a more limited range of conditions, and still later adapted for subsonic free-jet testing, although the latter capability is at present limited as no cold air supply is available. In its supersonic mode of operation the cell employs a variable Mach number blowing nozzle, a variable geometry rectangular spill diffuser system, and a capability to tilt the blowing nozzle in both pitch and yaw at rates of up to about 8 degrees per second. Figure 6 shows Cell 4 as used for supersonic free-jet tests of the Concorde powerplant, while Figure 7 shows the test envelope of the cell for the two supersonic blowing nozzles currently available.

Cell 4 has also been used for supersonic free-jet tests of the Tornado powerplant and for subsonic tests of the Jaguar, Tornado and missile powerplants. Two typical Cell 4 installations are shown in Figures 8A and 8B, while a full description of the Cell is given in Reference 7.

The leading particulars of the three cells are shown in Table II.

TABLE II

Cell	Dia (m)	Mach No range	Pitch/yaw capability	Free jet size dia (m) or area (m^2)	Type of nozzle	Air supply temp
1	3.65	Supersonic 1.8 to 3.6	0° to +25° in pitch	0.4 and 0.6 dia	Fixed circular	Min temp, ambient
		Transonic to 1.8	"	0.43 and 0.66 dia	circular slotted	Max temp, 210°C
		Subsonic	0° to +15°	made to suit	convergent	
3W	7.6	Subsonic	Nozzle fixed. Installation can be tilted approx 5°	up to 2.5 dia	convergent	Cold air to −37°C Max temp, ambient
4	9.1	Supersonic 1.5 to 3.5	±10° in pitch and yaw	1.1 area	variable M	Min temp, ambient
		1.75 to 2.3	"	2.3 area	variable M	Max temp, 470°C
		Subsonic	"	up to about 1.7 area	convergent	

5. SOME FREE-JET TEST TECHNIQUES

5.1 Blowing nozzles

For subsonic tests the blowing nozzle is usually made of sheet metal with external stiffening where necessary, and may be either circular or rectangular in cross section. For nozzles coupled directly to an air supply duct a gentle straight taper is employed, while for a nozzle fed from a plenum chamber a circular arc contraction followed by a parallel section has been used. An example of the latter type can be seen in Figure 8B.

The axisymmetric supersonic nozzles used in Cell 1 are of heavy cast iron construction with an accurately profile-machined bore.

The transonic slotted blowing nozzle (developed in Britain initially by Bristol Siddeley Engines Ltd for ramjet testing) is a device for producing a uniform parallel jet at subsonic, transonic and low supersonic Mach numbers without the complexity of variable geometry. It is particularly useful for simulating transient conditions such as the rocket boost phase of a ramjet powered missile where the flight speed changes from subsonic to supersonic over a relatively short time span.

The nozzle consists of a conventional convergent section where the airflow is accelerated to unity Mach number followed by a parallel section in the walls of which are cut longitudinal tapered slots having an increasing width towards the exit end of the nozzle. If a pressure ratio in excess of the critical value is applied across such a nozzle, the static pressure in the flow downstream of the convergent section is higher than the ambient pressure to which the nozzle is discharging and air is progressively spilled through the slots so that, with a correctly proportioned nozzle, the static pressure in the flow falls to the ambient pressure at the nozzle exit. A slotted nozzle of the type used in Cell 1 is shown in Figure 9. The slotted nozzle is extravagant in airflow demand and at M = 1.8 about 30 per cent of the flow escapes through the slots. In addition it is not practicable to use a spill diffuser system with such a nozzle. Both plant compressor and exhauster power requirements are therefore high. Current slotted nozzle designs are limited to a maximum Mach number of about 1.8. The variation of Mach number across the central 80 per cent of the jet diameter for a typical slotted nozzle is shown in Figure 10.

Two variable Mach number supersonic blowing nozzles are available for Cell 4 with exit areas of 1.1m^2 and 2.3m^2. They are of rectangular cross section with fixed exit area and variable throat. The vertical side walls of the nozzle are parallel and between them the flexible top and bottom walls move, deflected by movement of the throat blocks. The throat blocks pivot about a point such that the divergent section of the nozzle adopts a curvature which approximates closely to that required to give a parallel shock-free jet at the exit. The 1.1m^2 is constructed in stainless steel to withstand the high inlet air temperature corresponding to Mach 3.5 conditions (470^{o}C) and weighs some 38.5 tonnes exclusive of its supporting carriage. Figure 11 shows the entry of the 2.3m^2 nozzle as viewed from the cell plenum chamber. To keep the nozzle to a reasonable length and weight, bearing in mind that it has to be pitched and yawed at high angular rates (up to 8 degrees/s), it is accepted that the quality of the flow in terms of Mach number uniformity will not be as good as that achieved in a conventional wind tunnel. The variation of Mach number in the jet produced by the 1.1m^2 nozzle is shown in Figure 12. It will be seen that for Mach numbers in the range 1.8 to 3.0 the variation is less than $\pm$0.02. As the Mach number is increased above 3.0 the flow quality deteriorates due it is thought to the wall profiles departing from the required shapes.

5.2 Supersonic spill diffuser systems

To develop a parallel shock-free jet the isentropic pressure ratio corresponding to the recuired Mach number must be applied across the blowing nozzle. This pressure ratio increases rapidly with Mach number and at Mach 3.5 reaches a value of 76.3. If, however, some of the kinetic energy in the air which passes around the outside of the test intake is recovered as static pressure at the exit of a spill diffuser system, the overall pressure ratio across the cell need not attain such a high value. For example, with a typical spill diffuser recovery factor the overall pressure ratio required to achieve Mach 3.5 would be about 8.0. Since the blowing nozzle airflow in both cases is the same the exhauster volumetric flow capacity required to operate the cell is reduced by a factor of about 10 and the exhauster drive power reduced by a factor of 3. Spill diffusion is therefore an important technique for minimising plant size and operating power. This is illustrated in Figure 13.

At NGTE, spill diffusion systems have been developed for use with axisymmetric fixed Mach number nozzles and with rectangular variable Mach number nozzles. The latter systems were developed using models of about one fifteenth scale. Subsequent tests on the full-scale installations gave closely comparable performance in terms of pressure recovery. The axisymmetric systems are of fixed geometry but with a translating front ring to make cell 'starting' easier, while the rectangular systems have fully variable diffuser passages.

With both systems the model tests showed that a significant saving in the main exhauster capacity can be made if a small amount of air is pumped directly from the cell working section (about 4 per cent to 5 per cent of the blowing nozzle flow). If this working section bleed extraction system is made large enough it can also handle any low energy bleed flows from the test intake (ie ramp bleed, sidewall bleed, etc). The working section bleed has to be extracted from the section of the cell which is at the altitude pressure and this is achieved by using an air driven ejector for the first stage of compression, the ejector discharging into the main exhauster circuit.

The performance of a spill diffuser system is usually measured by the minimum overall pressure ratio necessary to maintain a parallel shock-free jet at the nozzle exit, and model test results for several rectangular systems of various spill factors and intake geometries are shown in Figure 14. The best performance achieved with axisymmetric systems is also shown for comparison.

5.3 Spill factor requirements

As mentioned in Section 5.2, the achievement of a high pressure recovery in the spill diffuser enables the size and power of the exhauster plant to be reduced. An equally important parameter is the spill factor, that is the ratio (jet area-intake area)/intake area, for this also directly affects the required exhauster capacity.

With supersonic installations it has been found possible to achieve spill factors in the region of 1.2 whilst retaining an ability to test over the appropriate ranges of pitch and yaw angles.

Once a test facility becomes available, practical considerations usually make it expedient to modify existing diffuser systems rather than build new ones so that tests are often run with larger spill factors than could be achieved using made-to-measure systems. This has been the case at NGTE and accounts for the fact that the Tornado propulsion system was tested at considerably higher spill factors than was Concorde.

The whole question of the spill factors required for subsonic testing is very much open to debate. It is considered further in Section 7.5.

Figure 15 shows the range of spill factors over which free-jet tests have been run at NGTE.

5.4 Boundary layer simulation

When an air intake is mounted close to an aircraft wing or fuselage, part of the boundary layer formed on the adjacent surface may find its way into the intake in some circumstances. Even if the boundary layer is not ingested directly, spillage flow from a boundary layer diverter may affect the main intake flow. In free-jet tests it would obviously be desirable to represent adjacent airframe surfaces in full, but usually the size of the test facility is insufficient to allow this. One solution to this problem which has been developed at NGTE and used during supersonic tests in Cell 4 is a device, known as a Boundary Layer Generator, which generates in a short axial distance a layer of low energy air with a pressure distribution approximating to that which exists in a turbulent boundary layer. The device is mounted within the blowing nozzle and adjacent to the test intake so that the artificial boundary layer is in the same position relative to the intake as the natural one on the aircraft. The development of the boundary layer generator is described in Reference 8.

Figures 16A and 16B show two such installations in Cell 4, the first simulating the wing boundary layer in the case of the Concorde intake and the second simulating a fuselage boundary layer.

6. INSTRUMENTATION

The majority of the instrumentation and data recording systems used at NGTE for free-jet testing follow conventional lines, but the need for two additional requirements has become evident.

These are:

a. Shock wave flow visualization systems for supersonic tests.

and b. an array of rapid-response miniature transducers for the determination of time-variant pressure distributions at the engine face.

Flow visualization of the intake shock system, particularly in the region of the cowl lip, is essential to enable the intake operating point to be defined and optimum matching with the engine achieved. A simple shadowgraph system has been found to be quite satisfactory, the shadowgraph screen being viewed with a television camera and the picture displayed on a television monitor in the cell control room.

The use of miniature transducers for the measurement of time-variant pressure distribution has been widely adopted and has been fully reported elsewhere. At NGTE a facility is available for continuously recording the signals from 40 rapid-response transducers mounted in an engine face total pressure rake. These records are processed off-line to yield either instantaneous distortion coefficients in digital form or computer-generated distribution patterns.

A detailed description of the cell instrumentation systems is given in Reference 9.

7. EXAMPLES OF FREE-JET TESTING

7.1 Engine face pressure distortion

Even under normal flight conditions the total pressure distribution at the engine entry may be far from uniform. This is particularly so at high supersonic flight speeds. Steady state and time-variant distortion patterns can readily be obtained from tests of a model intake, but whether an engine will tolerate a specific level of distortion can only be determined from full-scale tests with the engine installed behind its intake.

Steady state distortion is obtained from a conventional engine face pitot rake, the pressure being recorded on the steady state data gathering system. Because of the long pneumatic pipe lengths involved it is assumed that this gives time-averaged values. In fact the engine face pressures may be fluctuating continuously and this can result in the peak time-variant or dynamic distortion being considerably higher than the steady state value. Time-variant distortion is obtained by combining the steady state readings with the pressure fluctuations measured by miniature pressure transducers mounted in the rake adjacent to the steady state pitots, the pressure fluctuations being recorded continuously on magnetic tape.

Figure 17 presents some time-variant distortion records obtained from Cell 4, the circumferential distortion coefficient DC 135 being plotted against time. The steady state distortion level is also shown for comparison. It can be seen that the distortion level increases with Mach number, the peak time-variant distortion reaching about 1.5 to 1.7 times the steady state value.

7.2 Engine surge investigations

Engine handling tests are traditionally made in a connected facility using bias gauzes or plates to simulate specific pressure distortion patterns. This provides only a crude simulation of steady state distortion and cannot represent the changing patterns which occur as a transient progresses.

The great advantage of a free-jet facility lies in its ability to provide a far more accurate representation of conditions which exist in flight. This has now been firmly established as the result of extensive testing in Cell 4 and subsequent flight investigations.

Much of the later Concorde work was concerned with demonstrating surge-free operation over a range of conditions, including engine slam acceleration and deceleration, aircraft yaw and operation in 'cold day' conditions. These tests were made with the intake scheduled by its automatic control system.

Results from these tests are discussed in detail in Reference 3.

7.3 Thrust measurement

In a free-jet test the accurate measurement of thrust presents difficulties. However, thrust measurement has been attempted in certain specialized cases.

The first concerns altitude tests of ramjet engines where combustion performance can vary significantly with air inlet temperature and pressure and with flow distortions induced by the intake. To enable these effects to be quantified a force measuring system was developed for use in Cell 1. Briefly, the method was to measure the overall drag with the engine unlit over a range of cell conditions and, knowing the intake flow characteristics, to calculate the internal drag. The external drag of the installation could then be obtained and applied to the hot running condition. The combustor performance was obtained from the net thrust.

Another method of thrust measurement which has been used satisfactorily for a non-reheated turbojet installation is to use a pitot rake located in the jet exit. The engine airflow is obtained from an engine face rake and this is used with the measured values of exit total pressure and temperature to obtain the gross exit thrust. Because of the difficulty of obtaining reliable mean values of pressure and temperature in a non-uniform flow a high accuracy cannot be expected, but a value to within a few per cent can be obtained.

7.4 Dummy engine/real engine correlation

Tests with a dummy engine, ie a variable area plug nozzle, are useful when evaluating intake performance and establishing the optimum intake control laws. Experience has shown that data can be accumulated much more quickly and over a wider range of intake conditions without the additional complication and limitations imposed by an engine.

Comparisons of real and dummy engine results have been made on a number of powerplants and good correlation has been obtained. Figure 18 shows steady state and time-variant distortion coefficients measured on a subsonic intake in Cell 4 over a range of pitch angles. The steady state coefficients show good agreement between dummy and real engines, but the time-variant coefficients obtained with the real engine are slightly higher than those with the dummy. The intake pressure recoveries obtained with the real engine are also shown and these agree closely with those measured using the dummy engine.

7.5 Subsonic free-jet tests

Considerations of basic aerodynamics suggest that correct representation of the flow field in the neighbourhood of a subsonic intake requires that the area of the free-jet should be considerably greater than the capture area of the intake. This requirement is well known and i adhered to for conventional wind tunnel tests, especially if force and moment measurements are required, but it does not necessarily have to be interpreted so rigidly for propulsion testing. To examine this possibility, NGTE investigated whether the techniques it has developed for supersonic testing could be extended into the subsonic region.

In one such investigation, tests were made in Cell 4 using a two dimensional supersonic intake coupled to a dummy engine. The supersonic blowing nozzle was removed from the cell and replaced by a convergent nozzle having an outlet cross section 1.2 m wide by 1.6 m high. This gave a spill factor of 3.2. For comparison, the Concorde installation for supersonic tests in Cell 4 operated at a spill factor of 1.15.

The aim of the tests was to establish the extent to which the conditions in Cell 4 reproduced those existing in free flight. The success of the simulation was judged on the basis of the engine face pressure recovery. No full-scale free flight measurements were available and so results from wind tunnel tests on an extensively instrumented model of the complete aircraft were used to provide the datum for comparison.

The tests in Cell 4 covered a range of angles of intake incidence up to 30° whilst the wind tunnel tests on the model aircraft were taken to 20° aircraft incidence. A comparison of the measured intake pressure recoveries at Mach 0.5 is shown in Figure 19. The variation of pressure recovery with intake flow ratio follows the same general trend for both the cell and aircraft model data, but the curve for a given intake incidence in the cell is roughly equivalent to the aircraft model operating at half that incidence. It is thought that fuselage upwash and Reynolds number effects on the aircraft model could account for some of this difference although probably not for all.

7.6 Flow visualization

Occasionally circumstances arise when a qualitative indication of the flow field in an intake can be helpful in gaining a better understanding of its behaviour and thereby enabling its performance to be improved. A simple technique which has been used with success in Cell 4 requires the intake surfaces of interest to be painted in bands with a mixture of titanium dioxide and oil after which the cell is run at a fixed condition for 15 or 20 minutes and then shut down as quickly as possible. Although the resulting patterns relate to the flow in the boundary layer and are not necessarily representative of the main bulk of the flow, nevertheless they provide a useful indication of conditions within the intake, particularly when it is operating off-design.

Figures 20A and 20B show the results of two tests on the Concorde installation run at Mach 2. Figure 20A shows the path of the bleed flow into the intake ramp void and Figure 20B, which was taken after a run at 4° yaw, shows how the effects of yawed flow persist on the floor of the intake duct right up to the engine face.

7.7 Free-jet icing tests

The increased importance attached in recent years to free-jet icing tests, particularly on helicopter installations, was mentioned in the Introduction. At NGTE these tests are made in the largest altitude cell, Cell 3 West.

A general description of this facility is given in Reference 6, and results from free-jet icing tests on helicopters and on an aircraft powerplant (Concorde) are given in References 5 and 2 respectively.

The Concorde tests were made at a spill factor of 1.8 and covered a range of conditions representing encounters with icing conditions of varying severity with the engine operating at reduced power. At the end of each icing test period the engine was accelerated to maximum continuous power. Figure 21A, one of a series of photographs taken during tests on the Concorde powerplant, shows the ice deposition after 30 minutes at an air temperature of -10°C with a water concentration varied cyclically between 0.6 and 2.0 gm/m^3. The flight condition represented was Mach 0.5 at 5.2 km altitude. The tests in Cell 3 West were made before the aircraft had flown in natural icing conditions and at this stage there was some uncertainty as to the extent to which the cell was representative of free flight. However, subsequent cold weather trials on the aircraft gave results identical with those obtained in the cell and this added further to the background of comparative data which shows that free-jet cell tests can be used to give reliable indications of conditions expected to occur in flight.

Free-jet icing tests on helicopters are undertaken at NGTE primarily to examine the performance of the engine intake anti-icing system. For such tests to be representative it is necessary to test with as much of the fuselage surface ahead of the intake present as possible to ensure that the flow field in the region of the intake capture plane is correct. This can be done at NGTE because the facilities available are of such a size that full-scale helicopter fuselages can be accommodated in the test cell and, by correctly positioning them in the air stream, a large part including the windscreen, cabin roof and engine intakes can be subjected to icing conditions. No attempt is made to simulate rotor downwash, the tests representing forward flight in the speed range 45-80 m/s.

The question as to the extent of full-scale representation arose in an even more acute form when helicopter tests were first proposed since the test vehicle is only partially immersed in the free-jet, a practice wholly at variance with conventional wind tunnel practice.

To resolve this problem NGTE undertook tests using a one-twentieth scale model of the complete cell installation, including a model helicopter. Measurements were made of the pressure distributions over the fuselage surfaces for comparison with those made on the same model helicopter in a conventional closed circuit wind tunnel. The close agreement obtained between the two sets of data gave confidence that the Cell 3 West representation was good and this has been subsequently confirmed by flight tests.

Figure 21B, taken from Reference 5, shows the ice build up on the untreated front face of a foreign object deflector fitted to a Sea King helicopter. The test was of 30 minutes duration at -4°C air temperature with water concentrations of 0.75 g/m^3 for 27 minutes and 1.5 g/m^3 for 3 minutes.

8. CONCLUDING REMARKS

This paper has presented a summary of free-jet testing techniques as they have been developed at NGTE. Most of the work has been directed towards supersonic testing, originally to enable ramjet engines for missiles to be developed but later extended to cover powerplants for supersonic aircraft. More recently the potential of subsonic free-jet testing has been explored and has been found to be considerable, especially for icing tests on helicopters and on powerplants for fixed wing aircraft. Although when judged by conventional wind tunnel practice the techniques for subsonic testing appear crude and unrepresentative, a sufficient background of experience has now been accumulated to give confidence that the results obtained from a cell test closely represent what occurs in flight.

In the case of supersonic aircraft, where interactions between the intake and engine can critically affect the performance of the powerplant, the ability to test the complete system is immensely valuable. No connected test can completely simulate the flow conditions at the engine entry plane over the whole range of power settings and flight conditions. In a free-jet test, steady state and time-variant total pressure distributions are correctly reproduced by the intake so that aircraft manoeuvre and off-design cases can be examined without hazarding a test aircraft or demanding unnecessary risks to be taken by a flight crew. This is not to say that connected tests have no part to play in supersonic powerplant development, indeed this is far from the case, but rather to emphasise that a balanced development programme must include a significant element devoted to free-jet tests of the engine coupled to its intake.

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Fig.1 The engine test facility at NGTE

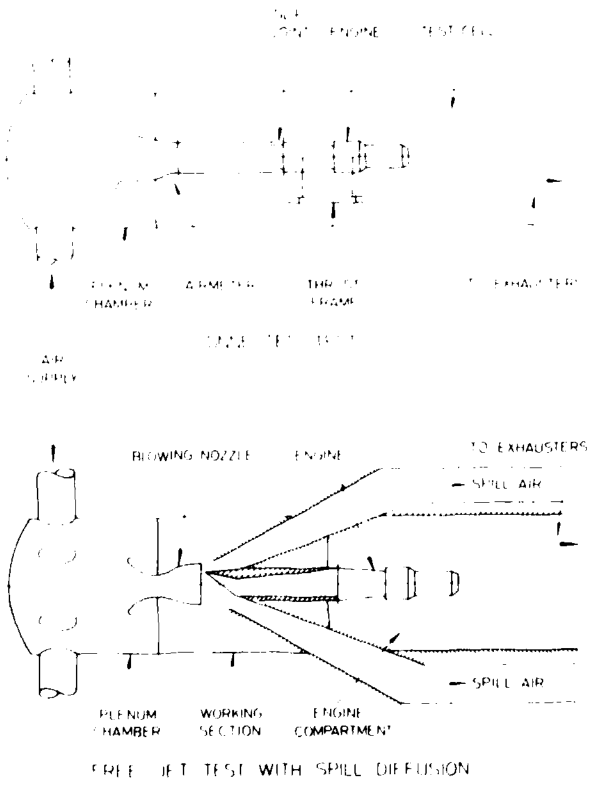


Fig.2 Connected and free-jet engine tests

ASSUMPTIONS MAXIMUM AIRFLOW AVAILABLE AT NOZZLE = 180 kg/S
MAXIMUM TEMP. AVAILABLE AT NOZZLE = 483K
MAXIMUM PRESS. AVAILABLE AT NOZZLE = 793 kPa ABS

OPTIMUM SPILL DIFFUSER SYSTEM

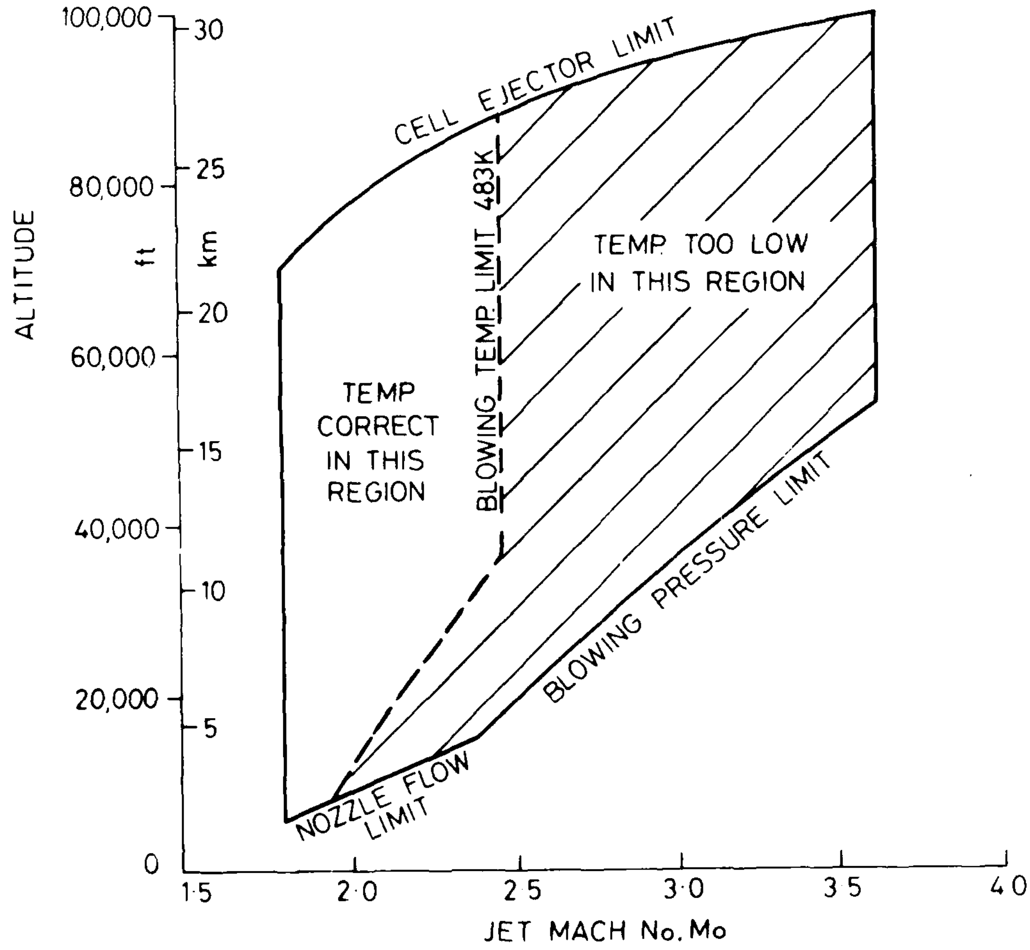


Fig.3 Cell 1 test envelope with 0.6m dia free jet

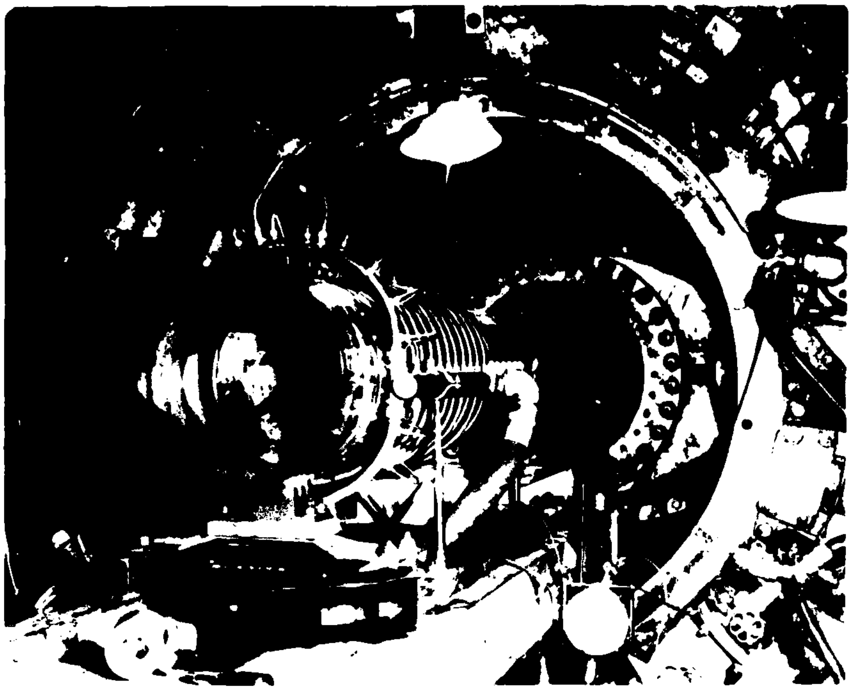


Fig.4A Ramjet

Fig.4B Turbojet

Fig.4 Free-jet test installations in cell 1

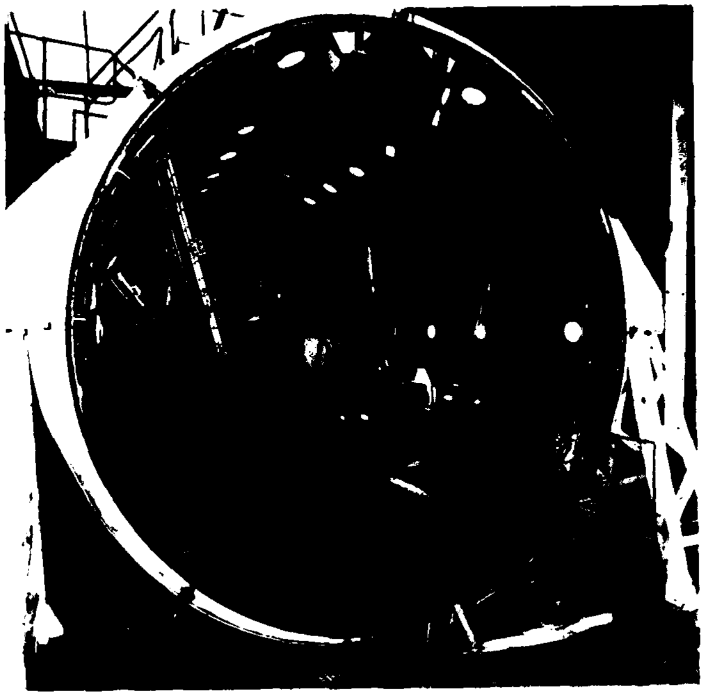


Fig.5A Helicopter fuselage with engine intakes

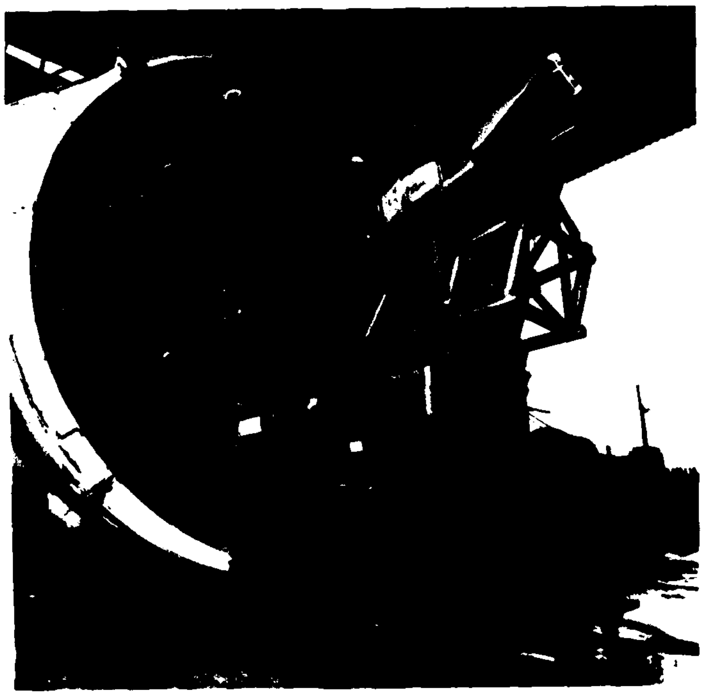


Fig.5B Aircraft powerplant intake and engine

Fig.5 Free-jet test installations in cell 3W

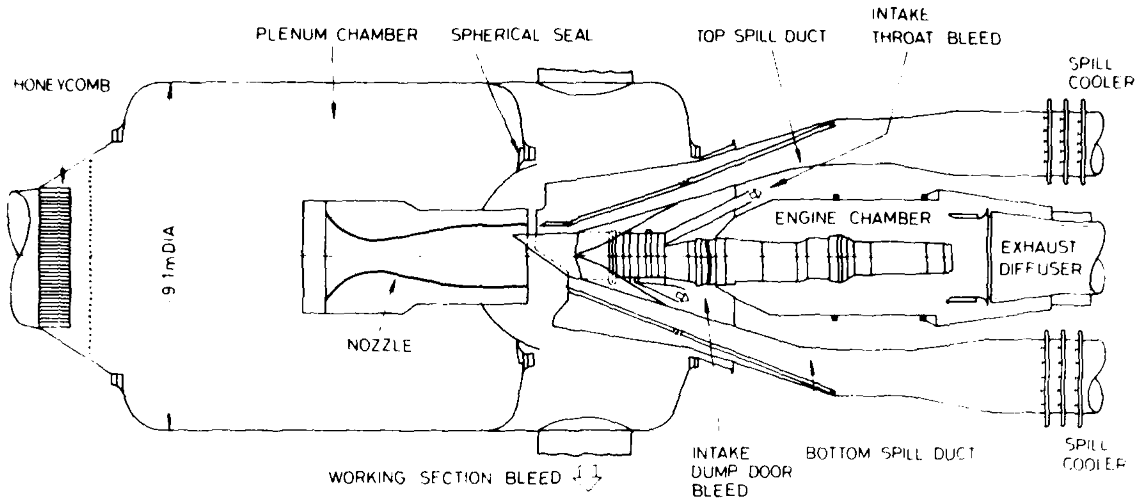


Fig.6 Supersonic free-jet test of Concorde powerplant

NOTE ISA FREE STREAM TOTAL TEMPERATURE AVAILABLE AT ALL POINTS

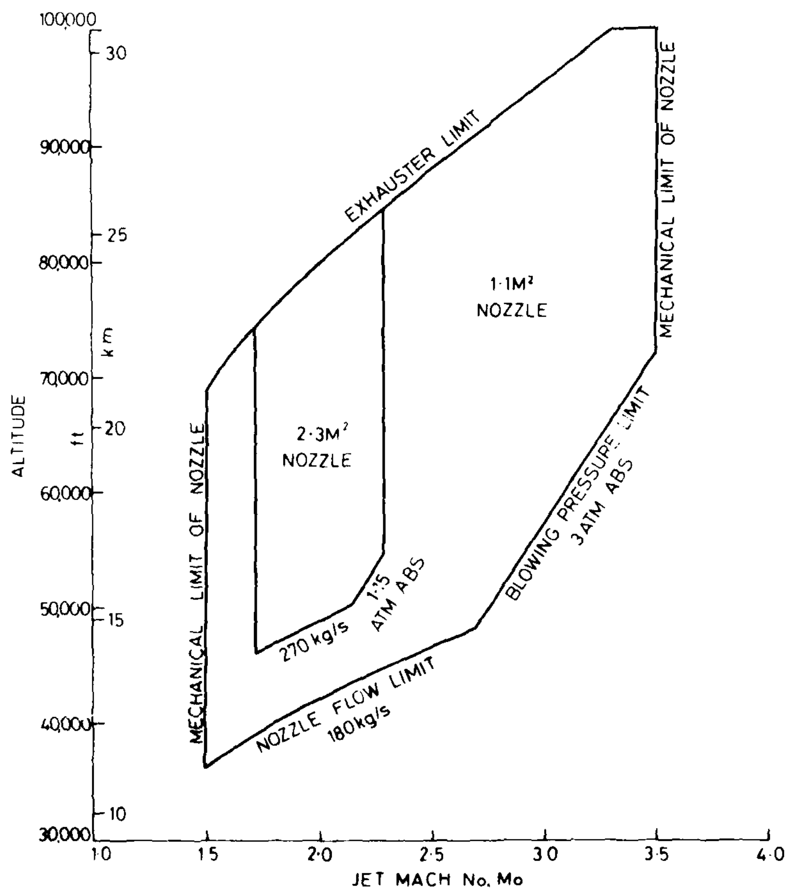


Fig.7 Cell 4 supersonic test envelopes

Fig.8A Supersonic powerplant

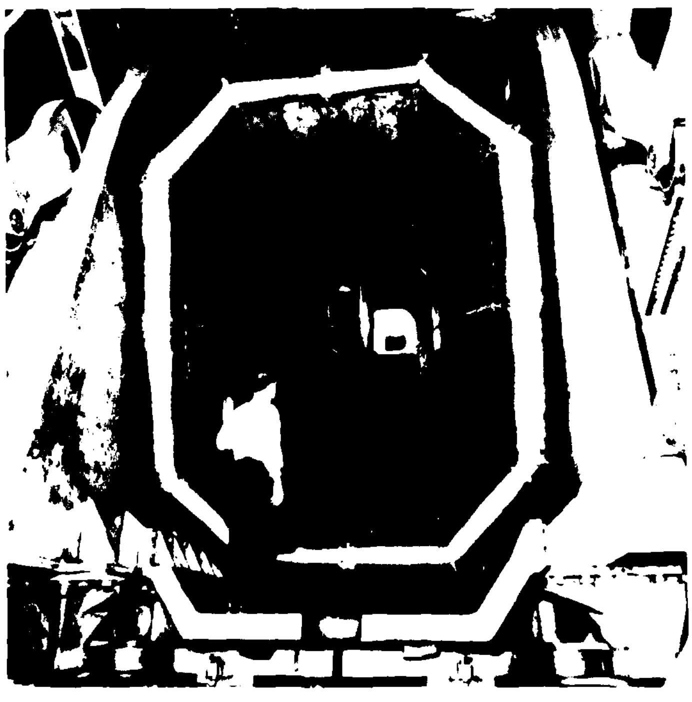


Fig.8B Subsonic powerplant

Fig.8 Free-jet tests in cell 4

Fig.9 Slotted blowing nozzle for transonic tests in cell 1

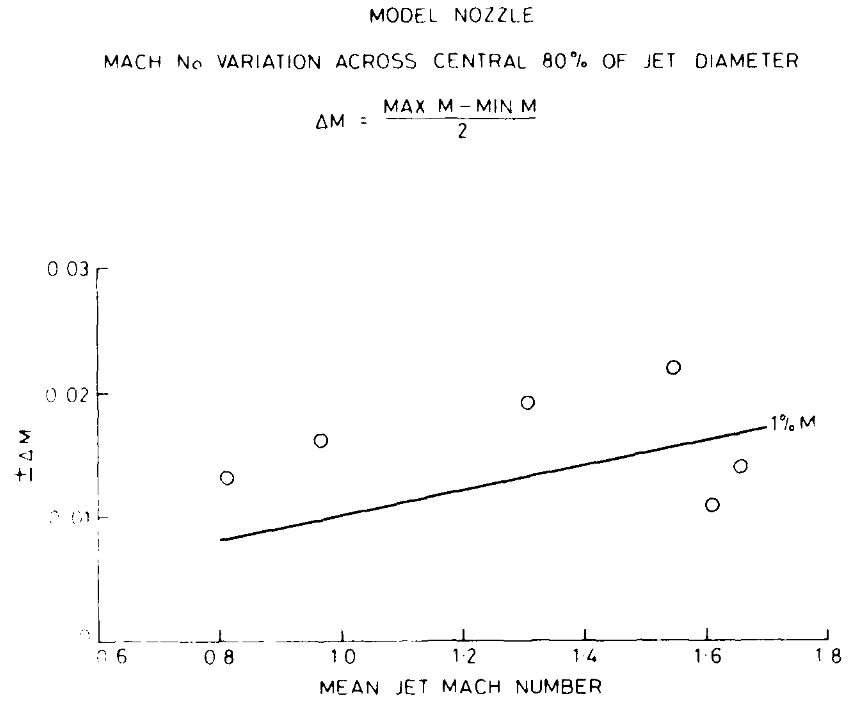


Fig 10 Slotted nozzle mach no. variation in jet

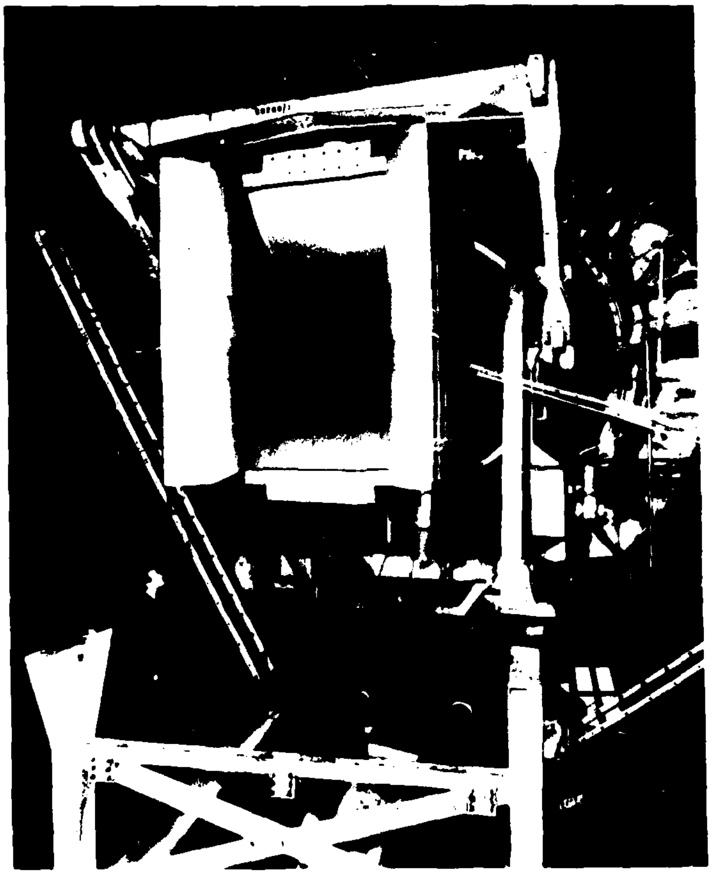


Fig.11 Supersonic nozzle for cell 4

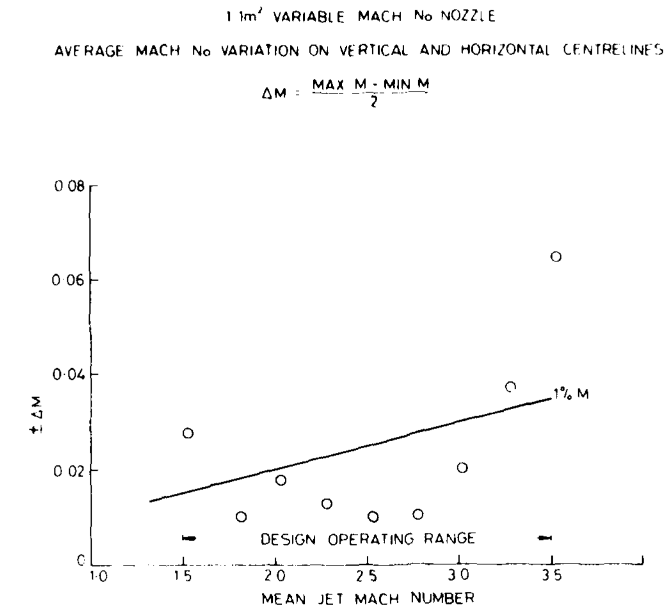


Fig.12 Variable mach no. nozzle mach no. variation in jet

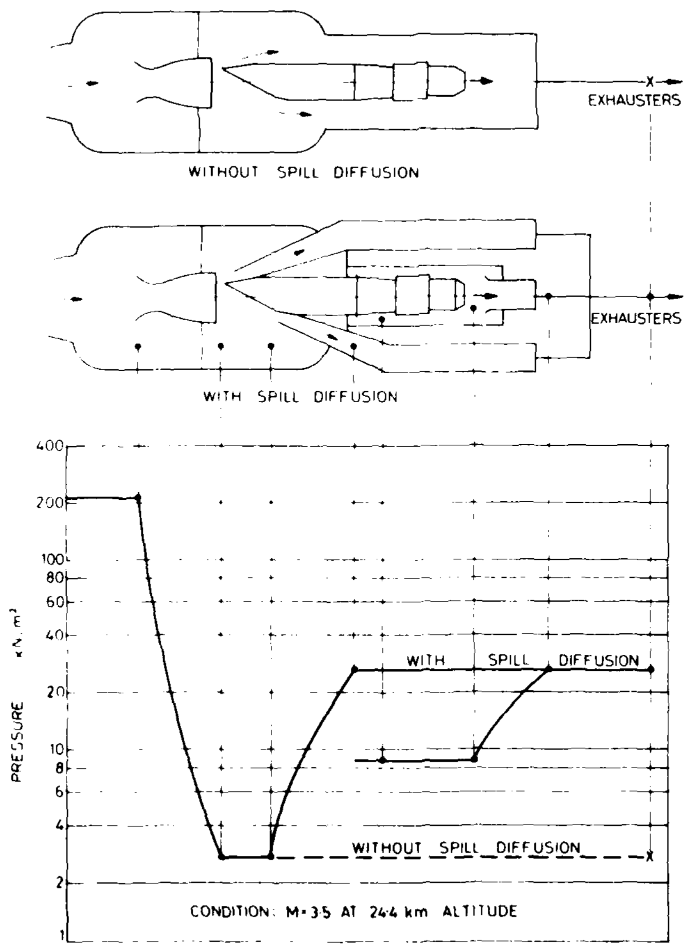


Fig.13 Pressure changes through free-jet test cell

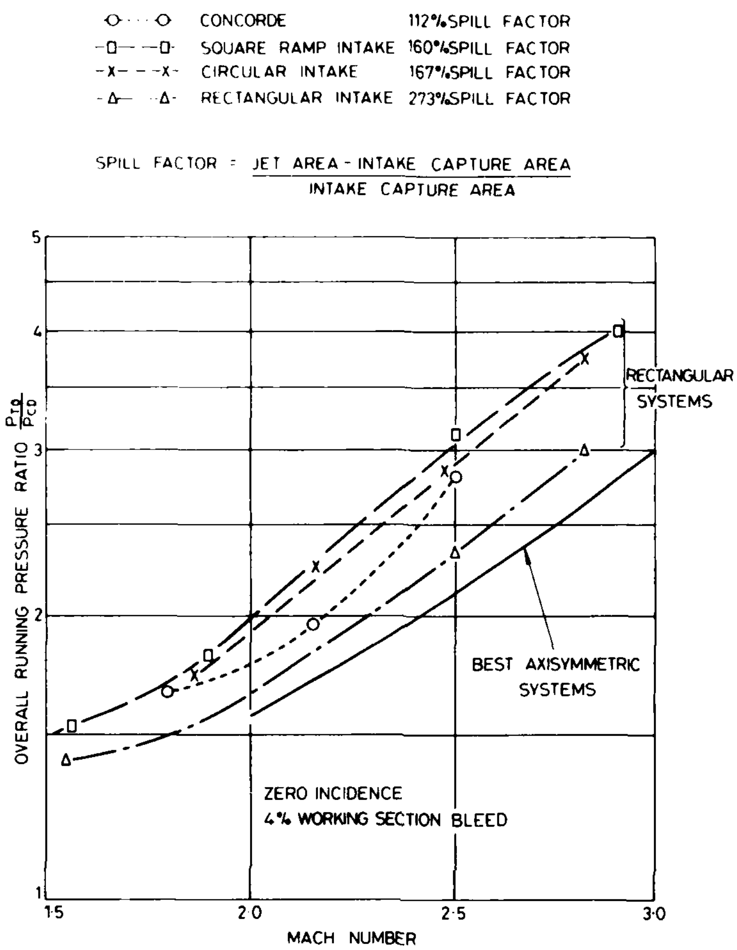


Fig.14 Performance of some NGTE spill diffuser systems

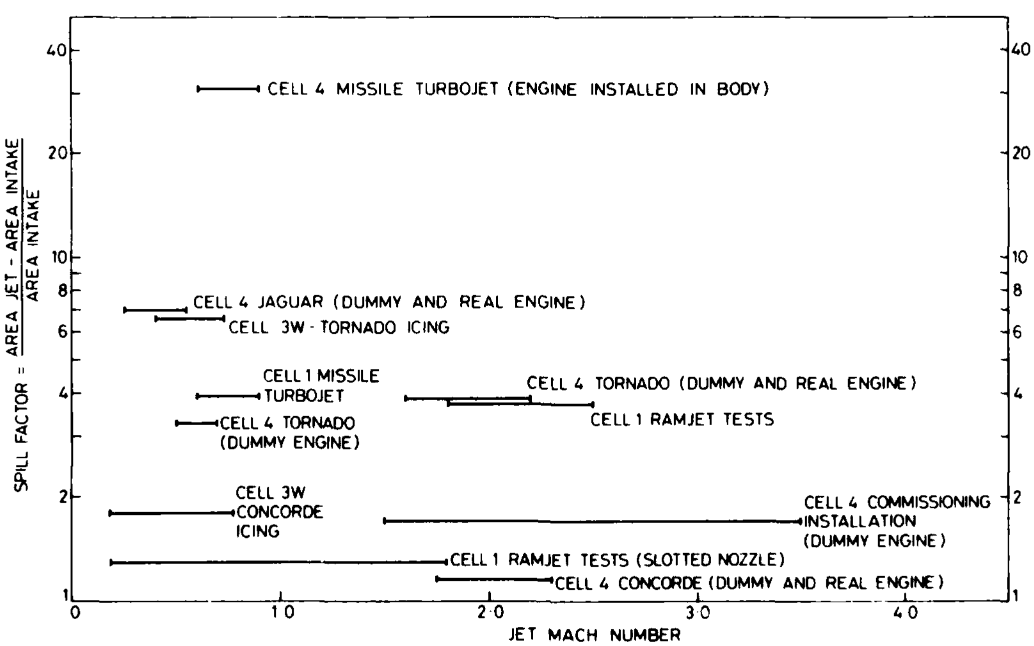
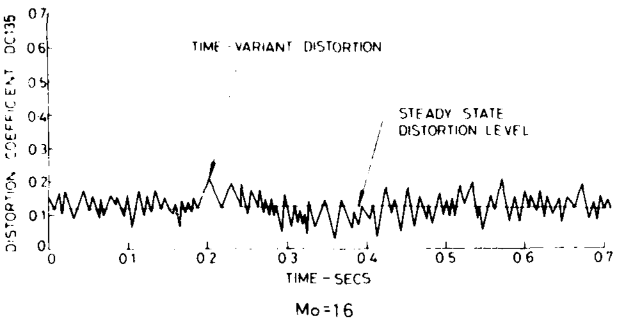


Fig.15 Spill factor of some free-jet test installations

Fig.16A Wing boundary layer simulation

Fig.16B Fuselage boundary layer simulation



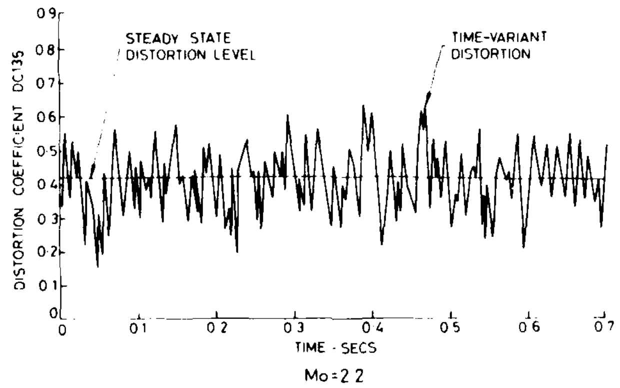
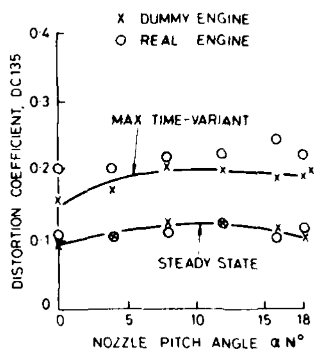


Fig.17 Engine face time-variant distortion



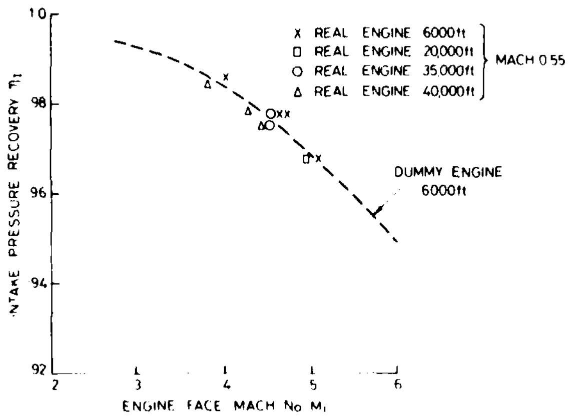


Fig.18 Correlation of dummy engine/real engine results

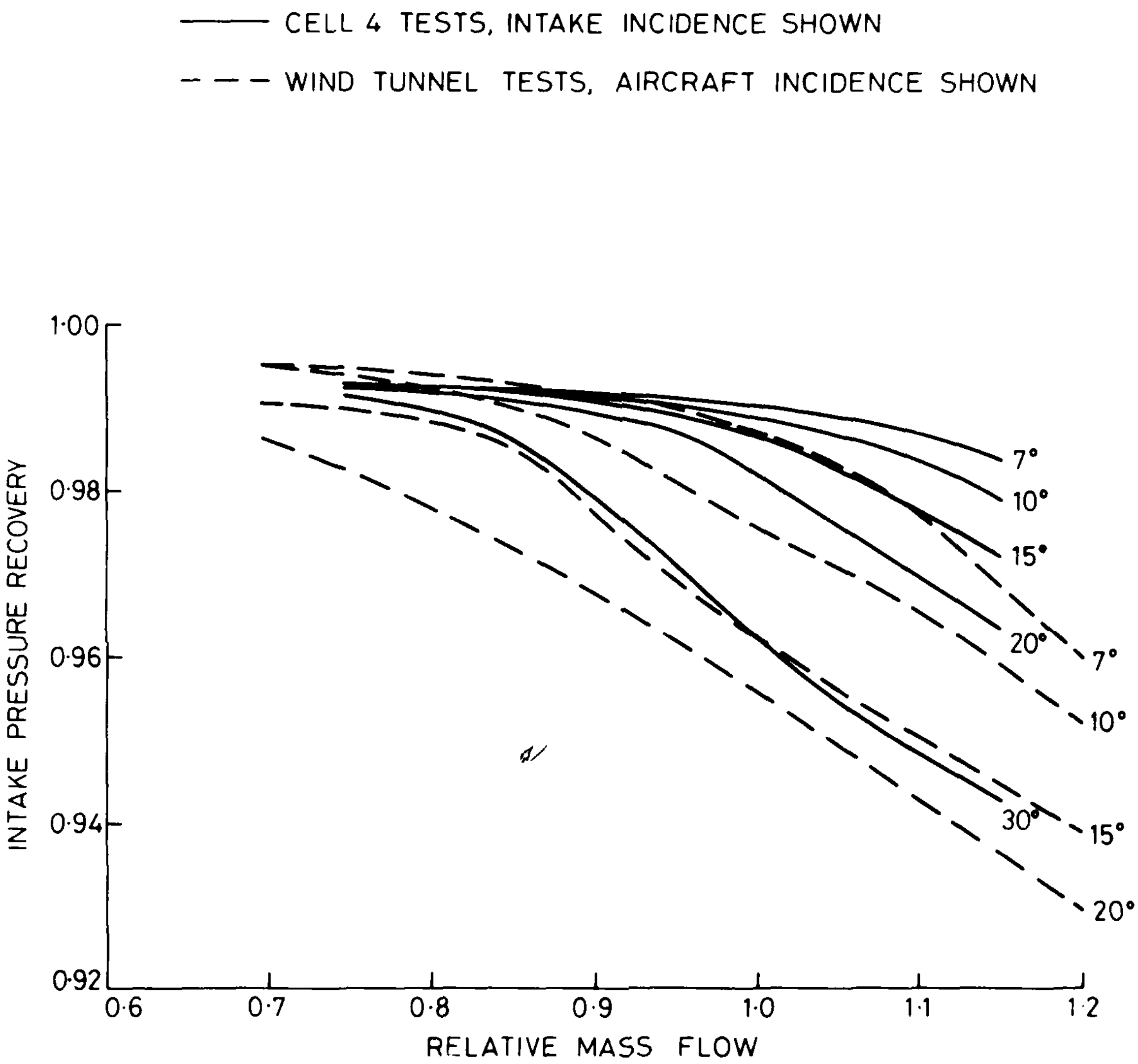


Fig.19 Cell 4/wind tunnel comparison. Mo 0.5

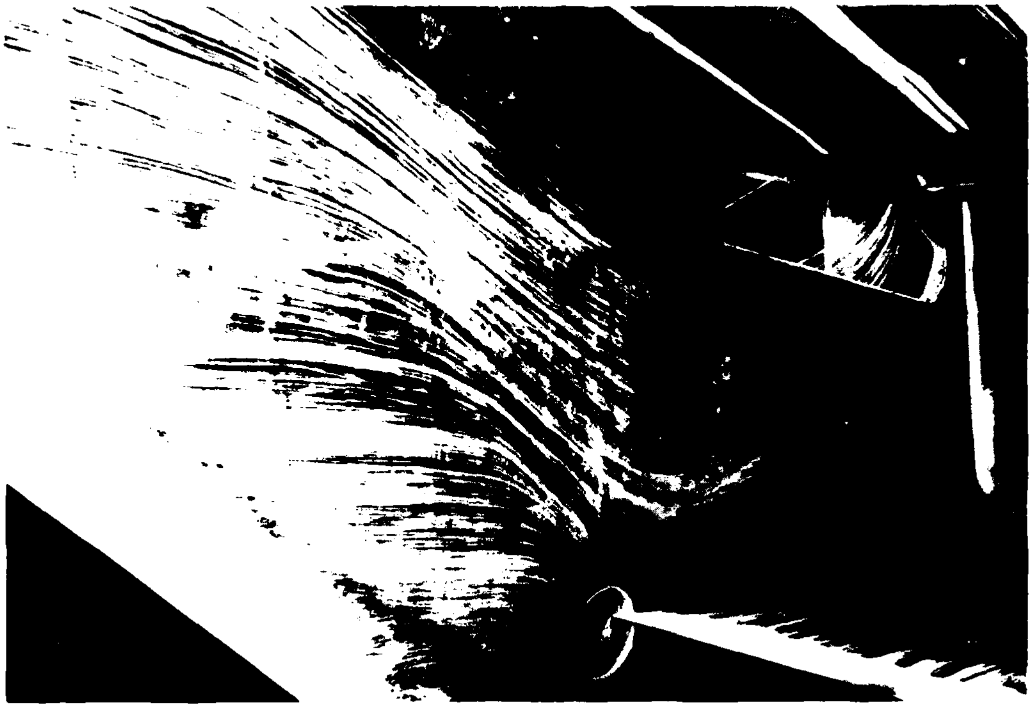


Fig.20A Flow visualization flow into ramp void

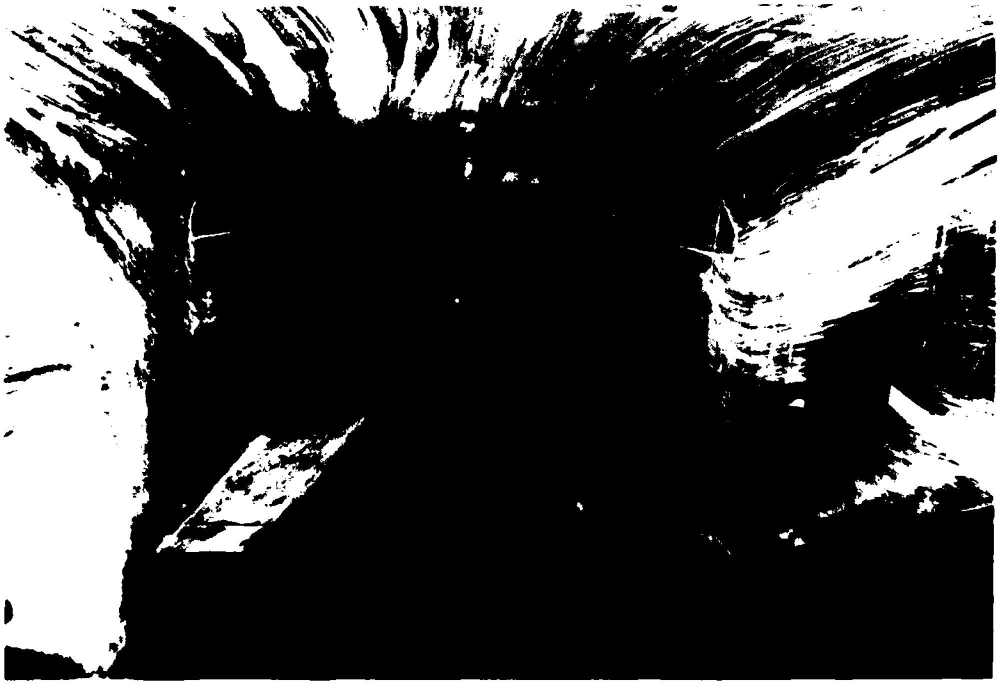


Fig.20B Flow visualization effect of 4° yaw

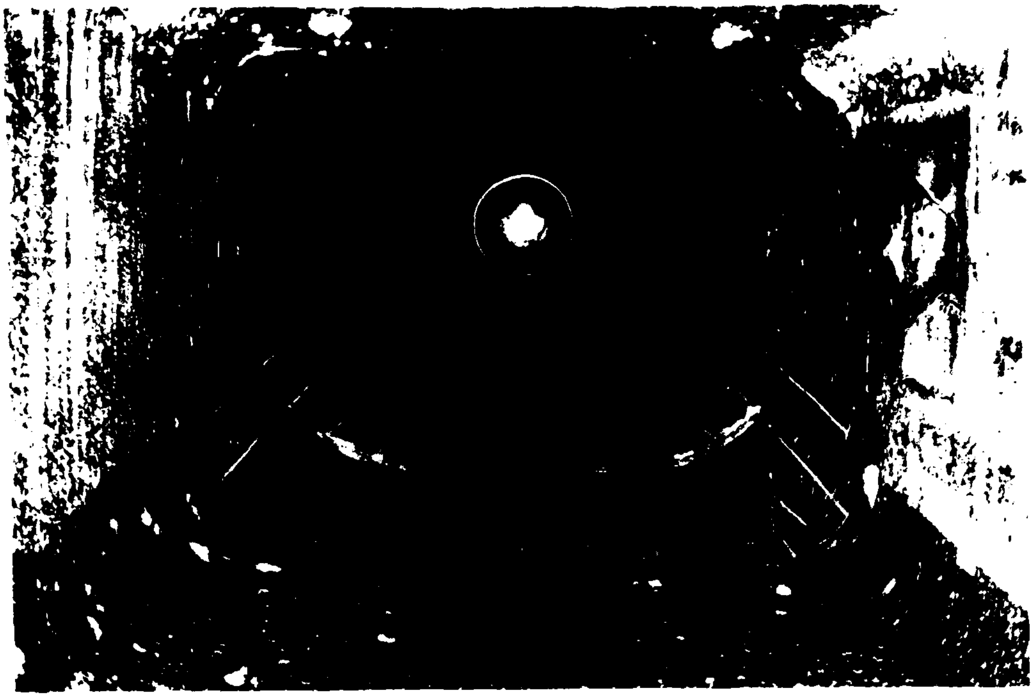


Fig.21A   Icing test of Concorde powerplant

Fig.21B   Icing test of Sea King helicopter

DISCUSSION

Eugene E.Covert, Mass. Inst. of Tech., US

(1) How was the spill air collection system arranged for subsonic testing?

(2) *In the subsonic installations, was the exhaust plane of the engine subjected to the spill air environment?*

Author's Reply

(1) The inner walls of the spill diffusers used for supersonic testing were removed to form a plenum around the inlet. No attempt was made to diffuse the spill air.

(2) The majority of the spill air was taken away through separate ducts, but some was passed over the engine to give cooling and ventilation. The engine exhaust nozzle discharged to the same pressure as at the inlet.

INLET-ENGINE COMPATIBILITY TESTING TECHNIQUES IN GROUND TEST FACILITIES

by
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SUMMARY

Inlet-engine compatibility is a recognized major concern in essentially every aircraft development effort. Testing of many configurations throughout the development cycle of a propulsion system is generally required to establish the performance and operational suitability characteristics of a production configuration. A disciplined test methodology based on the use of standardized test techniques is necessary to generate the technical data base required for effective program management decisions.

Established test techniques have been developed at the Arnold Engineering Development Center for the systematic and quantitative assessment of turbine engine performance and suitability. A review of the currently available techniques for the evaluation of turbine engine stability is presented. Recommended test matrix selection criteria, instrumentation and test equipment requirements, test procedures, and analysis techniques are discussed with respect to turbine engine testing with three basic engine inlet environmental conditions: uniform, steady flow; steady-state distorted flow; and time-variant distorted flow.

1. INTRODUCTION

Basic propulsion system requirements can be grouped into two primary classifications: steady-state thermodynamic performance requirements which are usually summarized in terms of thrust and range capability, and operational suitability requirements generally described in terms of system stability, durability, and controllability. The requirements for steady-state performance assessment are readily understood and historically have received much attention and emphasis. Standardized test techniques have been established with general acceptance and use throughout the aircraft industry. In contrast, operational suitability requirements have received significant attention only since the advent of sophisticated multi-mission aircraft designs. Inlet-engine operational difficulties substantially in excess of anticipated or tolerable levels were experienced with the introduction of turbofan engine installations in tactical aircraft.

Considerable effort has been expended over the last decade to develop a thorough understanding of engine suitability with particular emphasis on inlet-engine compatibility problems. As a result, there is general agreement throughout the industry concerning the cause and solution of compressor system instability, and a recognized need for industry guidelines to assess and systematically evaluate compressor stability (Ref. 1).

In response to this need, established test techniques have been developed at the Arnold Engineering Development Center (AEDC) for the quantitative assessment of turbine engine performance and compressor stability at mission-related test environmental conditions. A review of the compressor stability test techniques applicable to full-scale turbine engine system test requirements is presented.

Many of the techniques discussed are based on testing conducted at the Arnold Engineering Development Center with the sponsorship of the Air Force Aero Propulsion Laboratory (Ref. 2).

2. ENGINE STABILITY MARGIN REQUIREMENTS

During normal engine operation, compressor stability is affected by many factors. A graphic presentation of the primary factors is shown in Fig. 1 and discussed in detail in Ref. 3.

Evaluation of inlet-engine compatibility requires consideration of the compressor stability margin utilization due to inlet flow distortion and all other factors which either degrade the compressor surge line or increase the operating compressor pressure ratio. Stability assessments to quantitatively estimate the singular and cumulative effects of these destabilizing factors are necessary throughout the propulsion system development cycle. Engine test techniques which may be used to systematically determine margin utilization, in a building block concept, have been developed based on three basic engine inlet environmental conditions: uniform, steady flow; steady-state distorted flow; and time-variant distorted flow.

3. UNIFORM STEADY ENGINE INLET FLOW

Testing with uniform, steady, engine inlet flow is required to establish engine baseline performance (normal operating line and surge line) from which to quantify the effects of destabilizing factors on compressor stability margin.

Engine inlet flow quality for baseline testing can be defined in terms of measured inlet total pressure and temperature parameters (Ref. 4). Based on test experience at the Arnold Engineering Development Center, engine inlet flow conditions with one percent or less steady-state spatial total pressure distortion $(P_{max} - P_{min})/P_{avg}$, excluding the boundary layer, and one percent or less time-variant spatial inlet distortion, characterized by $\Delta P_{rms}/P_{avg}$ (0 - 700 Hz), and one percent or less spatial and time-variant total temperature distortion are satisfactory for baseline stability testing.

Stability requirements are generally divided into two major groups for consideration in establishing a test condition matrix: requirements caused by changes in environment, and the requirements for engine operation. The changes in operating environment include the effects of Reynolds number (flight altitude), corrected rotor speed (flight Mach number), nonstandard day conditions (inlet temperature), external engine thermal environment, and flight transients where stability characteristics may vary as engine internal thermal equilibrium is achieved.

The effect of Reynolds number on engine stability can be determined by investigation at selected engine inlet Reynolds number index test conditions. The maximum Reynolds number index selected should be representative of low altitude engine operating conditions. The minimum Reynolds number index condition should be selected to provide data along the lift limit of the engine operating envelope. Test conditions suitable for investigation of a typical current augmented turbofan engine are shown in Fig. 2. Testing over a range of inlet air total temperatures provides data at corrected compressor rotor speeds and thermal conditions representative of a variation of flight Mach number.

Typical requirements for engine operation include (1) operation with and without engine airbleed and power extraction, (2) operation with control system variations representative of control tolerances, (3) operation with steady-state nonaugmented and augmented control logics, (4) engine power transient operation, and (5) engine operation during flight trajectory transients.

3.1 TEST INSTALLATION - UNIFORM STEADY ENGINE INLET FLOW

Uniform inlet conditions can be obtained with a conventional direct-connect engine test installation (Fig. 3). Flow straightening screens downstream of a critical flow airflow measuring venturi, a large inlet plenum, and a bellmouth at the engine inlet duct provide metered airflow with uniform inlet total pressure and temperature profiles and minimum time-variant inlet distortion.

Steady-state instrumentation is required at selected stations within the engine to establish engine baseline total temperature and pressure profiles from which to evaluate variations attributed to induced inlet flow disturbances. Transient instrumentation (low frequency range of flat response to approximately 20 Hz) systems are required for selected sensors to obtain performance during engine power or flight trajectory transients. Limited dynamic instrumentation (high frequency range of flat response to approximately 1,000 Hz) systems are required to identify the critical compressor component or origin of compressor flow breakdown.

3.2 TEST PROCEDURE - UNIFORM STEADY ENGINE INLET FLOW

Stability margin, defined as the operating pressure ratio range between the normal operating line and surge line determined at constant compressor corrected airflow, is shown in Fig. 4. The change between the operating line and surge line with inlet distortion provides a quantitative measure of the stability margin utilization attributable to the inlet distortion condition. An array of compressor loading techniques is available to experimentally determine engine-installed compressor surge limits. In general, these techniques may be divided into two general classifications, transient and steady-state loading methods. The most extensively used transient techniques include variations of the fuel pulse method which is based on use of a controlled transient fuel pulse to increase the compressor pressure ratio above the operating line (Ref. 5). The technique usually requires only minor test equipment modifications, but does require transient measurements which are generally more difficult to obtain, with accuracy, than comparable steady-state measurements.

Steady-state loading techniques typically include flow blockage methods such as "inbleed" or mechanical blockage systems. A typical example of a steady-state loading system used with a dual rotor compressor is reported in Ref. 2. The fan compressor is loaded by reducing the exhaust nozzle area, and the high-pressure compressor is loaded with a high-pressure compressor inbleed system. By simultaneous use of the gas generator and fan compressor loading systems, valid matched component operating conditions can be maintained. A rotor speed ratio corresponding to the steady-state normal operating value, or a rotor speed ratio based on data from a mathematical model may be maintained for matched component operation.

3.3 ANALYSIS METHODS - UNIFORM STEADY ENGINE INLET FLOW

Typical results obtained during testing with uniform inlet flow conditions to determine engine baseline performance are shown in Fig. 5. Data defining the high-pressure compressor operating characteristics are presented; high-pressure compressor pressure ratio is shown as a function of high-pressure compressor corrected airflow. The data obtained at a compressor inlet Reynolds number index of 0.6 are representative of performance at moderate to low altitude conditions within the engine operating envelope. Corrected rotor speeds from approximately 9,900 rpm at high flight Mach numbers to 10,500 rpm at low flight Mach numbers are representative of engine operation at military and/or afterburning power for the typical turbofan engine reported in Ref. 2. The variation of baseline stability margin attributed to Reynolds number, or altitude effects, is indicated by the significant shift in the operating line and slight variation in the surge line obtained at Reynolds number index 0.3 test conditions. At a constant high-pressure compressor rotor speed of 10,000 rpm, the stability margin decreased from about 18 percent at Reynolds number index 0.6 to approximately 14 percent at Reynolds number index 0.3.

Stability margin requirements for engine transient operation such as engine acceleration and deceleration transients or engine flight transients are obtained using close-coupled (approximately 20-Hz response) pressure transducers and high-speed digital data acquisition systems. Typical results obtained during acceleration and deceleration engine transients are shown in Fig. 6 for the high-pressure compressor of a typical turbofan engine. Stability margin allowance is required for the acceleration transient; margin is increased during the engine deceleration transient. Margin requirements typically are reversed for the fan compressor; margin is increased during acceleration and utilized during deceleration.

4. STEADY-STATE INLET DISTORTION

Testing with steady-state inlet total pressure distortion is required to assess the effects of classical and composite distortion patterns on the baseline stability characteristics of the compression system.

The test matrix should be selected to provide data at the same engine and environmental test conditions which were investigated with uniform inlet flow. During development testing, classical or parametric patterns are generally selected to determine or verify the basic distortion sensitivity characteristics of the compression system. Testing is conducted to evaluate the effects of pattern shape, extent, and intensity on compression system stability and to determine the distortion transfer coefficients of multi-spool compression systems.

For qualification or certification testing, test matrix requirements are generally limited to specified engine and environmental "rating" conditions using flight-type distortion patterns. The flight patterns are derived from inlet model test results obtained at selected mission-related operating conditions. An established practice in industry has been to simulate inlet model derived peak time-variant distortion patterns with steady-state patterns for engine stability test assessments.

4.1 TEST INSTALLATION - STEADY-STATE INLET DISTORTION

Testing with steady-state total pressure distortion can be accomplished in a direct-connect installation similar to that used for testing with uniform inlet testing, except for the installation of a steady-state distortion generator approximately one engine diameter forward of the compressor inlet.

Screens have been accepted throughout the aircraft engine community as a standard steady-state distortion generating system. However, screens have an undesirable operating characteristic because the pressure loss is dependent on the screen porosity (blockage) and approach velocity; therefore, each distortion pattern variation and/or engine inlet duct velocity (engine power level or simulated flight Mach number) change requires test time utilization for a screen configuration change to maintain a desired total pressure distortion pattern at the compressor inlet.

Airjet systems are an attractive alternate method for generating steady-state distortion patterns for many test program applications. Airjet distortion generators use a counterflow (to the primary engine inlet airstream) airjet system in which the jet flow momentum cancels part of the primary compressor inlet airstream momentum with the accompanying total pressure loss. The flow distortion pattern is varied by remotely controlling the jet flow rate and distribution.

An AEDC-developed airjet distortion generator system (Fig. 7) is reported in Ref. 6. The system includes a secondary (airjet) air temperature conditioning system (to match the temperature of the primary engine airstream); a high-pressure (sonic) airjet nozzle array (56 equally-spaced flow nozzles); and a computerized airjet nozzle flow control system to provide a dial-a-pattern capability. The AEDC airjet distortion generator has been shown (Refs. 6 and 7) to be an efficient tool for providing steady-state inlet total pressure distortion during turbine engine stability tests. The system flexibility provides an order of magnitude increase in the available inlet patterns over current screen techniques. Desired patterns generally can be obtained with approximately the same accuracy limits obtainable with a screen distortion generator. Typical pattern characteristics, as shown by the isobar map at the engine inlet, are presented in Fig. 8. The

airjet distortion generator produced similar areas of high and low total pressure and maintained similar area contours to those produced by distortion screens.

Similar airjet distortion generators are reported in Refs. 8 and 9.

4.2 INSTRUMENTATION - STEADY-STATE INLET DISTORTION

The instrumentation requirements for testing with steady-state distortion are similar to the requirements for testing with uniform inlet flow conditions except for the increased survey required to measure distortion at the compressor inlet and to measure the propagation of distortion through the engine. Test assessment (and communication) of inlet flow distortion necessitates precise definition of the inlet instrumentation system. The aerodynamic interface between the inlet and the engine should be defined and maintained invariant throughout a propulsion system development cycle for all testing (inlet, engine, and inlet-engine tests). Recommended guidelines for selection of the aerodynamic interface plane (AIP) are reported in Ref. 1. The AIP criteria should include definition of the probe sensing characteristics and the detailed radial, circumferential, and axial location of each sensor relative to the engine inlet.

Measurement of the propagation of distortion through the engine is necessary to determine distortion transfer coefficients (for analysis of multi-spool compressor systems) and to determine the effects of inlet distortion on control system sensor locations (engine/control system interactions). However, physical installation of the increased number of sensors is generally not feasible in a turbine engine test. A technique to obtain the required increased instrumentation survey is the use of the rotating distortion pattern concept (Ref. 10). The distortion screen assembly is automatically rotated and positioned at discrete circumferential locations during the recording period of a steady-state data record to provide the capability of multiple data values from each installed sensor during each data record (Fig. 9). An effective multiplication factor of about three (the ratio of effective to installed instrumentation sensor locations) was successfully used in a demonstration test program; effective multiplication factors on the order of 10 or more are feasible.

4.3 ANALYSIS METHODS - STEADY-STATE INLET DISTORTION

Computer programs to describe the steady-state total pressure distribution at the engine inlet are useful tools for rapid visual assessment of the flow pattern. The total pressure distortion obtained with a 180-deg one-per-revolution circumferential pattern is described by the engine inlet face maps shown in Fig. 8. Areas of constant pressure (± small tolerance) may be indicated by use of standard printer symbols. Several variations of this concept are in general use by different engine and airframe contractors.

Numerical descriptions or indices to identify the relative severity of distortion patterns are used by most engine manufacturers. The descriptors, coupled with empirically determined compressor/engine distortion sensitivity factors, are used to quantify the effect of inlet distortion on engine stability and performance. A numerical distortion description provides a means of identifying critical inlet distortion patterns and of communication during propulsion system development. The Society of Automotive Engineers, Inc.-recommended distortion descriptor methodology is reported in Ref. 1. The purpose of the inlet-distortion computation procedure is to reduce the inlet distortion pattern (individual pressure values of the aerodynamic interface plane) to a set of numerical values which define the relative severity of the pattern in terms of circumferential and radial pressure defects. These distortion descriptions are combined with engine sensitivities to determine surge pressure ratio loss.

The distortion descriptor elements include definition of the circumferential intensity, $\Delta PC/P$, (magnitude of the circumferential pressure defect); the circumferential extent, θ^-, (angle subtended by a low-pressure region); a multiple-per-revolution factor, MPR, (number of low-pressure regions); and the radial intensity, $\Delta PR/P$, (magnitude of a radial pressure defect).

The descriptive elements are used to define compressor surge pressure ratio loss by the following equation:

$$\Delta PRS = \sum_{i=1}^{N} \left[K_{c_i} \left(\frac{\Delta PC}{P}\right)_i + K_{r_i} \left(\frac{\Delta PR}{P}\right)_i + C_i \right]$$

where

ΔPRS is the loss in surge pressure ratio due to distortion expressed or a fraction of the undistorted surge pressure ratio.

N is the number of instrumentation rings.

K is the generalized distortion sensitivities (empirically determined).

$\Delta PC/P$ is the circumferential distortion intensity.

$\Delta PR/P$ is the radial distortion intensity.

C is an empirically defined constant.

Subscript c refers to circumferential distortion.

Subscript r refers to radial distortion.

Subscript i refers to a particular instrumentation ring.

The sensitivity terms and constant C can vary with distortion (extent, multiple-per-rev, ring weighting factor, etc.) and with compression system design.

A detailed description of the SAE methodology is provided in Ref. 1. Although a universal descriptor that can meet the requirements of every compressor does not appear to be within the state of the art, the generalized approach recommended by Ref. 1 can be expanded to form nearly any distortion descriptor in current use.

Typical test results indicating stability margin with steady-state distortion compared to baseline data obtained with a uniform inlet flow condition at Reynolds number index 0.6 test conditions are shown in Fig. 10. A significant reduction in the available stability margin is shown at all operating rotor speeds.

5. TIME-VARIANT INLET DISTORTION

The techniques used to assess the effects of time-variant distortion on compressor stability are similar, in concept, to the techniques used for steady-state distortion, except that the inlet-engine test environmental conditions are assessed over "short" time-averaging intervals determined to be critical for the analysis of compressor response. The selected time interval must include consideration for all frequency and amplitude components which alter compressor stability or performance; it is generally determined by selecting an "averaging time" which will correlate the compressor response (i.e., loss in surge pressure ratio) with test results obtained with steady-state distortion. Time-averaging intervals corresponding to about one-half to one and one-half engine revolutions are determined for most compression systems (Ref. 11).

The test matrix should be based on the inlet system characteristics and may require consideration for several "types" of time-variant compressor inlet environments:

1. Random frequency pressure fluctuations
2. Discrete frequency pressure fluctuations
3. Composite time-variant pressure patterns

5.1 TEST INSTALLATION - TIME-VARIANT INLET DISTORTION

Testing with random frequency time-variant pressure fluctuations can be conducted with a modified direct-connect installation (Fig. 11). A critical flow convergent-divergent nozzle with a variable position centerbody is used to generate random frequency turbulent flow by interaction of the shock wave and boundary-layer systems in the same manner random frequency turbulence is generated in an aircraft inlet system. The turbulence level is increased by decreasing the nozzle flow area which increases the strength of the shock system. A turbulence attenuation screen located downstream of the turbulence generator may be used to impose a steady-state distortion pattern on the random frequency turbulent flow system. The random frequency generator centerbody may be offset to obtain asymmetric distortion patterns. The time-variant distortion characteristics of the random frequency turbulence generator are similar to the full-scale inlet characteristics if the generator is designed to approximate the length/volume characteristics of the aircraft inlet duct from the inlet throat to the engine face.

Geometric refinement of the technique may be used to simulate specific inlet configurations. The random frequency generator reported in Ref. 12 is designed to produce distortion patterns which are similar in shape, level, and dynamic content to test results obtained from two-dimensional inlet models. Further extension of the test technique to a semi-free-jet system permits the simulation of inlet spillage and bleed flows forward of the engine inlet (Ref. 13).

Discrete frequency pressure fluctuations can occur in an aircraft inlet duct as a result of inlet instability, such as buzz, or as a result of duct resonances (Ref. 14). The propagation characteristics of a discrete frequency, full-face inlet pressure fluctuation through the compressor components may be determined using discrete frequency generators. This type generator is an effective tool for evaluating the validity of analytical models which describe the dynamic behavior of compressors.

Several types of discrete frequency generators have been reported. The airjet distortion generator system designs reported in both Refs. 8 and 9 have discrete frequency pressure pulse capabilities. Test calibration results of a discrete frequency generator designed and fabricated at the AEDC is reported in Ref. 15. The generator consists of a rotor installed between matched stator assemblies. The output frequency of the generator is remotely controlled by varying the speed of the rotor. The amplitude of the pressure fluctuations can be varied by changing the solidity (blockage) of the rotor-stator assemblies. The output of the generator at discrete frequencies of 50, 100, and 300 Hz is shown in Fig. 12 as the normalized peak-to-peak total pressure variation as a function of frequency.

A similar design concept, the Planar Pressure Pulse Generator (P^3G) is reported in Ref. 16. The P^3G is a choked-flow device which uses a single stage rotor and stator combination to sinusoidally modulate the minimum area. The frequency of the planar waves is controlled by the rotational speed of the rotor, and the amplitude of the waves is governed by the rotor-to-stator spacing.

Complete propulsion system (inlet and engine) testing is desired to evaluate inlet-engine interactions prior to flight testing the aircraft system. A typical inlet-engine

test installation in the AEDC Propulsion Wind Tunnel Test Facility is shown in Fig. 13. The AEDC wind tunnel test facilities provide the capability to test full-scale inlet-engine installations over a portion of the design flight envelope. Currently, evaluation of pitch and yaw conditions is somewhat restricted due to tunnel blockage considerations. In addition, the current wind tunnel facilities do not have the capability to duplicate the full-spectrum engine inlet pressure and temperature levels experienced in flight.

The Aero Propulsion Systems Test Facility (ASTF), currently scheduled for completion in 1983, will provide a significant increase in turbine engine test capabilities at the AEDC. A schematic of the proposed freejet propulsion system test configuration is shown in Fig. 14. The ASTF is designed to duplicate inlet-engine environmental conditions of full-scale propulsion systems over the design flight envelope of most current and projected aircraft system designs.

5.2 INSTRUMENTATION-TIME-VARIANT INLET DISTORTION

High frequency response instrumentation systems are required at the compressor inlet station (AIP) to define inlet environmental conditions, and at stations within the engine to determine distortion transfer coefficients and component response to time-variant distortion. It was not until the mid-sixties that the time-variant nature of total pressure distortion was evaluated quantatively (Ref. 17). Since that time, significant advances have been made in unsteady total-pressure measurements, data acquisition, and analysis techniques.

High frequency response total pressure measurements require flush mounted or very close-coupled pressure transducers. Selection or design of the transducer/rake configurations requires consideration for particle impingement and thermal environment effects, aerodynamic (angle of attack/yaw) characteristics, system resonance frequency characteristics, vibration sensitivity, and structural and functional integrity. Absolute pressure measurements are generally not obtained directly from high frequency transducers because of the complex calibration systems and procedures required to obtain accurate measurement with currently available transducers. High accuracy low frequency response (5 Hz), "steady-state" measurements are combined with the lower accuracy but high frequency response of the "dynamic" measurement to provide absolute "instantaneous" pressure measurements. The high response transducer location in the probe is designed to avoid particle impingement. The system resonance frequency should be well above the data frequency range. Resonance frequencies on the order of five times the highest data frequency are generally considered acceptable without significant degradation in the test results.

Close-coupled high response pressure transducers are difficult and expensive to maintain in the high-temperature environment within the engine. A nonresonant (infinite tube) instrumentation system configuration (Refs. 18 and 19) which maintains the desired frequency response characteristics but which allows locating the transducer remotely from the severe temperature conditions within the engine is an attractive alternate configuration for some data requirements. The transducer is side mounted to an infinite tube (a continuous tube on the order of 55 ft long extending from a 0.25-in.-diam total pressure probe) as shown in Fig. 15. The long length prevents reflected pressure signals from reaching the pressure transducer. The transducer is readily available for servicing or inspection. However, analysis of the phase angle characteristics for the system indicate significant phase angle shifts at signal frequencies above 500 Hz.

The nonresonant probe system will serve as an "event indicator," e.g., a system for stall initiation detection, for engine transient measurements, and for spectral analyses from single probes. The system is not suitable for measurements where phase response at high frequency (greater than 500 Hz) is a requirement. Sizable corrections are required in determining "instantaneous distortion" using this system because of the phase lag characteristic.

5.3 ANALYSIS METHODS - TIME-VARIANT INLET DISTORTION

The analysis methods required to evaluate stability margin with time-variant distortion are similar to the basic approach discussed for steady-state distortion; however, more sophisticated data handling and processing methods are required because of the significantly greater data volume acquired. High-frequency response data requirements from a single "data record" or "test condition" may be greater than the data requirements for a complete test phase requiring only steady-state performance measurements. Rapid data quality authentication and data compacting techniques are clearly necessary. Real-time analog processors have been developed (Refs. 20 and 21) to calculate distortion descriptors which may be used to identify the time at which maximum distortion occurs. For selected test conditions of interest, short time segments of the data record containing the observed analog peak distortions may then be digitally processed to achieve greater accuracy than available from the real-time processor and to obtain the inlet-engine pressure profiles at the times of the peak distortion levels.

Several techniques have been developed to assess maximum distortion levels with limited high-frequency response instrumentation and/or with minimum data (test time) records. Although these techniques are primarily designed for use during inlet development programs, they may also be used during test assessments of full-scale turbine engines with random time-variant inlet flow distortion. Use of an analysis technique, based on extreme-value statistics, is reported in Ref. 22. The technique can be used to statistically predict the expected maximum distortion level corresponding to any time period of inlet operation, including estimates of the prediction tolerance, from a short time

segment of distortion data. Data acquisition time periods of approximately 2 sec may be used to assess maximum distortion levels for stationary test conditions.

Other techniques which address the analysis of time-variant distortion level or pattern, or the loss in surge pressure ratio, are reported in Refs. 23 and 24.

Typical data describing instantaneous distortion characteristics are shown in Fig. 16. The deviation of average (spatial) face pressure from the steady-state value (spatial and temporal averaged) provides an indication of in-phase, full-face pressure variations. The history of the inlet distortion index parameter, K_D, provides a tool for the correlation of pattern severity with duration. The distribution function data shown is used to statistically represent a data record history and to indicate probability of obtaining desired test conditions or instantaneous distortion index levels.

6. STABILITY CERTIFICATION/QUALIFICATION TEST METHODOLOGY

Engine stability certification or qualification testing is required to provide a demonstrated quantitative evaluation of stability margin at specified engine and environmental conditions. A typical format illustrating the effects which may require consideration, and the estimated margin allocations, is shown in Fig. 17 (Ref. 11). A number of the destabilizing effects are directly added, while others are statistical in nature and are combined in a "Root Sum Squared" (RSS) method. The presently accepted method of interpreting the assessment is to add directly the totals of the direct additive and RSS items and subtract this total from the available margin.

The margin requirements for each factor of the assessment must be derived using stability data obtained during engine development prequalification testing in conjunction with qualification engine test results; qualification test engines are not normally subjected to intentional surge because of the hazard of structural damage and the resulting unnecessary delays in engine qualification testing. The compressor surge line and total margin requirements must be determined during engine development and prequalification testing conducted with qualification configuration compressor components. Limited verification of the operating line excursions caused by the primary factors such as control requirements and distortion effects may be obtained during the qualification test. In addition, engine operation in the region above the maximum predicted compressor operating line, but below the predicted surge pressure ratio, should be accomplished whenever practical.

A test methodology to demonstrate remaining margin is illustrated in Fig. 18. In the example shown, an equivalent maximum operating line is established to account for the estimated internal effects (engine quality and deterioration) margin allocations of the stability assessment. By using the equivalent maximum operating line, a portion of the remaining margin (estimated net margin to assure stable operation) can be demonstrated during testing (Fig. 19).

7. CONCLUSIONS

The results of testing at the Arnold Engineering Development Center have shown that established test techniques are available for the systematic and quantitative assessment of turbine engine stability margin on a building block concept. With defined engine and environmental boundary conditions, engine stability can be assessed in the same quantitative manner that steady-state engine performance is evaluated.

A test matrix rationale has been established for the systematic acquisition of test data to define the stability margin of current augmented turbofan engines as a function of engine operating conditions and environmental factors which are representative of engine specification requirements and/or inlet-engine interface conditions.

Proven steady-state and dynamic pressure measurement systems have been demonstrated which are compatible with the total operating environment with a quantity and placement rationale.

An inventory of proven test equipment designs has been established which permits a building block concept for the systematic examination of different phenomena on engine stability margin.

Component loading techniques have been demonstrated which permit matched and unmatched component operating line excursions with a standard engine configuration.

Standard data analysis methods are available for the quantitative assessment of engine stability margin and for the definition of environmental conditions representative of inlet-engine interface conditions.

A recommended test methodology, applicable to inlet-engine compatibility certification/qualification testing requirements, has been demonstrated.

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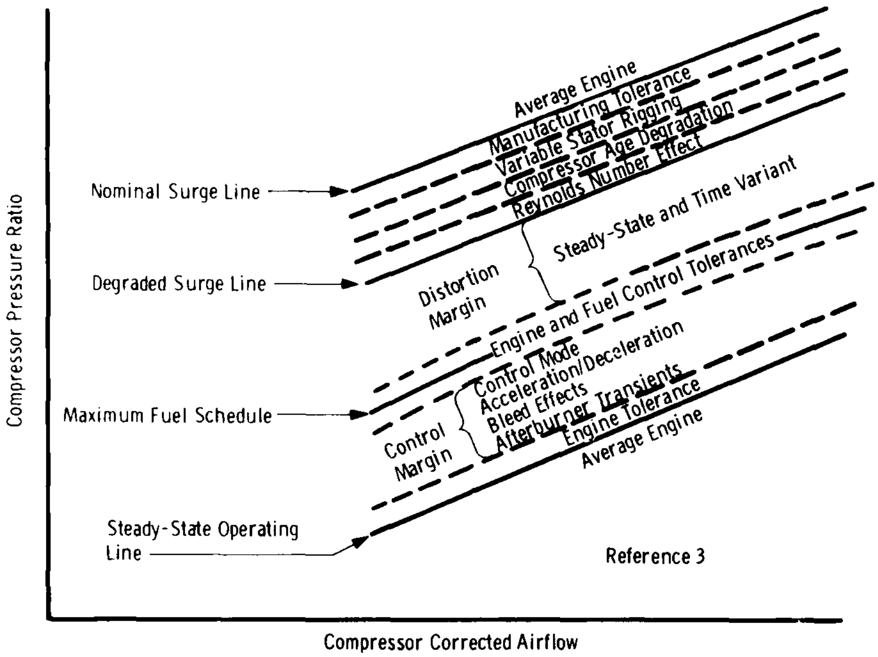

Figure 1. Cumulative Representation of Degrading Factors on a Compressor Performance Map

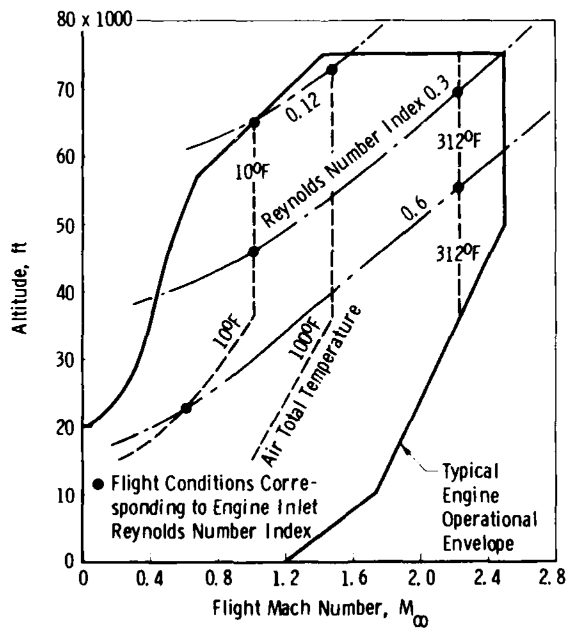

Figure 2. Typical Test Condition Matrix for Investigation of Altitude and Mach Number Effects on Stability Characteristics of a Medium-Sized Engine (2- to 3-ft Inlet Diameter)

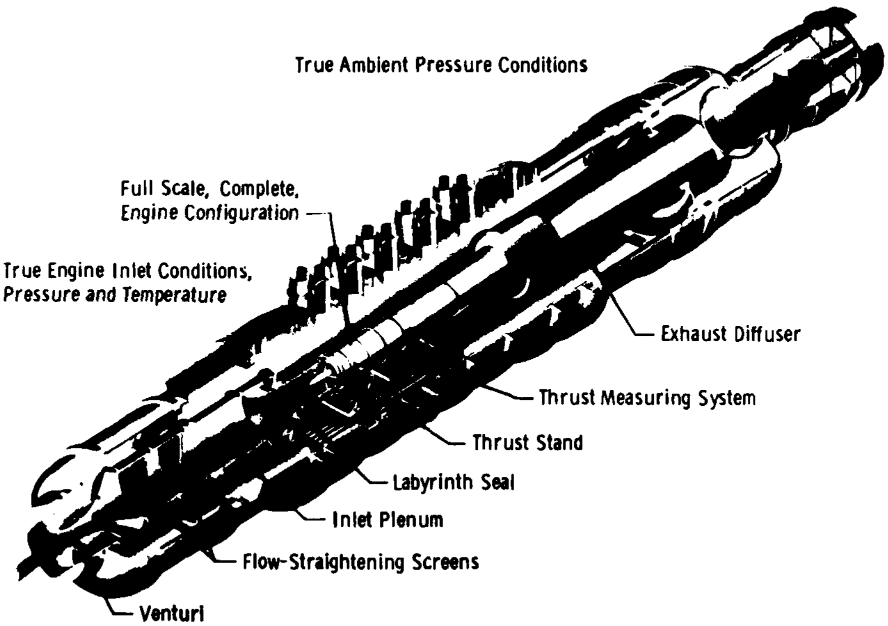

Figure 3. Typical Direct Connect Turbine Engine Test Installation Configuration

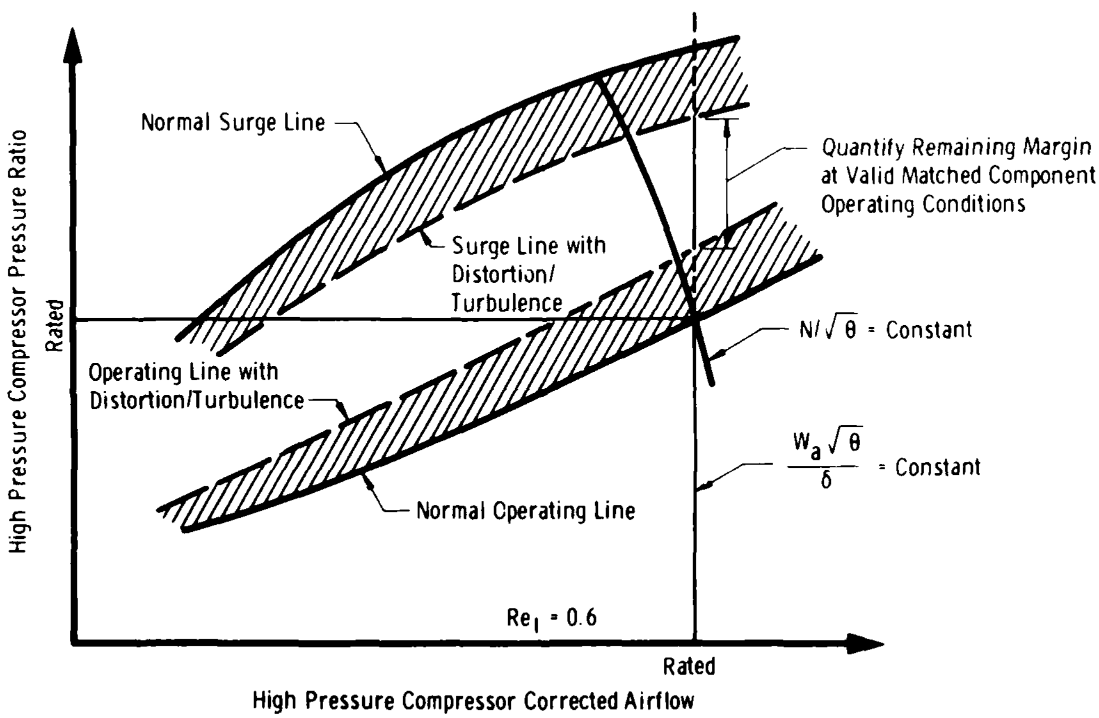

Figure 4. Simultaneous Loading of Fan and Gas Generator Components to Assess Stability Margin

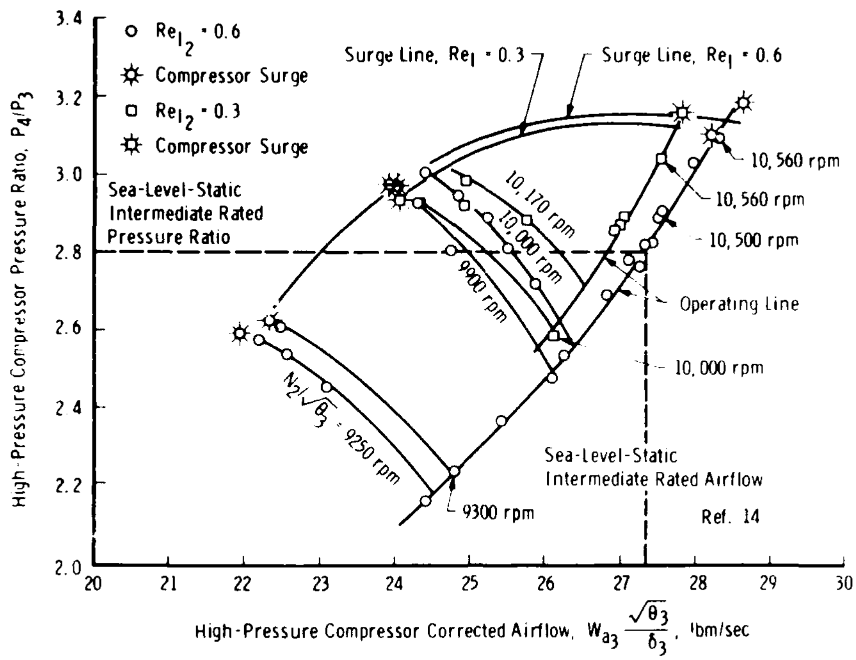

Figure 5. Effect of Reynolds Number Index on Compressor Component Stability Margin

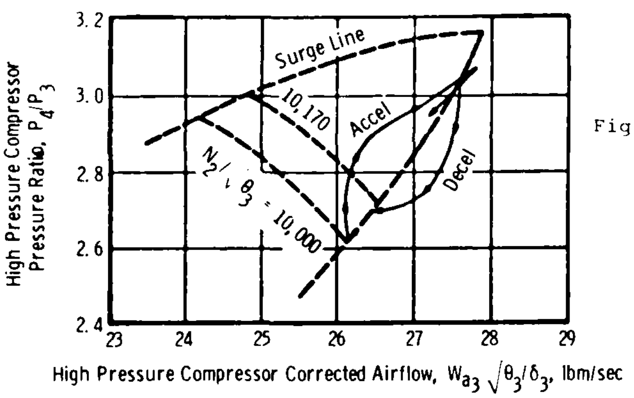

Figure 6. Typical Component Operating Point Excursion During Transients Between Idle to Intermediate and Intermediate to Idle Power Transients at $Re_I = 0.3$

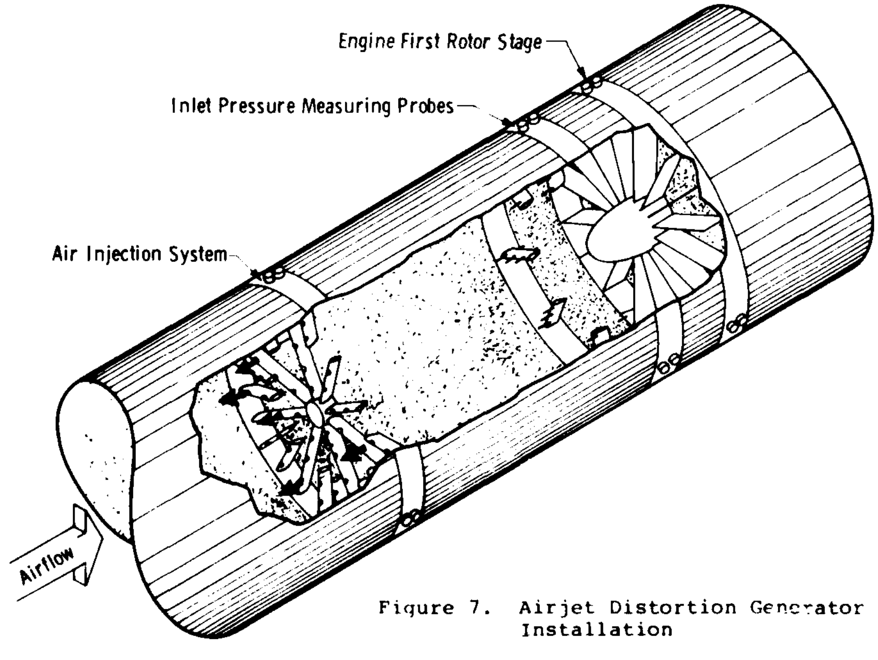

Figure 7. Airjet Distortion Generator Installation

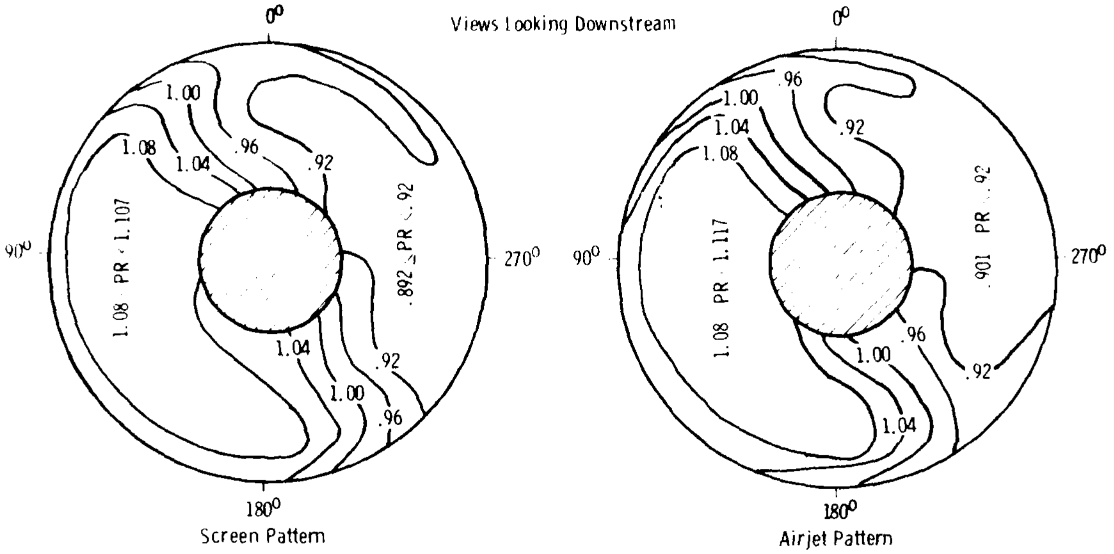

Figure 8. Engine Inlet Isobar Maps for Airjet and Screen Produced 180-deg, One-per-Revolution Pattern

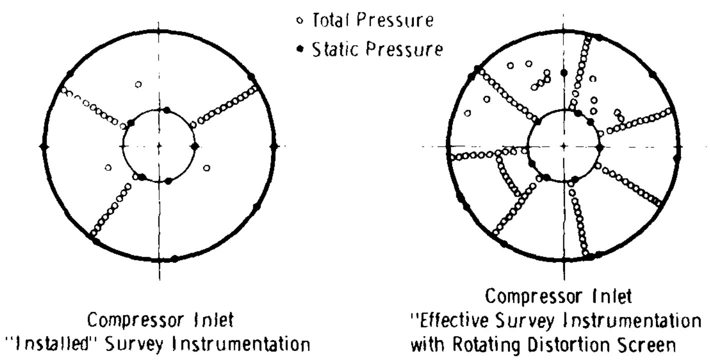

Figure 9. Improved Survey Instrumentation Capability with Rotating Inlet Distortion Pattern Concept

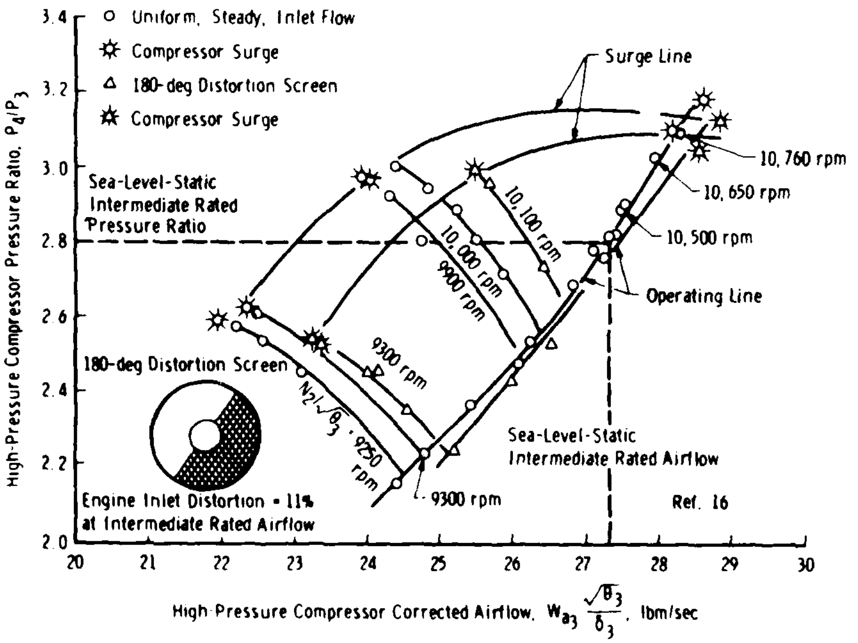

Figure 10. Effect of Steady-State Spatial Distortion on Compressor Component Stability Margin, Re_I = 0.6

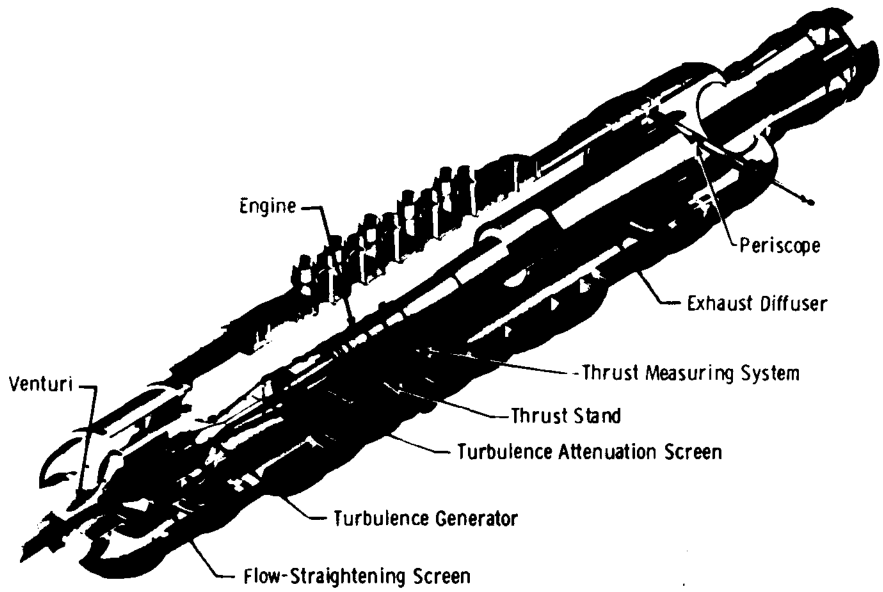

Figure 11. Typical Test Installation Configuration for Investigation of Engine Stability Margin with Time-Variant Inlet Distortion

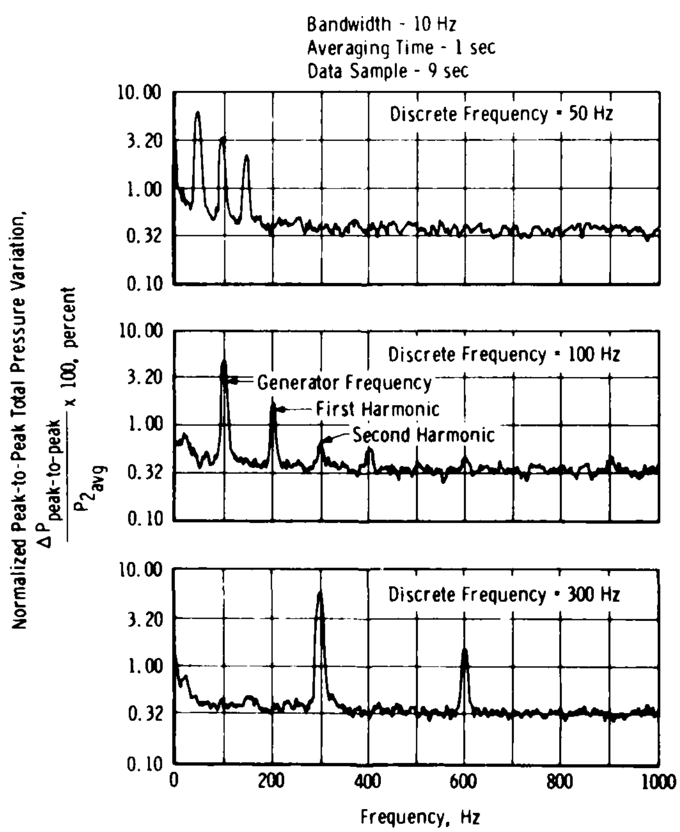

Figure 12. Typical Fourier Function for Engine Inlet Pressure Measurements Behind the Discrete Frequency Generator, $Re_I = 0.3$, $W_a\sqrt{\theta_2}/\delta_2 = 237$ lbm/sec, $P_{2_{avg}} = 3.85$ psia

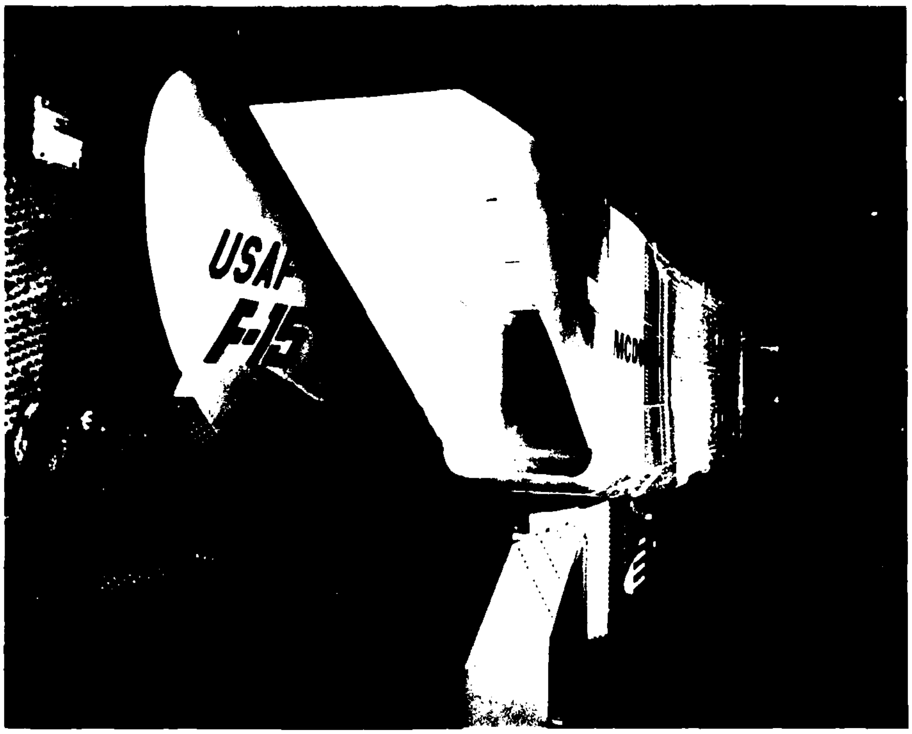

Figure 13. Propulsion System Test Installation in Propulsion Wind Tunnel Test Facility

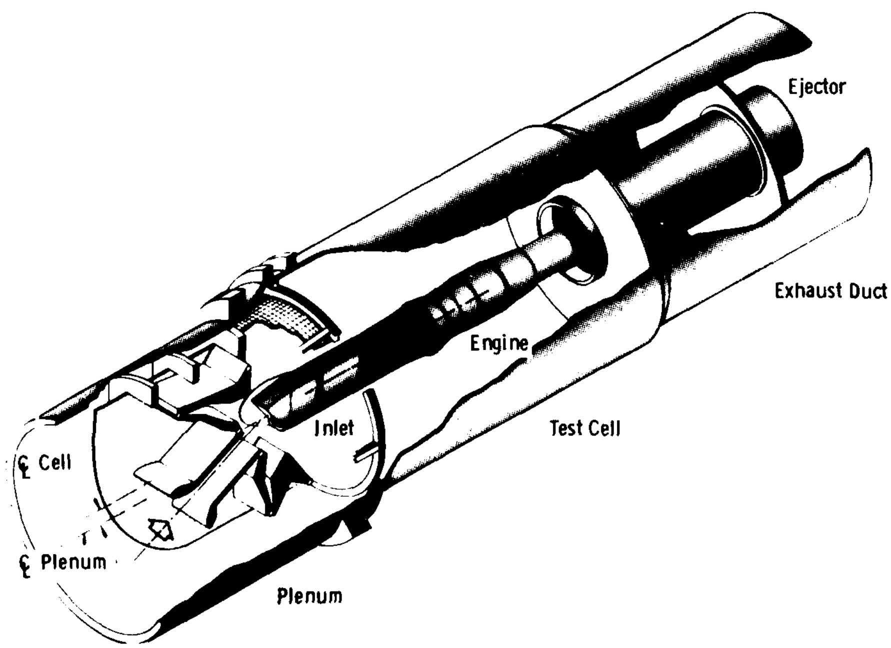

Figure 14. Schematic of Free-Jet Propulsion System Test Installation in Aeropropulsion System Test Facility

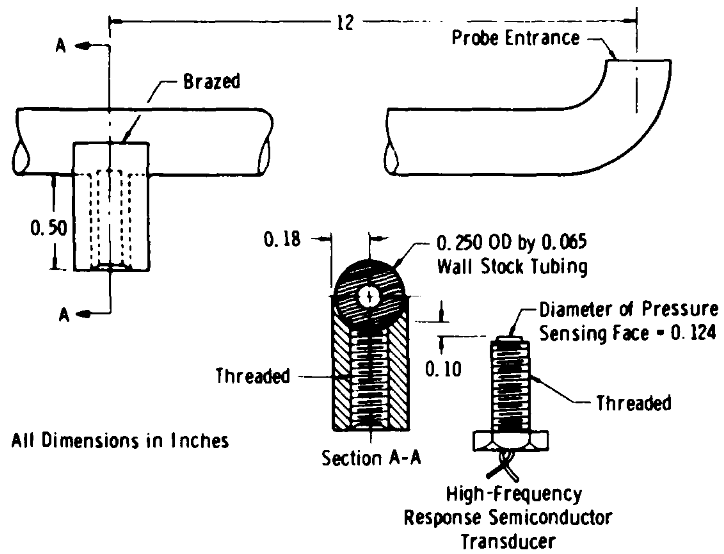

Figure 15. Nonresonant High Frequency Total Pressure Probe Configuration

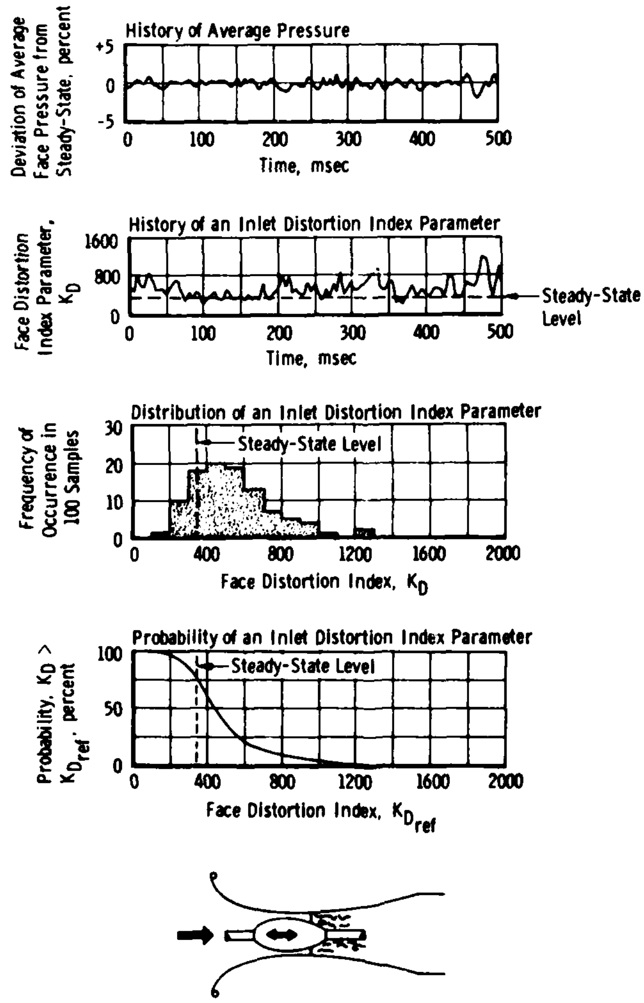

Figure 16. Time-Variant Inlet Flow Characterization Technique

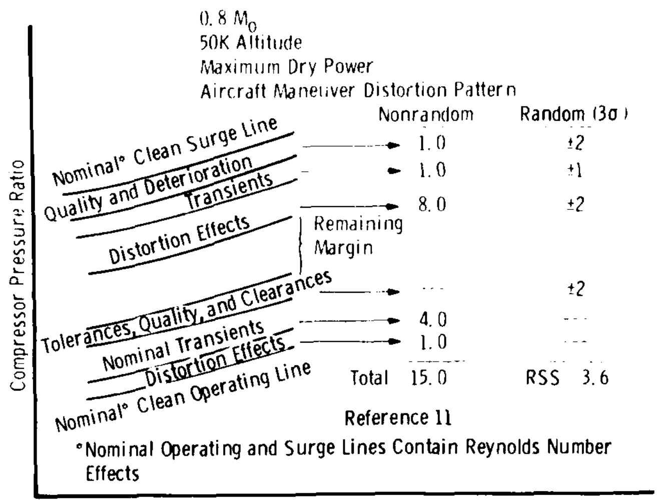

Figure 17. Typical Stability Assessment Format

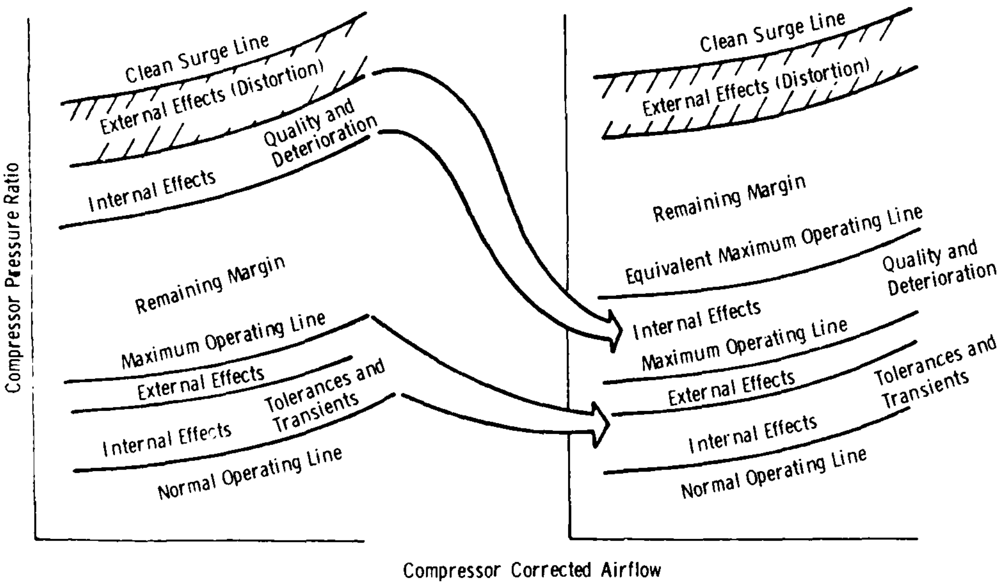

Figure 18. Stability Assessment Test Verification Methodology

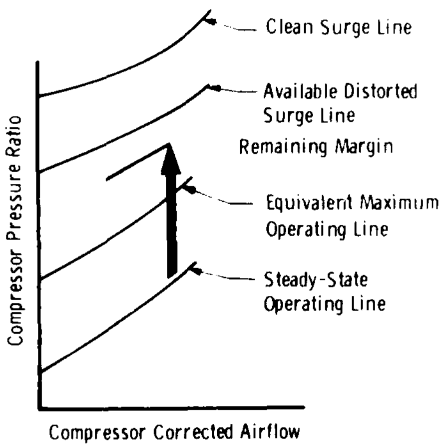

Figure 19. Test Procedure to Demonstrate Available Margin

DISCUSSION

P.F.Ashwood, NGTE, UK

What is the turbulence level in a direct-connect test installation at the engine face, and does the choked venturi air measuring device cause additional fluctuation?

Author's Reply

The turbulence or time-variant-distortion level in the AEDC direct-connect test installations is on the order of one percent or less, characterized by $\Delta P_{rms}/P_{AVG}$ (0 700 Hz) at the engine inlet. This is the target level for uniform, steady engine inlet flow quality with or without the air flow measuring venturi installed.

GAS TURBINE ENGINE TRANSIENT TESTING

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SUMMARY

To assist the Canadian Armed Forces in resolving engine operational problems in their CF-5 aircraft, the Engine Laboratory of the National Research Council of Canada has been conducting extensive steady-state and transient-performance tests on J85-CAN-15 turbojet engines. As the test cell was originally equipped only for sea-level-static steady-state tests, instrumentation and techniques were developed to monitor and record experimental data rapidly and accurately during rapid-transient engine operation. The current technique provides report-quality time-plots and compressor operating lines immediately after test, thus permitting rapid assessments of engine performance.

INTRODUCTION

During the early part of the seventies, the Canadian Armed Forces acquired the CF-5 fighter aircraft, a modified, Canadian-manufactured version of the Northrop F5 (Figure 1). The aircraft is powered by two General Electric J85-CAN-15 turbojet engines, built under license by Orenda Limited. Not long after the introduction of the CF-5 into service, reports of in-flight compressor stalls and combustor flameouts were received from the bases. It soon became apparent that the problems could not be traced simply to a single cause, and in 1974 the National Research Council of Canada (NRCC) was approached for assistance.

The Canadian Forces and NRCC agreed to attack the problems by comparative assessments of different engines, engine components, and engine adjustments during various transient power manoeuvres, under sea-level-static conditions. Since the Engine Laboratory's test cell selected for the test program was then instrumented only for steady-state testing, the instrumentation and techniques for the monitoring and reduction of transient data had to be developed.

This paper provides the background and defines the requirements of the test program, and specifically discusses the evolution of the transient data acquisition and reduction methods and instrumentation to the present state.

PROJECT BACKGROUND

Description of Engine and Problems

The J85-CAN-15 is a single-spool afterburning turbojet engine, consisting of an eight-stage axial flow compressor, an annular combustor, a two-stage turbine, a diffuser casing, and an exhaust section (Figure 2). A variable-geometry system controls the angle of the compressor inlet guide vanes and the position of two interstage compressor bleed valves to provide off-design compressor stability. The exhaust section contains an afterburner terminated by an infinitely variable exhaust nozzle that is fully modulated throughout the engine operating spectrum.

Reports of operational problems with the engine, experienced at Canadian bases, seemed to fall into certain patterns. For one, compressor stalls and combustor flameouts were reported more often in wintertime than in summer, leading to the assumption that internal engine controls did not respond properly to low ambient temperatures. Then, a batch of engines from the "new production series" seemed to have a greater stall and flameout propensity than those belonging to the "old production series". While nominally no differences were to have existed between the two types, the Canadian contractor admitted to certain changes in compressor stator material and to design modifications of some components. Finally, more problems with the engine were encountered by the training squadrons than by the operational squadrons at each base. Hence, the type of manoeuvres flown and the level of pilot training appeared to have some influence on the occurrence of engine misbehaviour. In some cases, the compressor went into stall only briefly, recovering fairly quickly, while in others the main combustor and/or the afterburner flamed out. It was against this background that the investigative programs at NRCC were planned.

Engine Test Programs

As mentioned earlier, only sea-level-static test capabilities exist at the Engine Laboratory, NRCC. This limitation precluded any investigation in which an engine could be subjected to simulated flight conditions under which the incidents occurred. Consequently, it was decided to establish through comparative testing whether a particular engine or engine group was more, or less, stall prone than another, whether a new component performed better, or worse, than the original one, or whether a particular engine adjustment or setting would be beneficial to engine performance.

Furthermore, some testing took place simply to establish whether a particular component functioned as indicated in the operating instructions or as claimed by the manufacturer.

Initially, complete engines were tested, e.g. a "new production series" engine followed by an "old production series" one. After some baseline steady-state and transient testing during which normal engine operation and performance were established, a series of transients were run in which the compressor inflow was statically distorted with an adjustable flat plate. The degree of distortion, correlated with an engine's reaction in form of stall and/or flameout, gave a qualitative assessment of the engine's proneness to stall in relation to another. The method's strength was its simplicity.

To facilitate the test programs, one engine was retained as the test vehicle, on which only certain components were exchanged. While all other conditions were kept as constant as possible, the consequent differences in engine behaviour were recorded and evaluated. Among the components thus tested were "old and new production series" T5 amplifiers and VEN power units for comparative evaluation, and a P3 dump system for comparison with simple engine anti-icing bleed. The consequences of a faulty T2 heater valve and the effects of deliberately downtrimming the engine were quantitatively assessed. Finally, considerable effort was spent on investigating the behaviour of the T2 speed cutback at low inlet temperatures and the CIT/T2 sensing system for heat transfer and, hence, temperature distortion.

Such a diverse series of test programs constantly demanded improvements in transient data acquisition equipment and techniques. In the following sections the evolution of the system is briefly traced, from the rather primitive, mostly manual, data reduction methods based on simple strip chart recordings, to the present system; we have greatly increased the accuracy and reliability of handling transient data at vastly increased speed, due to the virtual elimination of manual work input.

DEVELOPMENT OF TRANSIENT INSTRUMENTATION

Engine Test Installation

The Engine Laboratory's No. 5 test cell is a straight-through flow type, with the test section measuring 5 m by 5 m in cross section (Figure 3). Sound attenuation is achieved by a set of inlet-silencing splitters and an outlet silencer. The test engine is mounted on a frame which is suspended on flexure plates, permitting thrust measurement by means of a hydraulic or electric force cell. Direct observation during a test run is obtained through a window at the control room or through closed-circuit television.

Steady-State Data Acquisition and Reduction

Prior to the J85-CAN-15 project, No. 5 test cell was instrumented for steady-state testing only. Instrumentation consisted, typically, of manometers, pyrometers, voltmeters, pressure gauges, and digital indicators for fuel flow and gas temperature. In addition, the engine operator had some aircraft cockpit instruments for engine operation, rather than for performance monitoring. Raw data were recorded manually and reduced with the help of desk calculators. The reduced data were then hand-plotted for analysis. This procedure was quite acceptable as long as the volume of steady-state testing was not excessive.

Basic Transient Data Acquisition and Reduction Requirements

With the demand for transient engine testing, new methods for data acquisition and reduction had to be developed. A functional overview of the data reduction problem is shown in Figure 4. Information from the engine is sent to the visual display box, which has two functions. One is to provide a permanent record of the transient data and the other is to provide steady-state calibration data to enable subsequent data reduction. The raw transient information has to be digitized by some process, either manual or electronic, then converted to basic engineering units, using the signal calibration data. Certain parameters such as aerodynamic flows and corrected aerodynamic speeds are also calculated from the basic data. The final results are presented in the form of time-plots, cross-plots, and printouts.

Strip Chart Recording/Manual Reduction System

The first transient data acquisition and reduction system used a strip chart recorder for data recording, with subsequent data reduction done by hand. The ten basic variables to be measured were first identified, then wide band width transducers (e.g. pressure, position, force, and flow) were mounted as close to the engine as possible in order to obtain high-frequency response signals. Analog voltages from the transducers were connected through signal-conditioning circuitry to an eight-channel and a two-channel strip chart recorder to provide analog time records of a particular transient manoeuvre. A "blip" channel was also required to correlate time between the two recorders. Although this method supplied a reasonably accurate record on paper, the calibration procedure for relating engineering units to voltage levels on the strip chart, and the subsequent reduction of time traces to meaningful engineering quantities, required a considerable amount of manpower. Several transients, each typically 15 seconds long, were produced during a two-hour test run; the data reduction of each transient then took approximately two man-days. If during the course of a test a transducer produced erroneous readings, the fault often could not be recognized until the data were actually reduced. In many instances, the time interval between test and data analysis precluded a repeat test at the same environmental

test condition. Although these data reduction methods produced fairly accurate results, they were very laborious and presented a major bottleneck in the progress of the program. The solution to the problem was the development of an automated data reduction system.

Magnetic Tape/Hybrid Computer System

A major improvement to the acquisition/reduction procedure was made by recording the raw information on analog magnetic tape and processing the data on the hybrid computer facility at the NRCC. By replacing the engine block in Figure 4 with F.M. magnetic tape, the information flow could be duplicated on a hybrid computer. The analog portion of the hybrid computer contained the necessary hardware for signal conditioning, data display, and A/D conversion, while the digital portion controlled the data flow, processing, storage, and printing. Visual display of the data from the magnetic tape, from the output of the signal conditioning box, and from the output of the A/D converter block was essential in order to ensure that the data from the tests had not been corrupted by the reduction process. The software package, which replaced the all-manual activity from A/D conversion down, had to be interactive and easy to use, and had to be perceived by the users as doing functionally exactly what they had done by hand.

Program Design

Once a clear understanding of the data reduction process was obtained, a distinct program design phase was begun. Functional duties of the program were mapped out, such as reading calibration data, generating calibration curves, reducing raw data, printing, and plotting. The arrangement of data structure and flow of data between the major program modules is shown in Figure 5. Each rectangular box in this figure indicates a major functional module of the program; the ellipses represent data bases that are handled by the various blocks of code. Because most of the problems encountered in developing programs of this sort are related to the manipulation of data, inspection of data is the only possible means of program verification. Therefore, Figure 5 is a critical figure in the design process; a great deal of time and effort was expended on this diagram, to insure efficient specification of data base structure for the code. A clear understanding of the kinds of data being handled was imperative, so that they could be separated into different groups on the basis of their physical attributes. For example, raw data from the A/D converters are distinguished from their counterpart in reduced engineering terms, and problem constants are distinguished from scale factors. The A/D converters are a form of hardware that can be thought of as locations at which information can reside, i.e. a data base. This diagram also treats the human operator as another data base.

The human operator provides the source of program control by interacting with the executive. Each functional block is a separate subroutine, and control is always returned to the executive upon completion of a task. The executive then interrogates the operator as to his next desired function. Upon receipt of the information from the human operator, the executive then schedules that task and control is transferred to that block of code. Some argument took place among the program designers concerning the feasibility of completely automating this process. However, it seemed imperative that control of the data reduction process remain in human hands. This technique allows the operator to inspect specific results, stop the program at any point, or return to and repeat a previously completed function. Also, the selection of particular transient runs is made easy because the human operator has manual control of the tape recorder. In this way, transients that had been discarded by test personnel could be disregarded by the data reduction people before they wasted valuable human and machine time trying to reduce them. For these reasons, program control remained with the operator.

Signal Conditioning Problems

Two types of signal corruption were encountered while developing this program. Figure 6 shows an example of both of these forms of corruption. Figure 6a illustrates the noise that appears on magnetic tape for various reasons. It may be caused by noisy instrumentation or difficulties with recording the information on magnetic tape. The second form of data corruption is shown in Figure 6b. It is an inaccuracy known to test personnel and is caused by the large dynamic range of the required instrumentation. At low flow conditions the output of flow-measuring devices such as turbine wheels becomes somewhat oscillatory, the frequency of which is dependent on the flow rate.

Appropriate signal conditioning had to be developed to handle these two different forms of data corruption. The analog portion of the hybrid computer provides a very flexible design tool for developing signal conditioning circuits. The availability of strip-chart recorders and display scopes for monitoring the results of the signal conditioning provided invaluable insight into the design process. White noise on a particular channel, for example, was easily eliminated by using low pass filters. The varying frequency hunting that was exhibited by the flow measurement was, however, a much more difficult problem to solve. Here, the solution was found by reviewing the way data reduction had been initially handled by hand. Because the oscillation was a known characteristic of the measuring device, the person reading the raw data from the strip-chart recorder output simply smoothed out the data by eye. Because the frequency depends upon flow rate it followed that this characteristic could be removed automatically by using some form of adaptive filtering; however, in this particular case, because the level of fluctuation dropped sharply once the throttle lever was advanced, it was decided simply to begin the record of the transient only after motion of the throttle lever was detected.

If the frequency of noise is discernible and not randomly generated, it is important to determine whether the noise is a real phenomena of the engine, a function of the sensor, or some other cause. One interesting example of signal conditioning occurred in the measurement of engine thrust. A strong 18 Hz signal was measured from the thrust transducer when the engine was operated at high power. As this frequency was in the band width of interest, a low pass filter with a low, sharp cut-off frequency could not be used. Secondary engine indicators such as nozzle position did not correlate to the thrust oscillation. Investigation with a frequency analyzer showed that the thrust bed was ringing at its own natural frequency of 18 Hz. The solution to this noise problem was first to stiffen up the thrust frame thereby increasing its natural frequency to 30 Hz, then to remove the remaining noise with a six-pole filter set to a cut-off frequency of 26 Hz.

Data Compression

The second major step in the development of this program was in designing a method for compressing data from the transient records. Data compression was necessary for several reasons. Some transients were slow-throttle manoeuvres from a condition of engine idle up to maximum thrust over quite long periods of time (up to 60-100 seconds to complete the manoeuvre). Other tests involved throttle bursts from part power to full power, which were completed in as little as 5 seconds. Figure 7 shows a typical trace that had to be handled by the software for data reduction. This figure shows that data acquisition based on a fixed time-step size was not a very efficient use of either the computer time nor the storage space available. Also, data peaks or dips could be missed if the sample rate was not sufficiently high.

A system for data compression that considered a tolerance between the current reading and the previous reading for each channel of information being processed was developed. If the difference between these two readings was within the tolerance, the information was discarded. If, however, it exceeded the tolerance, the new information was stored together with the value of time at that point. Many trade-offs in efficient use of storage with this technique are possible. If time is maintained as a separate record for each channel of information, the tolerances could be used to compress each channel of information to its minimum. This procedure would of course require the storage of an array of time points for each channel of information. If, however, any particular channel could trigger the storage of the current reading of all channels, then a single record of time for all channels could be stored. For our engine tests, it was quite efficient to allow any channel to trigger the storage of the current reading; therefore we maintained a single time record. If, however, one were to encounter a wide difference in frequency content between the signals being processed, separate records of time should be maintained, if not for each channel, then at least for separate frequency bands.

Once the logic of the data compression techniques was established, the tolerances were tuned to provide adequate representation of the transient data without storage of excessive information.

Instrument Calibration

Substantial effort was expended in ensuring accurate calibration of the instrumentation. In the hostile environment of a gas turbine engine, deterioration and shifts in transducer calibration caused by vibration and temperature fluctuations are expected. Therefore, complete-system calibrations were done at the beginning of each day of testing, using the engine as the signal source for all the sensors, and a final check was made at the end of the test. Steady-state readout instruments, calibrated independently on a regular basis, were used for on-line calibrations. Before reducing the data obtained between these calibration runs, the calibration points were plotted using a least-squares curve fit of the data and compared against previous tests. A typical instrumentation-calibration plot is shown in Figure 8. The curve obtained from the current calibration points was compared to the previous fit to monitor the condition of the instruments. The figure shows two classes of error that have been encountered in the course of testing engines. The first is the occasional "bad point", obtained when the human operator misreads a gauge, or when a voltage spike appears on the data transmission lines. The second is a DC shift of the output occurring sometime during the course of the test; it results in a major deviation of all the calibration test points. This kind of error usually indicates a major fault in an instrument. Such information is crucial to the quality of the final test results and depends largely on visual treatment of the data by people with extensive experience.

The development of software for use as a tool by an experienced individual required a clear picture of the tasks involved as perceived by the user. Obviously, the system had to be able to generate a curve fit to the data points using a procedure that provided the same degree of visibility as previous hand techniques. The calibration curves could then be reviewed relative to the data from which they were generated. A mechanism to remove bad points as well as to recover original data in case the wrong point was removed was also required. The piece of software accordingly developed fitted a curve to the raw data obtained from the magnetic tape. This routine provided digital plotting of both the data and the curve, with the option of either automatically removing the worst point relative to the curve, or selecting specific points that the operator wanted removed. This module proved to be a powerful tool and was used with ease by persons totally unfamiliar with computers. It also provided rapid feedback to the test personnel concerning instrumentation problems or reading errors that occurred during the tests.

CURRENT DIGITAL DATA SYSTEM

The magnetic tape/hybrid computer method of data acquisition and reduction decreased the processing time for each transient from two man-days to 15 minutes. However, computer time on the hybrid system, located in another laboratory, had to be scheduled a week in advance. Another drawback was the F.M. tape recorder; it was limited in the number of input channels, and was quite cumbersome to carry back and forth between laboratories on a regular basis.

Because the cost of digital computers had decreased substantially since the project was initiated, the Engine Laboratory decided to install a dedicated minicomputer-based data acquisition system in No. 5 cell. Hardware components were purchased that achieved functionally the same things as did the magnetic tape/hybrid computer. Even the software, which was written in Fortran IV for the hybrid computer, was directly transferable with minimum change. The same degree of operator interaction designed into the hybrid computer system was retained as a desirable feature, but the workload was reduced by eliminating the need for a tape recorder. The layout of the hardware is shown in Figure 9. All components, with the exception of the X-Y plotter, were purchased from the Digital Equipment Corporation. Figure 9 shows two separate computers: a PDP11/03 connected to an X-Bus, and a PDP11/34. All the data acquisition and processing is controlled by the PDP11/03, which is located in No. 5 cell, while plotting and printing is done with the PDP11/34, which resides in a central location within the laboratory. Reduced data can be displayed on the video terminal in No. 5 cell several seconds after a transient has been recorded, to confirm that all transducers are operating satisfactorily. The reduced data are stored on magnetic floppy disks that are manually carried to the PDP11/34, from which time-plots and cross-plots are generated. Figure 10 is a photograph of the PDP11/03 minicomputer boxed in the large cabinet containing the data acquisition subsystem, the video terminal, and the floppy disk drive. Figure 11 shows the test cell control panel. The instruments for operating the engine are to the left, the steady-state readout instruments in the centre, and the signal conditioning equipment to the right. The manometer bank on the extreme right remains mainly for posterity, but it is still used to calibrate the low-range pressure transducers.

PRESENTATION OF RESULTS

A digital drum plotter, an analog X-Y plotter, and a line printer were available as media upon which to present results of the data reduction process. Generating printouts of the time histories of the reduced data was first thought to be a major use for the data reduction package; however, the usefulness of these data was limited. The printout only provided the test engineers with a check on the quality of the data relative to runs previously done without the aid of a computer. A package was developed to automatically scale the results and plot them individually on the drum plotter as a function of time. This routine produced acceptable results; however, it remained a relatively slow method. A second package was developed for plotting results on the X-Y plotter through D/A converters, with scaling established by coefficients in the input data set. The time required to plot results by this method was limited only by the response of the plotter.

By plotting five or six traces on a piece of preprinted paper, we could obtain a complete time history of the engine in a matter of several minutes. In addition, the cross-plots, displayed in the form of compressor operating lines, provided a useful method for assessing engine performance. Figure 12 shows one of these plots. Parts a and b of Figure 12 are time histories of the measured and computed engine variables, beginning with the initial movement of the power lever angle. Only 6.8 seconds of data are plotted, but approximately 12 seconds of data are stored. The operator can easily plot all the data, if desired, but usually only the first 5 or 6 seconds of the transient is of interest. For cross-plots, as for example the compressor operating line in Figure 12c, all the data points are usually plotted, but again, the quantity of data displayed is under the operator's control.

The number of data points required to produce a 12-second time-plot of this quality is 700 samples per trace. The sample rate is set at 90 scans per second. Therefore for a 12-second time-trace, 1080 data points would be needed; however, the data compression technique reduces the data storage requirements by 380 samples per trace.

One of the operational problems the Canadian Forces commonly encounter is a high incidence of compressor stalls during cold weather. An illustration of stall margin degradation as a function of compressor inlet temperature (CIT) is shown in Figure 13. Both the steady-state and transient operating lines move toward the reference stall line as the inlet air temperature decreases. This upward shift was readily measurable even though the CIT difference was only 6.2°C. This example illustrates the resolution that can be obtained with the data acquisition system. The production of report-quality plots immediately after each engine test series has greatly reduced the time required for evaluating the effect of engine modifications on engine performance.

CONCLUDING REMARKS

(1) A test cell at the Engine Laboratory has been equipped to accurately test the relative performance of gas turbines undergoing rapid changes in transient power levels.

(2) A software package has been developed that is highly flexible and easy to use, with the added advantage of being transferable to any other computer that accepts Fortran Language.

(3) The digital data acquisition system provided the necessary sampling speed to digitize 11 channels of engine transient data.

(4) Calibrations of all transducers can be verified, and if necessary, calibrated on-line.

(5) The turnaround time for processing data has been reduced from two man-days to several minutes per transient.

(6) An example presented shows the relative ease with which performance assessments can be made.

The author gratefully acknowledges the support of the Department of National Defence without whose assistance this facility would not have been possible.

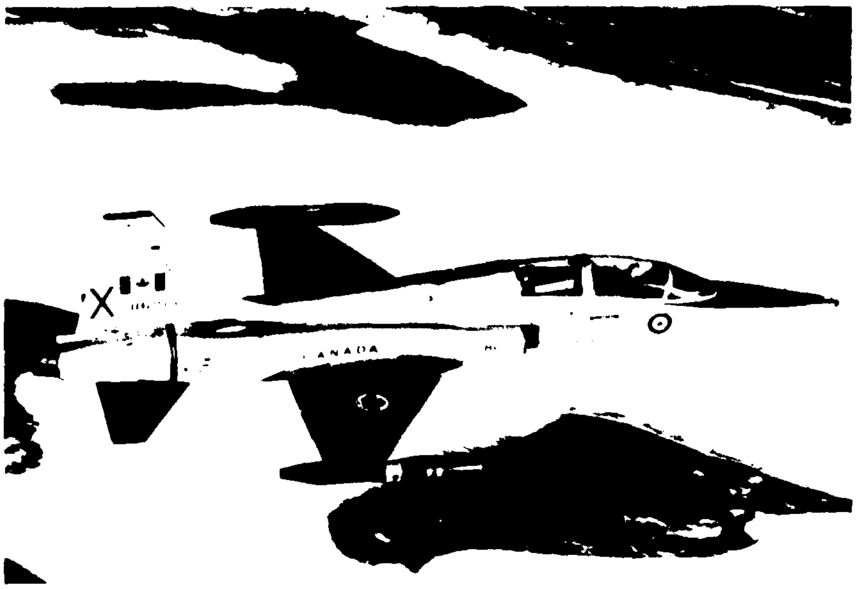

FIG. 1: CAF CF-5 AIRCRAFT

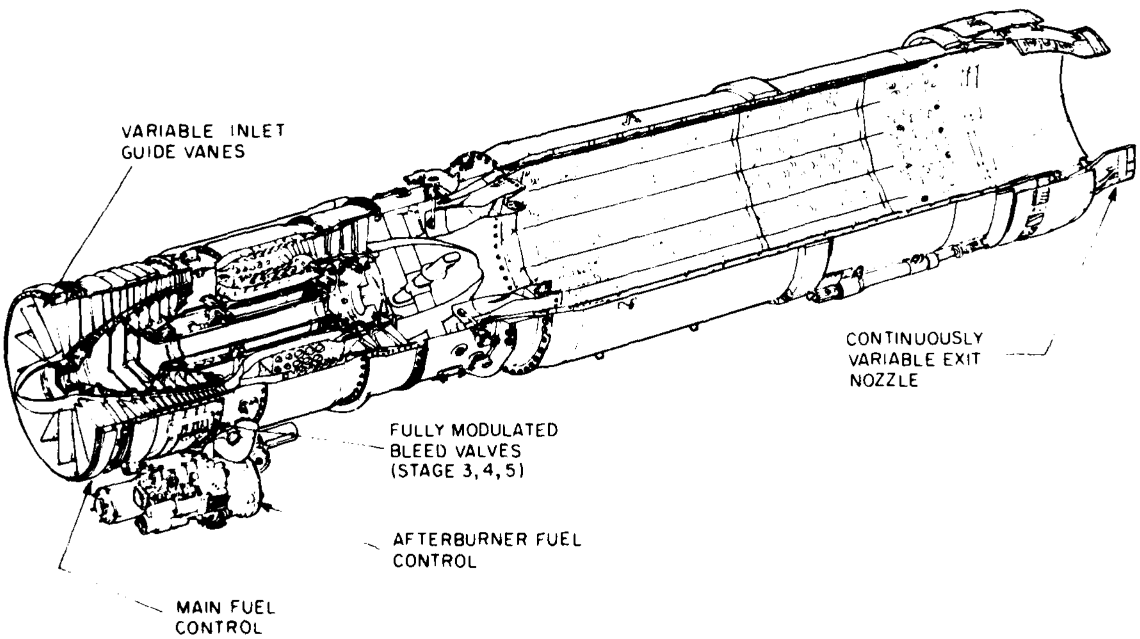

FIG. 2: J85-CAN-15 CUTAWAY VIEW

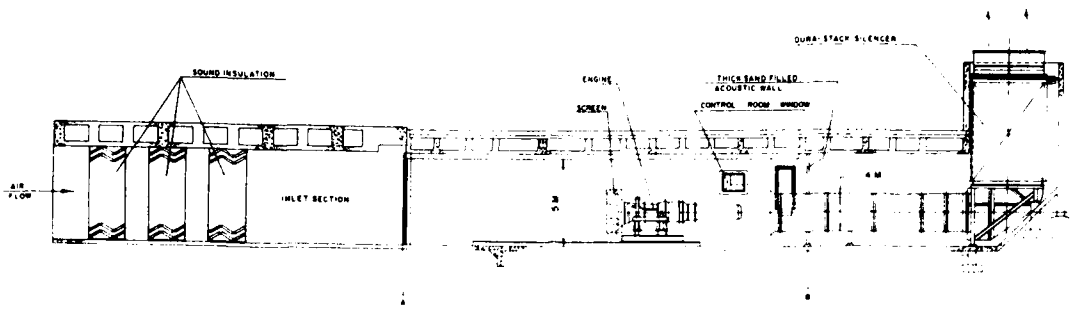

FIG. 3: NO. 5 TEST CELL, ENGINE LABORATORY, NRC

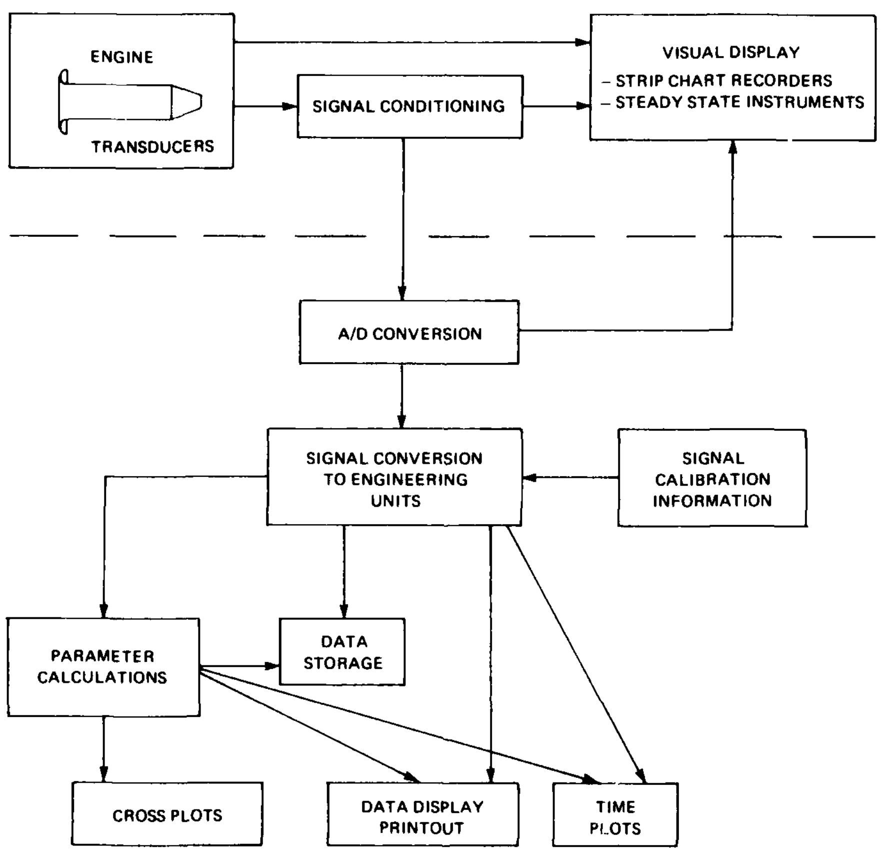

FIG. 4: DATA ACQUISITION – FUNCTIONAL VIEW

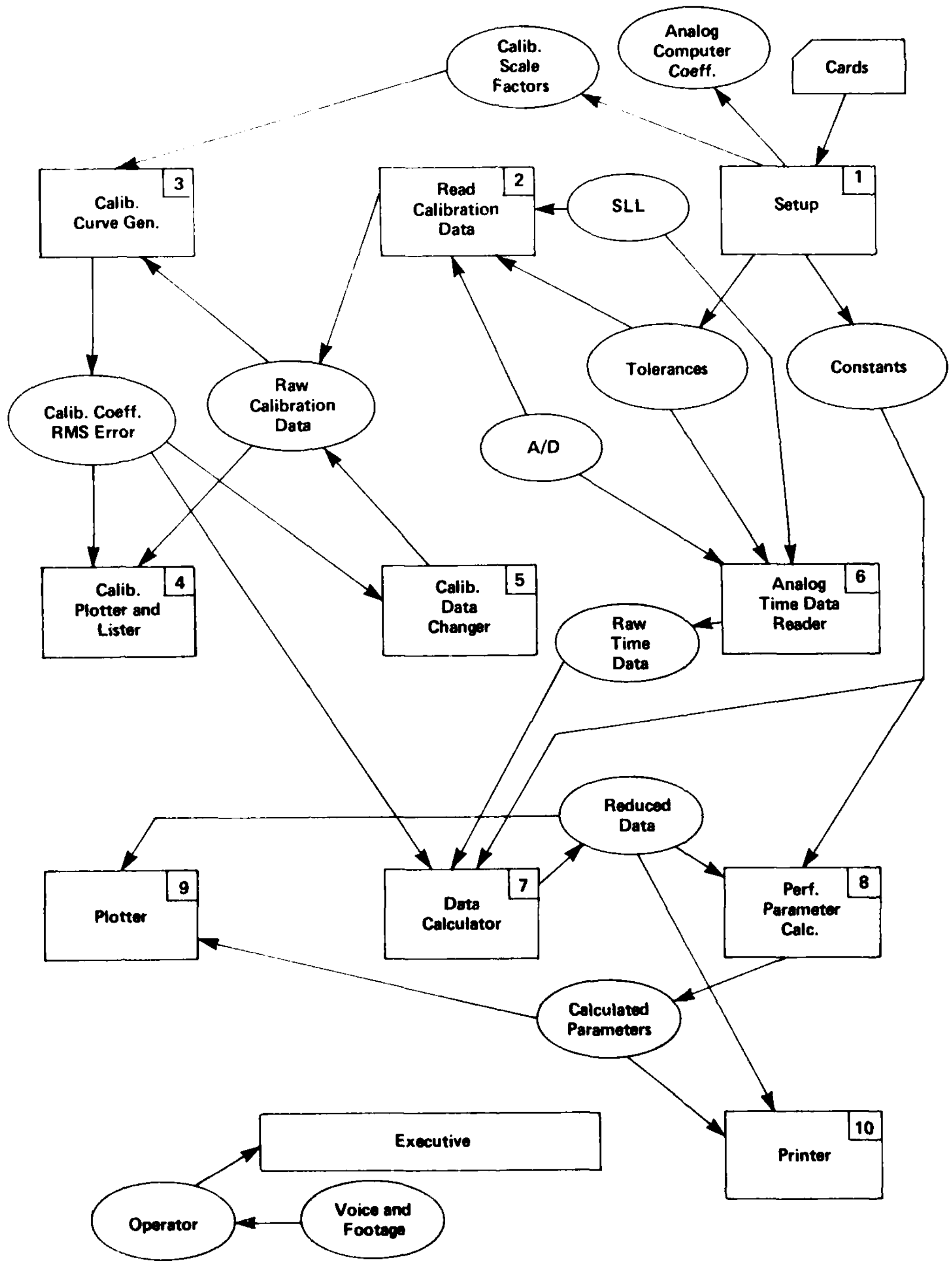

FIG. 5: PROGRAM DATA FLOW

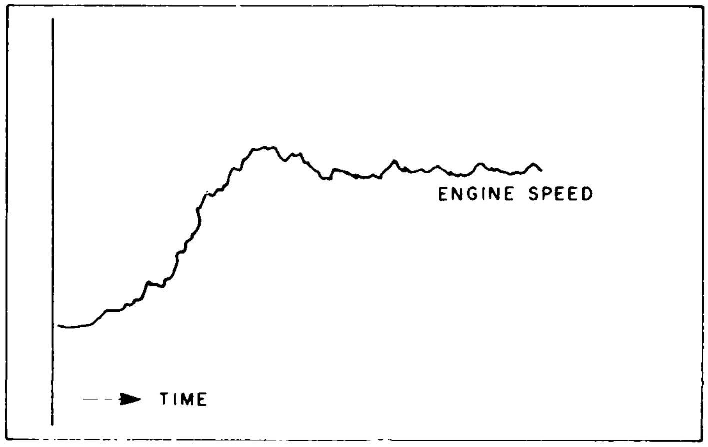

FIG. 6(a): EXAMPLE OF RANDOM NOISE IN DATA

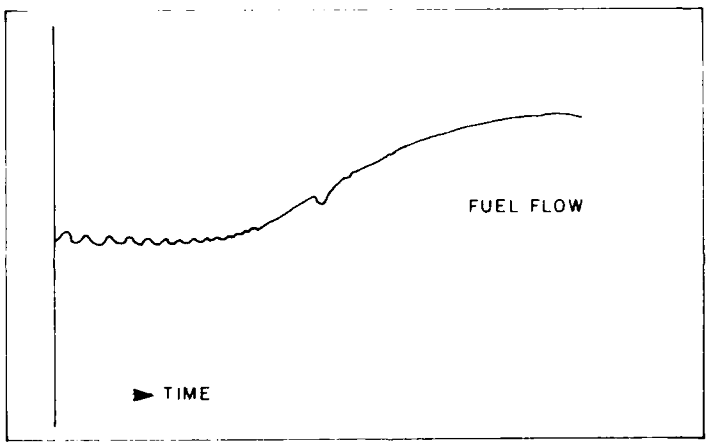

FIG. 6(b): EXAMPLE OF KNOWN INACCURACY IN DATA

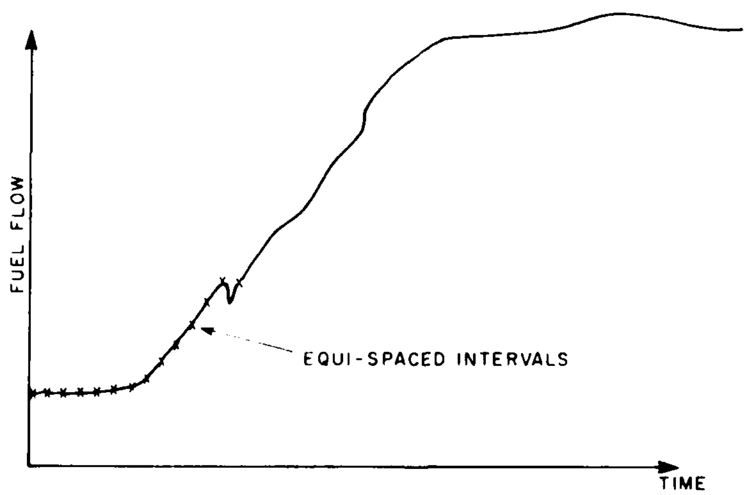

FIG. 7: DATA COMPRESSION BASED ON MAXIMUM ALLOWABLE VARIATION

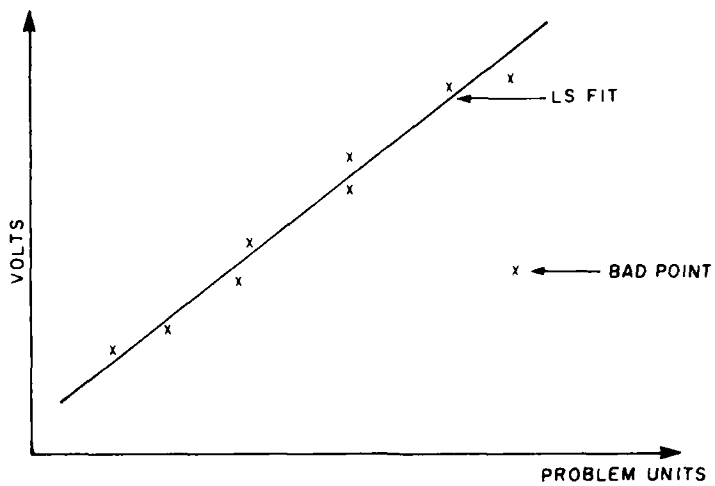

FIG. 8: TYPICAL INSTRUMENT CALIBRATION PLOT

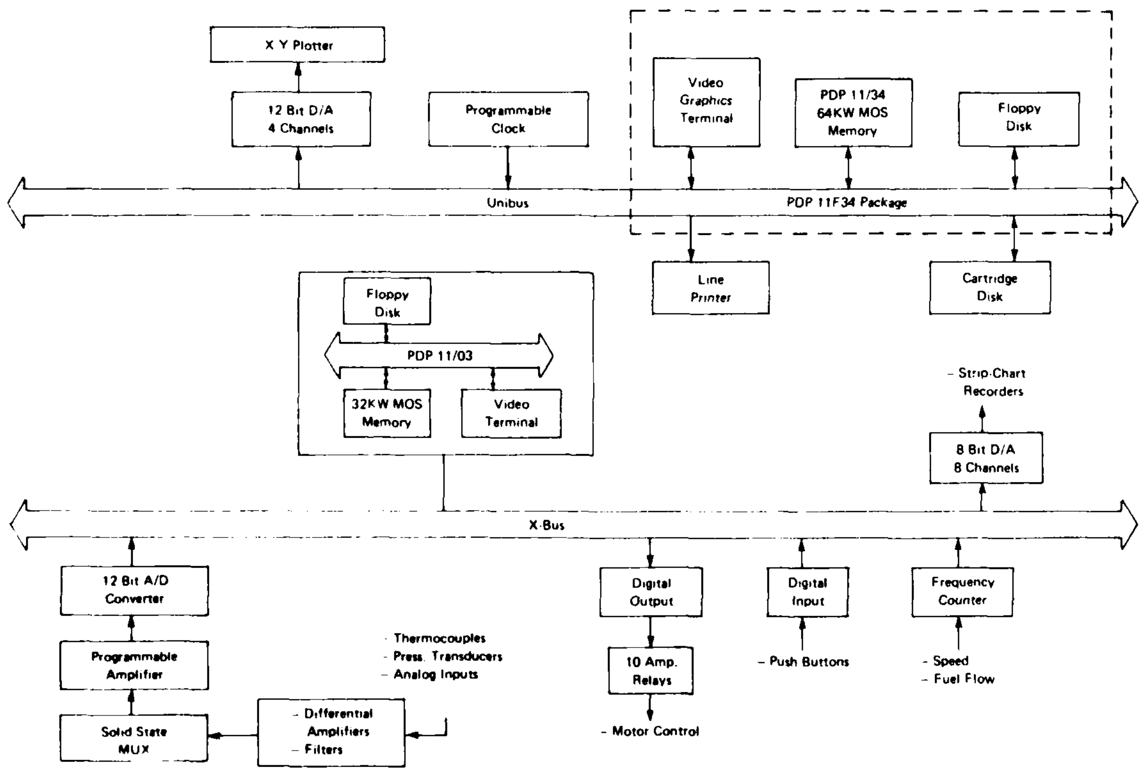

FIG. 9: LAYOUT OF DATA ACQUISITION/PROCESSING SYSTEM

FIG. 10: MINICOMPUTER-BASED DATA ACQUISITION SYSTEM

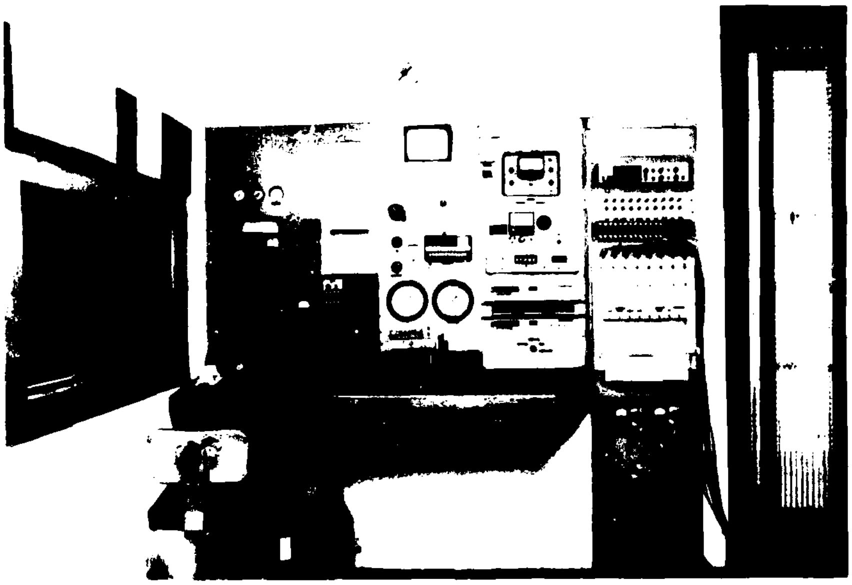

FIG. 11: NO. 5 CELL CONTROL ROOM PANEL

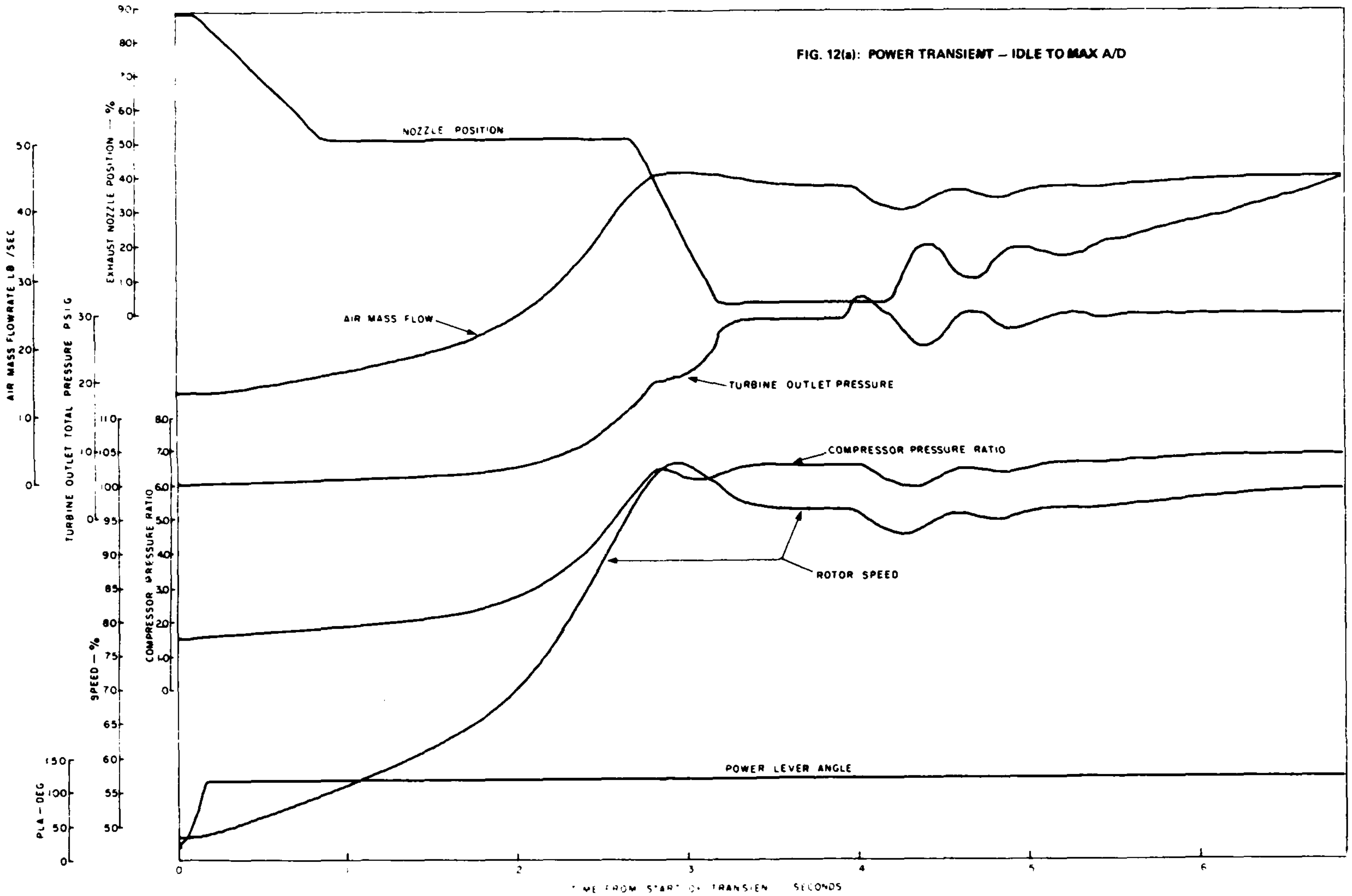

FIG. 12(a): POWER TRANSIENT – IDLE TO MAX A/D

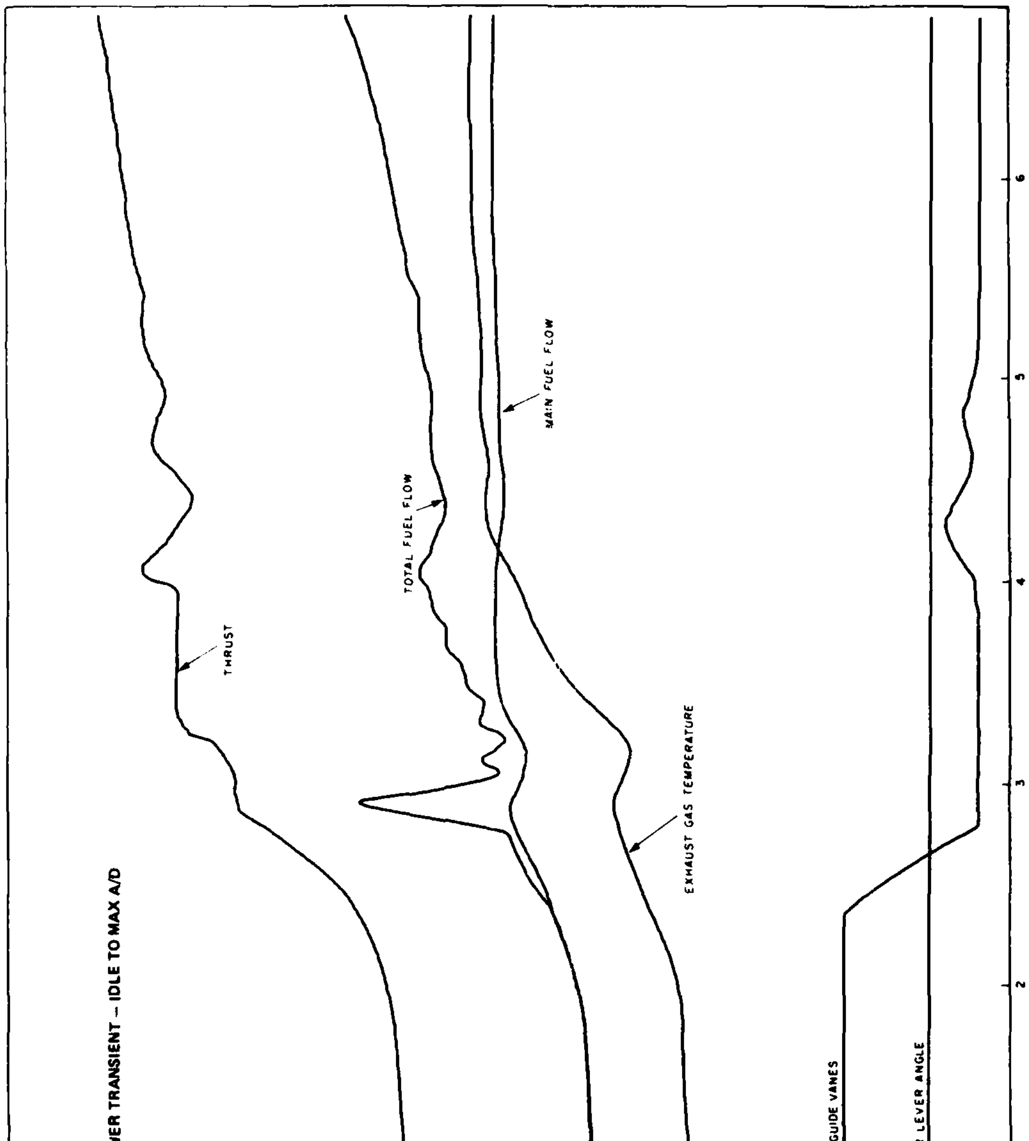

FIG. 12(b): POWER TRANSIENT – IDLE TO MAX A/D

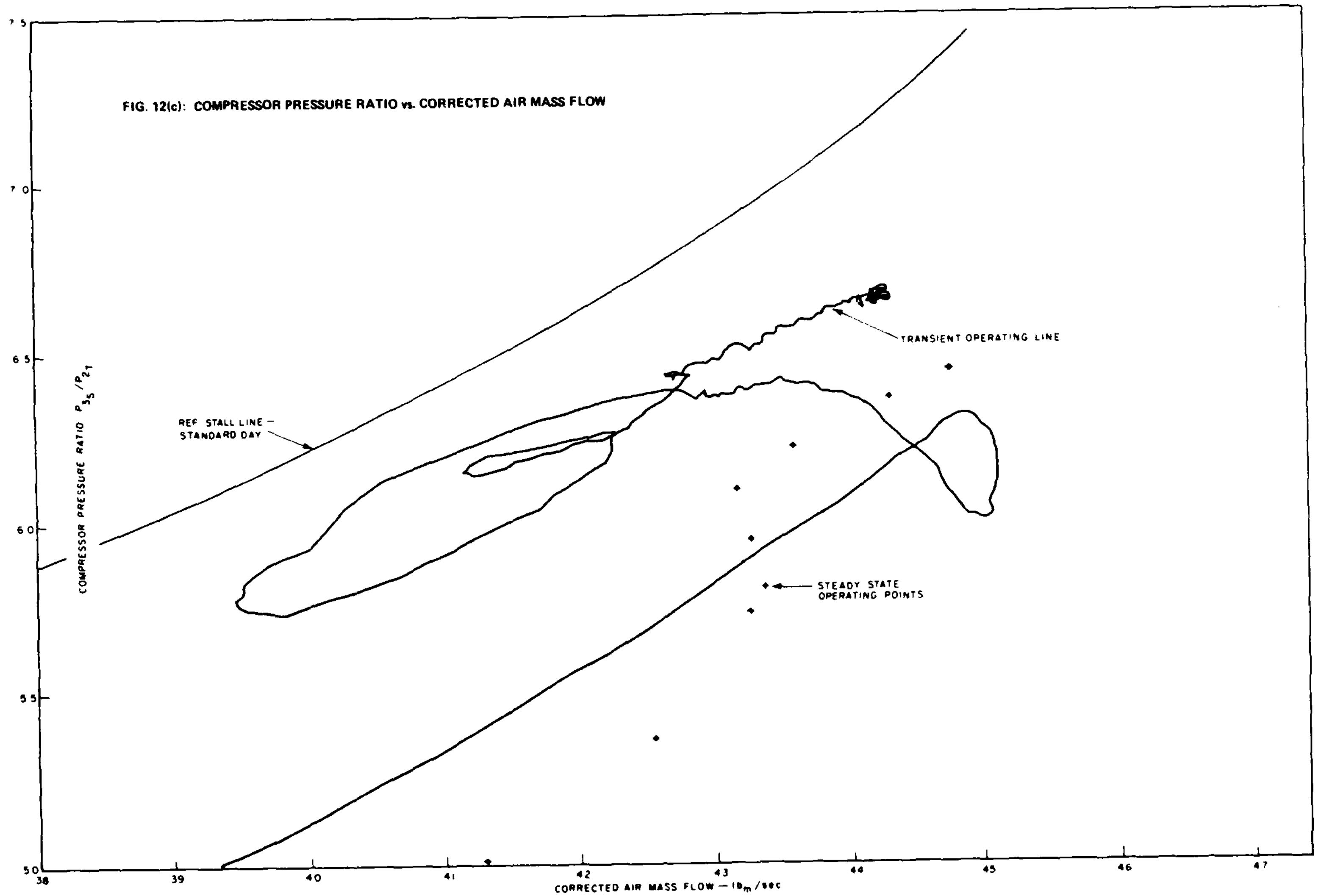

FIG. 12(c): COMPRESSOR PRESSURE RATIO vs. CORRECTED AIR MASS FLOW

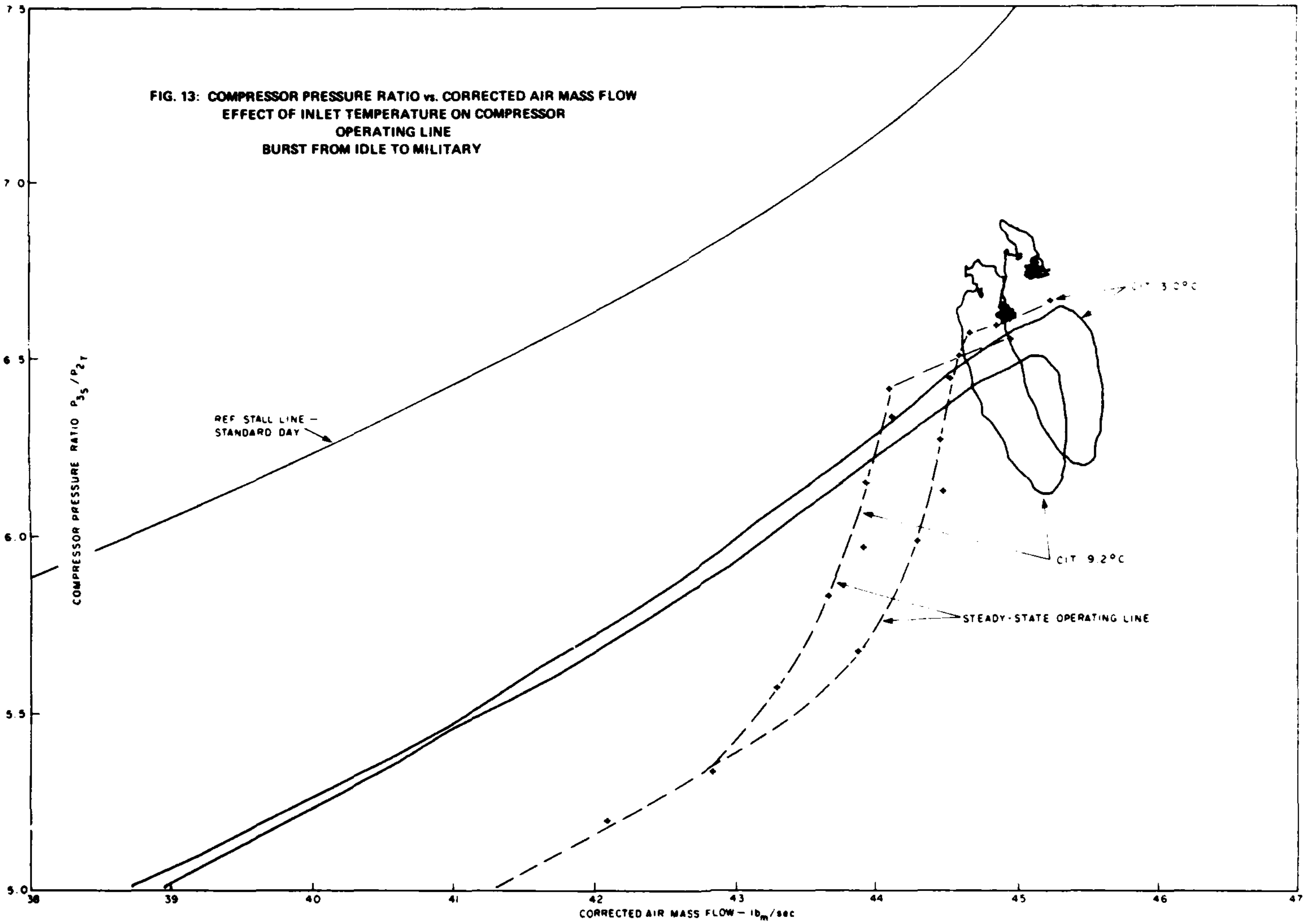

FIG. 13: COMPRESSOR PRESSURE RATIO vs. CORRECTED AIR MASS FLOW
EFFECT OF INLET TEMPERATURE ON COMPRESSOR
OPERATING LINE
BURST FROM IDLE TO MILITARY

EXPERIMENTAL VERIFICATION OF TURBOBLADING AEROMECHANICS

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Summary

Experimental verification in the laboratory is a vital link between the aeromechanical design and integrity validation processes. Based on this premise, this paper offers insights into the practical application of experimental aeromechanical procedures and establishes the process of valid design assessment, avoiding highly theoretical approaches and concepts. The procedures include methods used in design verification, pre-test preparation and instrumentation. Examples are given of typical classes of vibratory behavior and their sensitivities to both engine-system variables and in-service and flight environment effects. The paper further illustrates that early systematic explorations of these variables are necessary to establish these sensitivities and provide adequate margins for long service life.

Introduction

Increasing diversity in the application of modern aircraft and missions has resulted in an expanded spectrum of power plant designs and requirements. As a result, advanced technology has been required to provide energy efficient systems with long-term reliability. These expanded requirements primarily affect the turboblading components (fans, compressors, turbines) of turbojet/fan/shaft, variable cycle engines and their related augmentation systems. Such requirements have not only escalated the complexities of turboblading design configurations, but require serious consideration of extended and variable operational environments of both the immediate and the potential growth flight regimes.

An evaluation of aeromechanical behavior must consider practical operational effects and sensitivities, including aircraft manuever and flight transition distortion, and the integrated effects of a number of other variables, including variable geometry, bleed, power extraction, operating line, and other engine and inlet transient conditions, such as those associated with environmental and weapon delivery gas ingestion. The long range effects of deteriorations, foreign object damage, airfoil erosion and potential control malfunctions also need to be addressed. Predictions of vibratory responses, fundamental mode instability margins, and surge-induced stresses are not yet adequate to eliminate the need for experimental validation of these effects. Overall experience, guided by aeromechanical fundamentals, serves to establish systematic design verification procedures with considerations given to the total engine system.

Design Verification in the Experimental Laboratory

Certain key parameters are influencing the design of modern high performance gas turbines. In fan and compressor blading, the design trends are being dictated by increased tip speeds, relative mach number, lower radius ratio, and fewer stages. The resulting design trends are moving toward thin, highly twisted, low aspect-ratio turboblading with attendent complex, high, steady-state stresses, increased exposure to adjacent stage passing frequency resonance and decreased fatigue resistance to foreign object damage. The result has been unique geometry configurations, such as the integral blade/disk configurations (blisks). While life-cycle cost efficient in small sizes, blisks have limited the frequency and damping control options. Similar trends exist in turbine blading but are further complicated by increased inlet and cooling temperatures that result in thin-walled, complex cooling passage designs. These factors require an efficient iterative design process with aerodynamic, mechanical and aeromechanical disciplines, but more importantly the resulting hardware requires detail design verification by experimental laboratory test procedures.

Verification of design intent is established by test and comparison of the following characteristics with pre-design calculations and criteria:

1. Frequency and Nodal Patterns (Mode Shape)
2. Steady State and Vibratory Stress Distributions
3. Fatigue Strength
4. Structural Strength (Weak-Link Determination)

These data are also used to establish the vibratory limits of various modes from a single airfoil strain gage in subsequent component or engine development tests.

1. Frequency and Nodal Patterns

Of first priority is to determine the natural frequencies and mode shapes of all the modes that have potential resonances over the engines entire operating range. While a variety of standard test techniques are available, including holography, what is experimentally important is establishing statistical data on frequency deviations

resulting from manufacturing tolerances in order to verify resonant margins with respect to the extreme frequency blading. Assesment of the natural frequencies and their statistical deviations can also be indicative of general manufacturing quality. Careful consideration in duplicating boundary conditions is required as it significantly can affect the basic frequencies and statistics. Circumferential variations in boundary conditions are usually found in the stator vanes. In some cases, frequency "tuning" is more effective by modifying the boundary conditions. Such boundary conditions referred to include flexible shanks and other attachments, disk rims, stator vane platforms and supplementary shrouding.

2. Steady State and Vibratory Stress Distribution

In order to establish the allowable vibratory stress of a vibrating turboblade in a given mode, it becomes necessary to determine both the complex steady and vibratory stress distribution and relate the critical stress from the material's Goodman diagram to the stress of the generally located airfoil strain gage (see reference 1). The critical stress in certain modes may occur in the shank, attachment, shroud, or other locations of the structure. Special experimental tests may be required to establish these critical locations and the relative structural strength.

The experimental stress distributions sought have generally been obtained by applying large quantities of miniature strain gages over the airfoil surfaces and at a special location predetermined (by fatigue test) to be a critical stress point.

a. Steady-state stress distributions are obtained by generating stress influence coefficients from unit radial load, bending and twisting moments, and applying the pre-calculated, running loads and moments. Full scale spin-pit tests are sometimes employed for steady stress distributions, but are restricted to local zones due to limitations in read-out capability. Three-dimensional photoelastic techniques have significantly extended the evaluation of steady-state stresses in greater detail of the entire structure. The technique consists of replicating models from actual parts. The assembled photo-elastic cascade is then rotated at equivalent speeds at which point the strains are "frozen" by a cycle of heating and cooling (see reference 2). Such models have recently been used to evaluate stress levels in the attachment and the airfoil with particular attention given to the shroud location.

b. Vibratory stress distributions are obtained from the same matrix of strain gages by exciting the airfoil in each mode of interest to a sufficient amplitude. An obvious limitation of this procedure resides in its inability to yield the correct stress distribution for the mode operating under centrifugal field forces, which can be significant for certain modes. Full scale spin-pit testing has been used, but has the same restrictions noted as before.

Recent advances in efficient finite element programs have been utilized to calculate natural frequencies, nodal patterns, steady-state and vibratory stress in complete airfoil structures including attachments and shrouds. With pre- and post- processors and with suitable adjunct programs, allowable vibratory limits can be established analytically which correlate well to the available experimental data and which can help overcome the experimental limitations of spin-pit testing noted above.

3. Fatigue Strength

Design verification of manufactured turboblading fatigue strength becomes increasingly important with current design trends in geometry and stress. Some factors that can significantly reduce fatigue strength include:

a. Forging Grain Size - Recent tests with large integral-bladed titanium blisks show fatigue strength reduction that is A-ratio dependent:

$A = \infty$	10-15%	where: A = vibratory stress/steady stress
$.5<A<1.0$	30-50%	

b. Foreign Object Damage (F.O.D.) - Typical in-service FOD occurs predominantly at the airfoil leading edge and, in many cases, in regions of high tensile stresses. For these cases, replication of typical notches and tears need to be assessed with the steady state stress present.

c. Corrosion (Intergranular Attack) - The effect of this well-organized problem for certain unprotected alloys has been to reduce fatigue strength as much as 50-60%. As these three illustrative examples show, static fatigue tests at $A = \infty$ are not sufficient to evaluate all aspects of material fatigue strength. Specifically structured tests must be conducted to establish allowable vibratory fatigue limits.

Under conditions of stall/surge, turboblading structures can be subjected to overstress in excess of 2-3 times the material's fatigue strength limiting the engine to a finite number of stalls that can be tolerated. Generalized behavior in this high amplitude fatigue range exists (reference 3), but critical stages may require specific verification of overstress capability. The technique consists of determining crack initiation at various overstress ratios. Specific applications are discussed later.

4. Structural Strength

Just as certain factors can reduce a material's fatigue life, there are also certain geometric and mechanical configurations which can reduce its structural strength. These need to be evaluated. Examples include stress concentrations in the transition regions of the partial chord support spindles of variable stators, cooling holes in turbine buckets, and fillets of dovetail attachments. Low fatigue strength of brazed or welded joints is also a structural strength factor. In the total turboblade structure, its lowest structural strength member is called its "weak-link." Laboratory tests need be aimed at verifying that "weak-link" and evaluating its relative magnitude.

Engine/Component Instrumentation

Once testing in the experimental laboratory has verified that the design intent has been satisfied, the next step in the development process is to evaluate the aero-mechanical design integrity by early and extensive component and/or full engine testing.

The prime objectives of this phase are to define and perform systematic explorations that can evaluate and minimize vibratory stress responses and establish adequate margins with respect to such responses for long life in-service applications. Such tests are performed in conjunction with aerodynamic performance mapping, which when combined require operating the component over a wider range of extremes than in normal service. As a result, a great deal of pre-test preparation and instrumentation is necessary to ensure the vehicle's safety through effective on-site interpretation of the aeromechanical data.

1. Strain Gages

Strain gages, mounted to the vibrating airfoil, have long been the primary aeromechanical sensor, since its electronic signature contains the severity of vibration (amplitude) of a given mode (frequency). Moreover, its characteristic wave form is the result of the airfoil's response to instantaneous flow field and is indicative of the mechanism of vibration. Limitations in application do exist, however.

a. Quantity

In order to minimize the total quantity, each gage must be responsive to more than one mode. Approximately 3-4 strain gage locations per stage are required for proper evaluation of the first 6-8 modes. Assuming 4 gages per location for evaluating blade-to-blade variation and strain gage mortality, a 10 stage compressor would require 120-160 strain gages! The problem is somewhat alleviated by judicious selection based on anticipated mode/stimulus responses and/or multiple builds/tests. Even machines with fewer stages, such as 1-3 stage fans and turbines, still require large quantities of gages in order to give statistically significant assessment of the vibratory limits. Turbines with interblade mechanical dampers are notorious for their large blade-to-blade variation and gage mortality.

b. Performance Effects

Using large quantities of strain gages, particularly conventional high temperature ($>400^{o}F$) gages and small size units, causes a measurable loss in performance. Recent advances in miniaturization and thin film gages are alleviating this problem.

2. Light Probes

The emerging technologies in the light probe systems (see reference 4) have proven to be invaluable as a supplement to the strain gage in measuring non-syncronous vibration of every rotor blade in the stage. This is of particular value in detecting instabilities where there is a high degree of circumferential response variability. The system's current limitation is its inability to measure the syncronous (resonance) vibration of each rotor blade, particularly of the higher order modes.

3. Aerodynamic Instrumentation

As indicated earlier, concurrent aerodynamic and aeromechanical mapping is conducted. The aerodynamic mapping requires inter-stage total temperature and pressure instrumentation which is generally mounted on the static vane cascades. In a particular case, the internal distortion produced by the circumferential spacing and axial superposition of wakes has caused resonant vibratory stress responses in the aft stages that exceed the the endurance limits. Subsequent removal of the vane-mounted aero instrumentation eliminated the vibratory response. Miniaturization in aerodynamic instrumentation is required to eliminate such problems in development testing.

4. Supplemental Instrumentation

Correlation of certain component or engine parameters or structures to turboblading's vibratory characteristics can provide valuable insights to identify and attenuate the forcing mechanisms. Pre-test considerations, for example, might include instrumentation to detect and correlate the shaft and bladed-disk torsional vibration with the fuel and combustor pressure fluctuations resulting from the fuel pump vane passing frequency. A simple strain-gage torsion bridge on the shaft and a fuel manifold pressure pick-up

is indicated. Incidentally, a simple fix for resonances of these systems is a change in the number of vanes in the fuel pump. Bearing cage accelerometers have supplemented diagnosis of blade vibration resulting from interacting tip rubs and non-syncronous rotor whirl. Similarly, strain gaging of fan casings have been used to detect traveling waves in the structure interacting with blade tip rubs.

These are but three examples from a long list of supplemental instrumentation and data that requires more attention in aeromechanical evaluation.

Obviously, simply strain gaging blades is not enough. To fully understand and diagnose vibration problems a wide range of supplementary data must be taken, including at least the following:

a. Rotor Speeds (N_1,N_2)

b. Inlet Pressure/Temperature (P_{T2},T_{T2})

c. Variable Geometry Position (where applicable)

d. Exhaust Nozzle Position (A_8)

e. Starting/Customer Bleed Position (where applicable)

f. Output Torque

g. Discharge Temperature (T_3,T_5)

Data Monitoring and Data Acquisition

1. Slip Rings and Telemetry

Slip rings and telemetry systems are used to transmit strain gage signals to external monitoring and acquisition systems. A brief survey of typical slip ring capabilities is given for comparison with requirements indicated earlier.

Stage Diameter (in.)	Speed (RPM)	Capacity Number of Gages
70	5000	100
24	15000	50
16	20000	50
7	50000	8
4.5	75000	6
3.5	100000	4

Telemetry packages, necessary for monitoring core rotors in two spool designs, have been limited to 20-24 gages for the space and environment available in speed ranges up to 20000 RPM.

Generally, these capabilities have been adequate for the larger machines and single-stage components. However, urgent development is required to increase capacity for the small engine sizes (diameters less than 16").

2. Monitoring Equipment

Since the electronic signature of the strain-gaged vibrating airfoil contains the nature of the aerodynamic flow field, the forcing function mechanism (aerodynamic, mechanical or both) and the response severity, the monitoring aeromechanical engineer's prime tasks include:

- Providing insights into the aerodynamic characteristics of the component and their sensitivities to the test variables which contribute to on-line optimizations, explorations and general test guidance.

- Insuring vehicle safety, particularly in off-design testing, by overriding operational control and returning the unit to a pre-determined safe operating point.

In order to accomplish these tasks, the test facility must provide an aeromechanical monitoring/recording complex, integrated with the operational control station, consisting of the following equipment:

a. Oscilloscopes

A high density array of miniature oscilloscopes (approx. 2" x 3") is required for displaying each strain gage signal as an overall signature of amplitude and wave form. Sweep triggering at 1/rev aids in establishing the existence of non-integral order vibration. An experienced engineer can monitor approximately 16-24 signals. Switching arrangements are necessary to parallel any desired signal to a master oscilloscope and spectral frequency analyzer for more detailed analysis.

b. Spectral Frequency Analyzer

The natural frequencies of the airfoils' modes and their individual amplitudes that constitute the overall signal is determined here. With memory capabilities, trends in amplitude and mode involvement are apparent with slow transients.

c. Fast Fourier Transform Analysis

Currently, soft-ware systems have been utilized for modal analysis as described above which plot the results on Campbell diagrams. More significantly, recent advances utilize the system to synthesize and display wave forms and frequencies of signals generated by the light probe system (see reference 5).

d. Oscillographs

Turbobladinq fatigue cracks and failures can occur during cumulative stall testing in that surge-related stresses can exceed 2-3 times endurance limits for brief encounters. By applying amplitude and cyclic duration of the surge events to material overstress and high-amplitude fatigue characteristics, an estimate of expended life and/or fatigue degradation can be made (reference 3).

The high speed oscillograph record provides the time-history events of stall, and generally requires playback immediately following each event.

e. Recorders

Obviously, with such instrumentation generating invaluable data, magnetic tape recorders are mandatory for post-test detail data reduction and analysis. For correlation purposes, simultaneous recording of the supplemental instrumentation with strain gages is required.

Facilities

Successful design verification of development engine/components requires validation of adequate vibratory stress response margins such that the production units be free from fatigue when operated over a wide range of inlet pressures and temperatures set by the flight map (see appendix). Specific air-frame induced inlet distortions, including the effects of crosswinds, transition of vectored thrust aircraft, and thrust reverser operation (see reference 6), also need to be examined.

Additionally, such stress-response margins can be diminished by certain engine-related variables that range from typical in-service tolerances, transient behavior and deterioration to control and system malfunctions. The corollary to this, however, is that these variables can be used to maximize stress-response margins and thereby provide growth potential for the unit.

1. Typical Engine Variables

The prime variable in the development of compressors and fans is the variable geometry system in which each stator stage is independently variable with the key objective of optimizing stator vane scheduling for both aerodynamic and aeromechanical matching over a wide range of operating conditions. Additionally, systematic explorations are aimed at determining the need for and/or minimizing the number of variable stages. At the same time, this unique aeromechanical exploration of the variable geometry system provides a means for investigative extreme cascade migration, individually or ganged, to establish limiting stress responses or stall margins. Similar objectives are achieved with independent control of such variables as exhaust area, by-pass ratio, bleed and power extraction. In full engine testing, transient speed excursions (bursts/chops) affect the maximum range in operating line. System thermal lags during bursts and chops precipitate tip rubs for aeromechanical evaluation. Introduction of contaminants duplicating long service flow deterioration establishes cleaning requirements based on the limiting vibratory responses caused by this mechanism of cascade migration.

2. Test Plant and Facilities Requirements

The fundamental objectives of the test plant are to provide inlet and discharge environments of pressure and temperatures consistent with the flight-envelope or discharge conditions of the low pressure system when evaluating core compressors. Separate control of these variables is desirable for systematic tests but emergency reset to "safe" regimes should also be provided. In new component rigs and for special explorations in full engines, sub-ambient inlet pressures to approximately 1/4 to 1/2 atmosphere precludes overstressing during initial check-out procedures.

Special considerations for inlet distortion testing could range from simple distortion generating screens to subjecting complete nacelles to actual crosswinds and thrust reverser flow reingestion.

For turbo-shaft engines, systematic aeromechanical mapping of the power turbine is best conducted by holding constant gas generator speeds, smoothly loading and unloading the output shaft affecting a torque-speed "sweep". In this manner, resonances can be evaluated as a function of torque. A dynamometer of appropriate size is required.

The above are illustrative examples of test techniques and the rationale of utilizing the supplementary instrumented variables indicated for both the engine or component and the plant facility. Specific examples of the vibratory stresses and their sensitivities to the variables and indicated solutions are given later in this paper.

Vibration Signature Analysis

It has been emphasized that the characteristic signature of a strain-gaged vibrating airfoil is its response to the instantaneous surrounding flow-field in which it is immersed. As such, it is indicative of not only the severity of the mode(s) involved but also the mechanism of vibration causing it.

Vehicle safety might be insured by assessing only amplitudes of the mode(s) and their potential combinations; however, experience has shown that interpretation of wave-forms, with their time-history response and sensitivities to the variables explored, has contributed importantly to an integrated aerodynamic, aeromechanical and mechanical development process by providing "on-line" diagnosis and remedial actions as indicated.

Turboblading vibration can be classified into three main categories with typical wave forms given in Table 1 (a-m).

CATEGORY	TABLE
1. Resonance (Forced) Vibration	
• Aerodynamic	1a,b
• Mechanical	1c
• Combined Modes	1d,e,f
2. Flow Induced (Aerodynamic) Vibration	
• Separated Flow	1g,h
• Rotating Stall	1i
• Surge	1j
3. Self Excited (Flutter) Vibration	1k,1,m

Key characteristics supplementing the wave forms of Table 1 are given here for purposes of illustrating typical diagnostic interpretations.

1. Resonances

All resonances are typified by frequencies that, when in proximity to peak resonance response, will be exact multiples of rotor speed and "track" with slight rotor variations. There are three basic kinds of resonance which must be considered: aerodynamic, mechanical, and combined.

a. Aerodynamic

While Table 1a is an example of responses due to wakes from an 8-strut frame, Table 1b is unique in that the response is the 7th harmonic of a 1/rev stimulus caused by a reversed stator vane segment. (The design was a 12-segment banded stator vane configuration in which one of the 12 segments was misassembled.)

b. Mechanical Resonance

Mechanical resonance due to tip-rubs is indicated by Table 1c. Inspection of the wave form shows two rub spots exist approximately 154° apart.

Total system damping is obtainable from the decay in the wave forms. Introduction of such an impulse stimulus has been used to determine the total damping of cantilevered compressor blades. Typical values of log decrements in the first flexural mode range from 5% to 8%.

Other examples of stimulus sources that cause potential resonances include adjacent blade or vane passing frequencies, fuel nozzles in combustors, starting gear tooth passing frequencies, partial entry turbine air-impingement starters to name but a few.

c. Combined Resonances

Combined mode wave forms are self-explanatory as indicated in Tables 1d-1f. It might be of design interest not to design blading whose second to first mode frequency ratio is 4:1 when potential stimuli exist at 4:1 (such as 4 front frame struts and 16 inlet guide vanes.)

2. Flow Induced (Aerodynamic) Vibration

This type of vibration, referred to as "separated-flow vibration," is characterized by non-periodic amplitude modulation in the fundamental modes at their natural frequencies, independent of rotor speed. The mechanism is separation and re-attachment of flow. Increasing severity increases both amplitude and amplitude modulation until the cascade develops a rotating stall and propagates surge. This typical progression is shown in Tables 1g-1j. The slip speed of the rotating stall is estimated at approximately 45% of rotor speed.

Turboblading response to surge is of special importance since the mechanism (see reference 7) results in brief but damagingly high overstress ratios. Table 1j gives specific characteristics of surge responses, such as pulse frequency, mode, overstress ratio and cascade damping which governs stress duration.

3. Self-Excited Vibration

By far the most destructive type of vibration encountered in turboblading, self-excited (flutter) vibration is characterized by subtle indications of onset and explosive amplitude build-up to levels that can approach 2-3 times endurance limits. Frequencies correspond to the natural frequency at onset and are independent of rotor speed. In some cases, we note frequency suppression in the order of 5-8% as entrainment progresses. Such a controlled sequence is shown in Tables 1k-1m.

An instability boundary by definition is the locus of zero total damping of the cascade in a given mode. Consequently, one should logically expect that mode's resonant peak amplitude to increase dramatically when in close proximity to that boundary. This characteristic has been used to assess proximity to instability by making small speed excursions and noting the systematic increase in peak resonant response.

Illustrative Experiences in Aeromechanical Experimental Verification

Prediction of the vibratory responses in forced vibration, instability margins and surge related stresses are presently inadequate due to the extreme complexities in adequately defining aerodynamic flow fields, forcing functions, total damping and non-steady aerodynamic coefficients over a wide range of operating conditions, particularly at significantly off-design conditions where experience indicates most of our problems lie. Furthermore, the prognosis for short range developments that may preclude the need for experimental design verification is poor. Such developments will best be utilized in guiding the advanced technology engine designs. In the foreseeable future, systematic design verifications will need to be continued and improved, utilizing existing data-bank experiences.

In an attempt to guide that experimental data bank, we give illustrative experiences gained at General Electric during the past 25 years related to three general topics:

Resonant Vibration
Self-Excited Vibration
Surge Overstress and Fatigue Capability

An attempt is made to consider practical gas-turbine designs and operational effects that have caused problems and to establish effective fixes but, more importantly, to establish the variables required in aeromechanical experimental verification.

1. Resonant Vibration

It is not possible to avoid resonance for multi-moded airfoils in multi-stimuli, variable speed machines. Current methodology for minimizing resonant response is through the following approaches:

Frequency Control/Stimulus Selection
Stimulus Control and/or Attenuation
Internal Distortion (Harmonic Content)
Damping and Structural Strength

Typical examples of implementing these approaches are given in Table 2, which provides the relative risks and trade-offs in accepting certain resonant potentials. One needs to rank the modes of vibration assuming the consequences of fatigue in order to assess relative risks. As shown in Figure 1, a Damage Severity Criteria is established for compressor/fan blading with implications and impact given.

a. Illustrative Examples

- Frequency Control/Stimulus Selection

Here we consider a design or perhaps a redesign approach after tests have indicated such necessary action is required. Figure 2 depicts two methods of avoiding resonance. The first example assumes the airfoil frequencies are known and resonance is avoided by selecting (or changing) the excitation stimulus to n1/rev or n3/rev. The second example considers designing the appropriate adjacent blade rows with the required aspect ratio to avoid the rotor or stator's first two-stripe (1-2S) panel mode resonance with its adjacent row's passing frequency. Based on current solidity trends, we find this criterion to be met when the ratio of the blade to stator aspect ratio is approximately 0.6.

- Stimulus Control and/or Attenuation

During development it is sometimes expedient and cost-effective to accept a potential resonance provided its stimulus can be attenuated so as to reduce vibratory responses to acceptable levels. Such is the case in the following example shown in Figure 3. Figure 3a illustrates the configuration of a damperless high pressure turbine design

with its original 5 equally spaced T_5 harness probes. Shown in the Campbell diagram of Figure 3b is the 5/rev first flexural frequency margin provided in the original design. Production versions required improved T_5 harness correlation by converting to a 7-probe configuration which has a potential 7/rev resonant response at approximately 93% of design speed.

A strain gage instrumented test in which the variables of operating line and probe axial location was conducted based on prior experience with discharge struts. As shown in Figure 3c, the resonant response was reduced to acceptable levels by increased axial positioning of the probes, the mechanism at work being pressure field attenuation with axial distance as indicated in Figure 3d.

- Internal Distortion (Harmonic Content)

Table 1b shows that a resonant response can be excited by a single discrete pulse. More generally a resonant stress response is dependent on the harmonic content of the circumferential distortion patterns. In the example presented in Figure 4, the internal circumferential distortion pattern was caused by aerodynamic vane-mounted probes producing a superpositioning of upstream distortion wakes. This figure sequentially illustrates the reasoning and rationale of the test events. During the first excursion to maximum speed, an excessive 12/rev first-flexural response occurred at 14700 RPM. Upon detail playback, nearly all integer n/rev responses were noted in groupings of 6, with 12/rev being the highest response, as illustrated in Figure 4a.

Because the differential in number of upstream stator vanes was 12 and they were heavily instrumented with the aerodynamic probes, a Fourier analysis of the combined wakes was made and compared with experimental data as shown in Figure 4b. The obvious solution was the removal of the probes which eliminated the responses as shown in the 12/rev response of Figures 4c and 4d.

Earlier applications of this approach can be found in reference 6 with regard to responses due to multiple disengagement of variable stator vanes.

- Damping and Structural Strength

When frequency control and stimulus attenuation cannot be used effectively, damping can be a powerful design tool. Three sources of damping exist:

i. Material Damping: The hysteretic damping of materials used in modern turbo-blading ranges from 0.5 to 1.5% log decrement (600>Q>200).

ii. Aerodynamic Damping: The amount of aerodynamic damping is generally dependent on the mode and state of condition of the working fluid. Some data are currently being evaluated for effects of cascade migration. Typical experience with total damping for various modes is as follows:

MODE	% LOG DECREMENT
1F	4-8
1T	2-3
2F	.5-1.5
2T	1-3
1-2S	.5-1.0

Note: Since most of these data were derived from integral blade-disk configurations, mechanical damping is assumed zero. Damping data of axial dovetail designs appear to fall in the same range.

iii. Mechanical Damping: Perhaps the most effective means of controlling resonant response is mechanical damping. The following example describes a growth step redesign for production that was limited by design constraints from utilizing beneficial frequency and/or stimulus control.

The general design configuration is shown in Figure 5 along with its Campbell diagram. Note that the extreme frequencies, based on platform/dovetail boundary conditions, are shown. The interblade phasing during resonance establishes the intermediate frequencies shown (and provides the mechanical damping) with 4/rev resonance existing near design speed. In other cases effective tuning can be achieved by appropriate shank-damper designs. Figure 6 schematically represents the bench test optimization procedure and resulting 4/rev engine response. Figure 6a depicts the configurations tested with the prime variables being damper relief angle and loading. Note that structural strength improvement was also introduced by thickening the shank in configuration 2. Some earlier testing, along with external literature (see reference 8), suggests that damper load is the prime variable governing damper effectiveness. We find, however, that damper angle is significantly more important, as shown in Figure 6b. Engine test results of configuration 2 are shown in Figure 6c. The production version of configuration 2D reduced that engine's stress response to levels within the range of our composite turbine bucket stress response experience shown in Figure 6c.

Similar effective damping control of resonant response applied in compressors is shown in Figure 7. After initial development, it became possible to "lock" the

initially-designed variable vane simply by installing a lock-plate and nut which "hard-mounts" the vane into the casing.

Tests of this configuration revealed the existence of a significant response in a high-order torsion mode which exceeded endurance limits at approximately 97% fan speed. Adding a simple Teflon bushing, which "soft-mounts" the spindle in the casing and thus provides torsional damping reduced the responses to negligible levels.

b. Resonant Response Sensitivities to Engine Variables

Because the integrated gas turbine variables are subject to transient deviation in actual service, experience has shown that specific tests need to be conducted to evaluate resonant response with respect to:

Variable Geometry Systems
Flight Map Environment
Compressor Interstage Bleed
Output Torque

The following examples not only show the significant effect of these variables on resonant response, but also suggest the test facilities required to analyze them.

- Variable Geometry System

Corrected speed ($N/\sqrt{\theta}$) governs the tracking of a variable geometry stator system utilizing a temperature sensing element in the compressor inlet as shown in Figure 8. Resonance, however, occurs at a given physical speed for which the variable geometry can vary depending on local ambient temperature. Since temperature environments can vary significantly, this sensitivity can be thought of as an ambient temperature sensitivity. Figure 9 shows the systematic exploration of this mechanism. A stage 1 stator Campbell diagram shows potential passing frequency resonance with its upstream rotor 1 and downstream rotor 2 passing frequency stimuli at 94 and 85% physical speed. On a 59°F standard day, the V.G. position at those conditions are 24% and 73% closure, respectively, as shown in Figure 9b. Stress responses passing through these speeds are 32% and 28% of endurance limits. By opening and closing the V.G. tracking through mechanical biasing, a maneuver which simulates lower and higher inlet temperatures, the response is mapped as shown in Figure 9c. Two important conclusions are noted here:

i. Lower inlet temperatures (and opened schedules) tend to increase stress response. Fixing this stator in the open position would be unacceptable.

ii. Contrary to intuitive reasoning, the downstream rotor generates the more significant resonant response amplitudes.

- Flight Map Environment - Inlet Pressure & Temperature

Extending the previous example to include the effects of inlet pressure in the flight-maps' pressure and temperature regime, we now recognize the need to test for pressure or density effects at constant corrected speed (constant T_{T2}) in order to maintain a constant variable geometry position over the pressure range tested. Figure 10 illustrates these points, noting that in vacuo ($\rho/\rho_o = 0$), vibratory stress is zero. We find that:

$\sigma_v \propto (\rho/\rho_o)^{\eta}$ ...η being mode and density dependent

where: $.5 < \eta < 1.0$

- Compressor Interstage Bleed Effects

Interstage bleed, generally provided for customer use, anti-icing air, cooling and/or sump pressurization, can be an effective exploratory variable in that it can alter the matching of upstream and downstream stages, affect the operating line, and be a powerful influence on turboblading stress-response, as shown in Figure 11.

During development of the original design (Figure 11a), second torsional mode responses with respect to the adjacent rotor passing frequency were excessive and could only be reduced by an overall operating line reduction. Interestingly, the stage 4 stator row bleed caused unloading of stage 3 and loading of stage 5. Minimizing responses to 100% endurance limits with 10% bleed was the best that could be achieved, resulting in a redesigned third and fifth stator by tuning and avoiding the passing frequency. As indicated in Figure 11b, attempts to shut off stage 4 interstage bleed to sump pressurization requirements (1-2%) caused the stage 4 stator response to increase rapidly, leading to a final redesign to avoid the 32/rev passing frequency.

- Output Torque Dependency in Power Turbines of Turboshaft Engines

The gas force loading and the vibratory forces resulting from a given pressure or velocity defect are proportional to the torque developed at a particular turbine speed. Based on this premise, aeromechanical mapping techniques were developed to evaluate resonant responses occurring in the operating range as shown in Figure 12.

The Campbell diagram in Figure 12a indicates the power turbine resonant speed-range of interest. Power-turbine speed-excursions (speed "sweeps") are accomplished by setting a constant gas generator speed and loading and unloading the dynamometer, allowing the power turbine speed to "sweep" through the blades' resonant speed. Resonant stress is then correlated to the resonant speed torque as shown in Figure 12b.

In Figure 5, tangential platform clearance is shown which is set to preclude "arch-binding" or loss of this clearance due to differential thermal gradients. The platforms essentially "bind-up", causing reduced damper effectiveness, which then increases the stress-response. The response data were taken from a development design with inadequate clearance to prevent arch-binding. Responses approached 140% endurance limits at high torque. Clearances were increased on the instrumented unit and retested. Stress responses at the maximum torque value did not exceed 50% endurance limits - a reduction to almost 1/3!

This type of turbine bucket mapping is also useful in establishing turbine bucket instabilities, which will be discussed in the next section of this paper.

2. Self-Excited Vibration

Many excellent papers in the open literature give comprehensive treatments of the mechanism and theoretical approaches used to describe self-excited vibrations. Limitations are also adequately described. While it is beyond the scope and purpose of this paper to further elaborate, our prime objective is to suggest practical considerations regarding the aeromechanical verification procedures that achieve the objective of assuring adequate instability margins. The consequences of not providing such margins are very serious since the explosive severity of vibration is such that a complete blade row can fail in seconds as shown in Figure 13. Approximately 2% speed penetration into the instability regime (equivalent to 1° incidence for this case) results in stresses approaching 2.4 times the endurance limit.

3. Instability Regimes and Representation

Early work by Carter and Kilpatrick (reference 9) idealized instability regimes in terms of the reduced velocity-incidence (angle-of-attack) map redrawn in Figure 14, based on experimental tests of an early compressor rig. Regimes identified then were related to subsonic stall and choke. Increasing values of the reduced velocity parameter are generally destabilizing. However, other variables affecting instability are incidence (or proximity to stall or choke), fluid density, solidity, Mach number, and blade twist/bend deflection ratio, all of which further complicate such representations. While this method is perhaps the most useful to the aeromechanics engineer, other methods have been used for convenience in presenting the data or in generating experimental instability data, and include the following:

Compressor Map
Variable Geometry Map
Speed-Temperature-Pressure Map
Flight Map

- Compressor Map

Figure 15 identifies 5 distinct regimes on the standard compressor map:

1. Subsonic Stall
2. Supersonic Stall
3. Supersonic Shock
4. Choke (Note two regimes)
5. Blade-Disk-Shroud System Modes

It is important to recognize that the assumptions implicit in this method are nominal operation of variable geometry and bleed with constant inlet temperature. As indicated by two regimes of choke instabilities in Figure 15, this method is subject to misinterpretations, as will be discussed later. If one chooses to map the compressor as a function of the variable geometry position assisted by operating line adjustment with discharge area, the compressor map of Figure 16 is generated.

- Variable Geometry Map Representation

An invaluable development tool in optimizing compressors for both aerodynamic and aeromechanical objectives is the variable geometry system. This consists of a number of variable vane rows whose stagger angle can be varied and scheduled as a function of corrected speed by means of mechanical activators positioned by the main fuel control. Such a production system was shown in Figure 8. During development, optimization is accomplished by varying each row remotely so as to effect an extreme cascade migration. Instability margins are then evaluated by systematic off-schedule operation of the ganged schedule as shown in Figure 17. Additionally, resonant response sensitivities, as discussed earlier, are also evaluated.

- Speed-Temperature-Pressure Map

Self-excited vibration is governed by the blade incidence (a corrected speed variable) and the absolute value of the relative inlet velocity (an uncorrected variable). Since these are not directly measurable, we seek to define the corresponding engine variables.

Variable		Relation	Engine Parameters
Incidence	-	$f(N_p/\sqrt{\theta})$	N_p, T_{T2}
Velocity	-	$\dfrac{V_r}{\sqrt{\theta}_1} \cdot \sqrt{\theta}_2$ or $\dfrac{V_r}{\sqrt{\theta}_1} \cdot \dfrac{N_p}{N_p/\sqrt{\theta}_2}$	N_p, T_{T2}

Thus, the appropriate plane of instability mapping is in the physical speed (N_p) temperature (T_{T2}) plane. Composite mapping of various rotor designs is shown in Figure 18, with evaluation conducted at various levels of constant inlet pressure (P_{T2}).

An extensive data bank of this type of testing and experience is now being utilized for further investigations by various government agencies (see reference 10).

- Flight Map

From mapping given instability in the N_p-T_{T2} plane for various constant pressures (P_{T2}), one can now superimpose the instabilities onto the flight-map by transposing the boundary point-for-point at constant physical speed. The method is illustrated in Figure 19. In this particular example, the growth version could become a problem for two reasons:

i. The extended flight map regime increases both pressure and temperature.

ii. In order to maintain thrust at elevated temperatures, an increase in engine physical speed may be required.

b. Compressor Map Representation Anomaly

During variable geometry mapping of a mid-stage compressor blade, an instability regime was mapped to the extent of finding its closed "island", as shown in Figure 20a. Aerodynamic data evaluation indicated the regime to be in close proximity to the stage choke limit, with data generated to construct its reduced velocity-incidence map and normal operating line migration. By performing speed excursions on lower operating lines (migrating the incidence more to the negative), the compressor map representation of this "choke" instability appeared to be consistent with other experimenters; that is, it is located below the nominal operating line as shown by zone 4 for the nominal blade in Figure 20b.

Similar tests of a blade closed at its tip section generated nearly the same stability boundary (in terms of $\bar{V}$-i) as its predecessor but the design could not safely exceed 93% N_G. Mapping this bounded instability "island" with the "closed" blade can be seen to produce the compressor map representation of this regime as indicated in Figure 20b, labeled "closed blade". Although it has all the characteristics of a transonic "stall" instability, the cascade remains choked. A stator stage with a similar choke-instability regime has been recently tested with resulting compressor map representation as described above.

If the supersonic shock boundary (or any instability boundary) is a closed regime, similar difficulties in interpretation may exist.

c. Illustrative Examples of Cascade Migration Mechanisms

Before considering specific mechanisms by which cascades can be migrated for exploratory and empirical testing, designers need to consider some practical, "real world" examples of engine operation and environmental exposure that can cause loss in instability margins.

Type of Operation	Engine Parameters Affected
• Transient Power Bursts/Chops	Operating Line
• Afterburner Transients	Operating Line and Speed
• Flight Map Operation	Inlet Pressure and Temperature
• Fouling Environment (Sand, Salt, Oil, Smog)	Flow Degradation
• Inlet Distortion From Aircraft Inlets/Ducts	Flow Degradation
• System Failures	A_8, A_{28} - Operating Line V.G. - Flow/Speed
• Combinations of Above	All

The consequences of the problems listed above and some methods to assess their individual contributions to instability margin loss are discussed in the following paragraphs. Simple cascade migration methods used in aeromechanical mapping are also presented.

- Incidence Migration Methods

There are three fundamental methods of migrating the cascade's incidence and relative velocity. These are the effects on the vector diagram due to:

i. Operating Line
ii. Flow Degradation
iii. Variable Geometry

Figure 21a illustrates incidence-velocity migration due to operating line effects and flow degradation (including that caused by pressure distortion) at constant inlet swirl angles or V.G. setting at a given tip speed. Flow variations $(\frac{W\sqrt{\theta}}{\delta})$ are the primary cause as indicated in the velocity diagram.

Figure 21b illustrates a similar migration, assuming flow is held constant, caused by varying the cascades' upstream stagger.

From these cases, high operating line, flow degradation, distortion, and opening of upstream stators are seen to increase incidence and generally be destabilizing.

- Illustrative Practical Examples

Typical practical applications and examples of these fundamental mechanisms are given in Figures 22-26 and are self-axplanatory:

i. Figure 22: Transient high operating line due to power burst, penetrating subsonic stall instability boundary.

ii. Figure 23: Transient low operating line due to afterburner shutoff, penetrating choke instability boundary.

iii. Figure 24: Flow degradation; migration caused by systematic flow reduction induced by "fouling" with soap/carbon solution.

iv. Figure 25: Variable geometry mapping techniques utilizing "ganged" or individual stator vane settings to establish margin (measured in $\Delta N/\sqrt{\theta}$).

v. Figure 26: Combined effects of flow degradation, V.G. setting and tip radial distortion on subsonic-stall instability.

> Note: This last example is a classic case history emphasizing the need to design cascades to stall prior to encountering instability. In one application, deterioration-causing power loss prompted the operator to open the V.G. to increase flow and power. Needless to say, after successive adjustments, power loss was sudden - with broken blades. The obvious solution was and has been to incorporate periodic wash procedures.

- High Inlet Temperature (T_{T2}) Migration

In various military applications, certain engines are subjected to bursts of hot gas ingestion resulting from gun or rocket firings. This type of environment is conducive to stage migration which can result in penetration of instability boundaries. Self-excited vibration, as indicated by the boundary on a stability map, is governed by the blade incidence (a corrected variable) and the absolute value of the relative inlet velocity (an uncorrected variable). Once a stability boundary has been determined as a function of reduced velocity and incidence, it becomes a simple matter to parametrically determine the blade row migration as a function of inlet temperature, physical speed, and V.G. setting as shown in Figure 27. Note that the temperature transient could be such that the response of temperature sensing devices might cause the V.G. schedule to lag considerably, since the schedule is controlled as a function of corrected speed. Figure 28 is a typical trace of elevated temperature migration into a first torsion subsonic instability regime and shows the effectiveness of V.G. reset while at elevated temperatures.

Figure 29 is the result of a parametric study of two stages of a multi-stage compressor showing the instability boundaries of the stages as functions of inlet temperature, physical speed and V.G. setting. These results concluded the following:

i. Stage A, which had a very respectable stability margin at the design point could become unstable if a temperature transient reached 260°F without V.G. schedule response (typical rocket-gas ingestion temperatures).

ii. Stage B could become unstable in the idle range with temperature values of 140°F, such as could occur from ingesting exhaust gas from a leader aircraft during formation taxi conditions.

iii. Relative to the potential problem of stage A, V.G. preset prior to gun or rocket firing appeared to be the most effective method of preventing high temperature migration and has been used to preclude both instability and/or stall during weapons delivery.

- Turbine Instabilities

While the self-excited vibration of turbine buckets has not been a problem in production engines, some machines have been operated excessively off-design to explore the potential for this type of vibration and to establish margins for future growth applications. As indicated earlier in the resonant vibration section of this paper, power-turbine mapping techniques uniquely provide instability margins of an unshrouded power turbine as shown in Figure 30.

d. Self-Excited Vibration Response Sensitivities

In systematic evaluation and design verification programs, certain variables affecting the threshhold sensitivities of instability boundaries have been determined. Some have had significant impact on improving existing minimal margins, while others unique to a given engine size and/or manufacturing technique are second-order effects. Some of these experiences are offered here to illustrate the need for future designs and/or test procedures.

- Effects of Inlet Pressure and Variable Geometry

Inlet pressure sensitivity on the choke instability of a mid-stage compressor blade was of concern due to the requirements of a cold-day, high Mach number application. Figure 31 depicts the results of mapping the compressor with nominal tracking variable geometry and discharge area. Clearly, increasing pressure is destabilizing. However, note that the production design incorporating a 4.5° opening of its upstream stator schedule increased speed capability from 97 to 103% (ΔN_c = 6%) and provided an additional 10 psi inlet pressure capability. It should be evident that incidence (or choke proximity) control is significant.

Note also that a pressure threshhold for this stage is defined at approximately 14.7 psia which has significant implication for test procedures. A case in point was the aerodynamic mapping of an advanced technology unit in an overspeed exploration. Testing was performed at 1/2 atmosphere to minimize dangers in "inadvertent" stalls. Supersonic shock instability was not discovered until tests at ambient inlet pressure was conducted two weeks later. The threshhold pressure was at 13 psia. The message is clear:

i. For development rigs, test at least to the full atmosphere at overspeed conditions to the maximum and minimum operating lines.

ii. Conduct ram tests prior to production commitment.

- Effects of Frequency Tuning

Destabilizing effects of the cascades' frequency distribution was conducted shortly after publication of reference 11 in both the subsonic stall and choke instability regimes. This effect is not of academic interest since frequency statistics are a function of manufacturing process and machine size for a given tolerance.

Results are shown in Figure 32. In the stall-flutter regime (Figure 32a) an initial stall cascade with a standard deviation (1σ) in its first torsional frequency of 2.0% was not limited by instability. Selectively reducing the deviation to 0.64% resulted in an instability near the operating line. Similar results were obtained in the second flexural mode choke-instability regime shown in Figure 32b.

It is emphasized that both stages, while sensitive to "frequency tuning", were corrected by altering incidence (stall and choke proximity)

- Effects of Solidity

Upon encountering "choke" related and/or shock stall instabilities, preliminary rationale indicated that improved margin could be achieved by altering or reducing the inter-blade shock strength. Early tests of variable stagger around the row ($\pm 3°$) failed to show significant improvement. A reduced solidity test did, however, improve the design margin. (Circumferential dovetail designs allow solidity changes to be

made quite easily.) Quantitative derivatives had to wait for the USAF/GE Annular Cascade program for such design derivatives (see reference 12). Derivatives for typical mid-stage cascades are shown in Figure 33, which is approximately 1° incidence improvement for 10% solidity reduction. Similar experimental data is now being generated in the subsonic stall regime for front stage blading with preliminary data indicating incidence margin loss with solidity reduction.

- Effects of Twist-Bend Coupling

F.O. Carta recognized the destabilizing effect of torsion-bending coupling in the open literature in 1966 (see reference 13). In that paper, structural coupling was derived from the kinematics of part-span shrouded blades with aspect ratios on the order of 3-4 (based on inferred data given in the paper). Furthermore, the constraints of the assumed system mode kinematics restrict the torsion-bending phasing to $-\pi/2$ (torsion lags bending). Nonetheless, this commendable work reactivated the interest of our existing empirical correlations of non-shrouded blading with aspect ratios in the range of 2-3 with torsion-bending phasing of 0°. These empirical data, presented in arbitrary scales in Figure 34, depict the subsonic stall flexural instabilities along with their composite torsional instability regimes in Figure 34a. Of significance is the diminishing critical reduced velocity threshhold, or "floor", with an increasing twist-bend parameter defined in Figure 34c. When these data, (which include the torsion threshhold floor), are cross-plotted against this parameter, it can be concluded that for this class of turboblading in the subsonic stall regime, the instability threshhold is strongly governed by the degree of twist-bend coupling. The transition from flexural to torsional modes is seen to relate to the twist-bend coupling parameter ($\phi b/\delta$). This is consistent with Carta (reference 13) and Bendiksen/Friedman (reference 14).

In this last reference, the authors state that "structural coupling is also believed significant in non-shrouded rotors due to high pretwist found in fan blades". Structural dynamicists recognize, however, that the twist-bend parameter is governed not only by the total twist but inversely with aspect ratio: i.e.,

$$\frac{\phi b}{\delta} = f\left(\frac{\text{Total Twist}}{\text{Aspect Ratio}}\right)$$

Additionally, recent low inlet radius-ratio designs with high hub ramp angles that result in relatively short trailing edges also contribute to increasing the twist-bend coupling.

We now review Figure 34 with concern, recognizing that a "rugged-looking", low aspect-ratio blade (with a large twist-bend parameter), if made thin enough, has the potential for a fundamental mode instability. Such a case has recently been reported with a blade whose aspect-ratio (based on root chord) was approximately 1.4.

As turboblading design trends continue towards thin, highly twisted, lower aspect-ratio configurations (shrouded or unshrouded) with design points of increasing relative Mach numbers and pressure ratio per stage, twist-bend coupling may become of prime importance in all regimes of instability including the unstalled supersonic regime indicated by Halliwell in reference 15. Halliwell indicates that "there are no ready techniques to overcome supersonic (system mode) flutter". However, General Electric experience indicates that the method implicit in the empirical data suggests that control of the twist-bend coupling parameter may be one important variable. Another parameter under consideration includes local leading-edge chordwise coupled deflection for low aspect ratio blading with thin leading edges.

While introducing this variable into the GE/USAF Annular Cascade program for systematic explorations has been considered, a similar approach must be made for full-scale research rigs, particularly for the supersonic regime. Such experimental programs can establish the sensitivity of the variables involved as well as correlation/evaluation of the emerging flutter codes.

3. Surge Stress-Response Characteristics

A principal objective in designing modern fans and compressors is to provide adequate surge margin for inlet pressure and temperature distortion, particularly with renewed interest in "nonrecoverable stall" of these new power-plants. Increasing experimental validation to establish existing and/or develop improved surge margin is of particular aeromechanical interest and concern.

During such testing, the surge-resulting overstress and related cumulative fatigue damage can cause failure in such abusive tests or limit the turboblade life in the production unit. Related deflections can also influence axial and tip clearance requirements.

Early development of high amplitude stress fatigue capabilities of materials and applications to overstress and fatigue reduction in turboblading was developed at General Electric by CE Danforth in 1959-1961 (reference 3). Successful results were achieved in his application of the technique soon after its development. One test compressor was interrupted for blade replacement of a particular stage because application of the procedure during on-line evaluation indicated loss of overstress capability. Subsequent fatigue strength reduction was as predicted. While the generalizations

and methodology are beyond the objectives of this paper, its application is illustrated only to emphasize the need for early aeromechanical test data.

a. Typical Behavior

Modes of vibration during the surge sequence described by Mazzaway (reference 7) are generally confined to the fundamental modes, with the first flexural mode predominant.

A typical engine stall response consists of multiple high-amplitude pulses with repetitive surging occurring until a flame-out or throttle chop occurs; or, in the case of a component, when the fast-acting discharge valve is opened. Damping in these uncomplicated "machine-gun" surges is in the order of .5-.10% log decrement, sufficient for each pulse to decay well below endurance limits prior to the next surge. Experience indicates surge frequency of 10-15 cps for machines with corrected flows of 5-50 lb/sec. Pulse duration is generally 1/4 the time for surge repetition. Usually, the first 2-3 pulses are the most damaging with each succeeding pulse diminishing the magnitude.

Not all compressors are typified by this characteristic, since their blade responses can be complicated by the presence of rotating stall (and bursts of instability) after the initial pulse.

b. Illustrative Examples

Overstress response in surge, given in percent of endurance limits, is shown in Figure 35 for both axial and centrifugal compressors. Note that the overstress occurs in the 90-100% corrected speed range, which is typical.

The axial compressor responses (Figure 35a) are typical of early low pressure-ratio machines with maximum overstresses of 1.5-2.5 (at 1/2 atmosphere) in the aft stages. Incidentally, this development configuration incurred an axial interference in stage 7 rotor when stalled at full density. The centrifugal inducer vane responses (Figure 35b) show multi-pulse diminishing stress response with the maximum loading of 140-155% limits occurring at 95% corrected speed. It is this unit for which the following overstress surge capability assessment is made.

c. Overstress and Fatigue Capability

A unit whose aeromechanical surge data shown in Figure 35b, incurred 33 stalls without incident of inducer vane cracks. An assessment of fatigue capability indicated a capability of 300-400 stalls with overstress ratios of the first 2-3 pulses in the 95% $N/\sqrt{\theta}$ range of 140-155% endurance limits. Subsequent testing of another unit to explore high speed stalls incurred 26 stalls. Upon teardown, cracks were detected at the leading edge root section of three vanes. These vanes were measured to be .006 inches under nominal thickness.

Parametric overstress capability was analyzed by modeling the surge-pulse as shown in Figure 36a, utilizing the overstress data from the instrumented test and applying corrections for local leading edge thickness. Results shown in Figure 36b, indicate potential crack initiation at 15-22 stalls compared to the actual 26 applied. The "fix" for this case obviously was a strength improvement to account not only for thickness tolerance, but also for in-field erosion.

Conclusion

The cost to produce and maintain advanced turbine engines is increasing and is tending to curtail development of new-technology, mission-oriented engine systems. The current trend is to provide a power plant for a new weapon system by producing derivatives of existing engines by the use of a common core, for example. When a "new" design or derivative version development is embarked upon it is recommended that present military specifications for engine qualification/verification be significantly modified to include aeromechanical procedures outlined in this paper. During previous development programs, the testing of component rigs, core engines, etc., has been accomplished with primary emphasis on performance and operability parameters such as component efficiency, specific fuel consumption, thrust, stall margin, etc. These parameters are important, but the justification for extensive development of multiple configurations to obtain small improvements is questioned. More emphasis should be directed toward aeromechanical life/durability and minimum maintenance costs.

This paper illustrates that, while design criteria do exist, the overall complexity of the problem of assuring structural integrity does necessitate experimental verification for each class of design. No realistic assessment of durability can be made on a sea-level test stand. The necessity to develop and verify integrity as a part of the qualification/acceptance process prior to production cannot be overemphasized. Test facilities must have the capabilities to extensively stress map engines up to and beyond the temperature and pressure extremes of the intended flight envelope in order to insure that the engine has vibratory strength capability and growth potential for adapting the engine to changing requirements within the intended application.

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Mr. C.E. Danforth, during his 35 years of service to the General Electric Company, pioneered the aeromechanics technology with remarkable perspective and energy. The authors acknowledge not only his total contributions, but his sound approach in fundamental aeromechanical design and experimental verification procedures upon which this material is based.

Appendix

Figures A-1, A-2, and A-3 are graphical representations of engine inlet conditions as functions of total temperature, total pressure, mach number and altitude. The data are based on the following:

(1) Fig A-1 Standard Day Data - U.S. Standard Atmosphere, 1962

(2) Fig A-2 Cold Day Date - Mil. Std. 210A, 2 August 1957, Climatic Extremes for Military Equipment, Table II

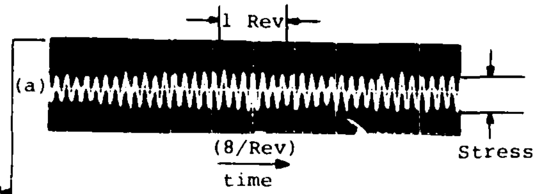
(3) Fig A-3 Hot Day Data - Mil. Std. 210A, 2 August 1957, Climatic Extremes for Military Equipment, Table III

(4) Isentopic Ram Recovery

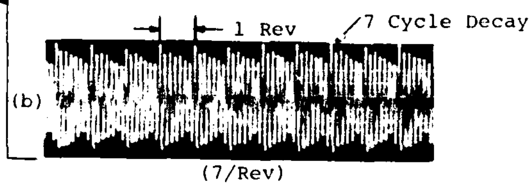
Table 1: Strain Gage Signal Waveform Characteristics and Interpretations

RESONANCE

Aerodynamic

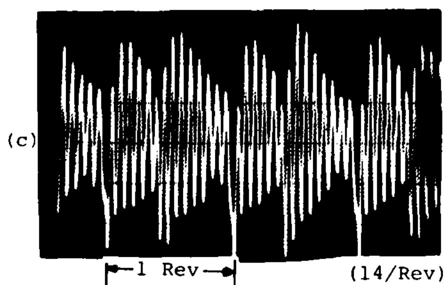


Constant amplitude at a given rotor speed due to first harmonic integral order blade resonance.



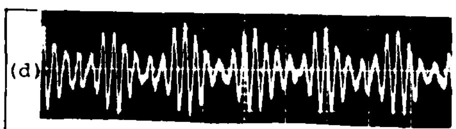
Non-constant, cyclically varying amplitude at a given rotor speed due to seventh harmonic integral order blade resonance. The rate of signal decay is a function of the blade system damping.

Mechanical

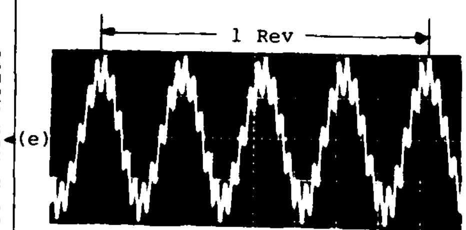


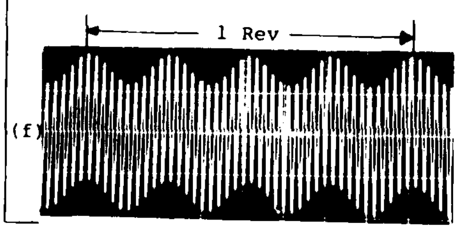
Blade tip rubs produce waveforms similar to waveform (b) with cyclic decay evident between rubs. This waveform resulted from two rubs separated by approximately 154°.

Combined Modes



Regularly occurring maxima and minima with a sinusoidal envelope of amplitude (beating) whose period is the difference of the two excitation frequencies.

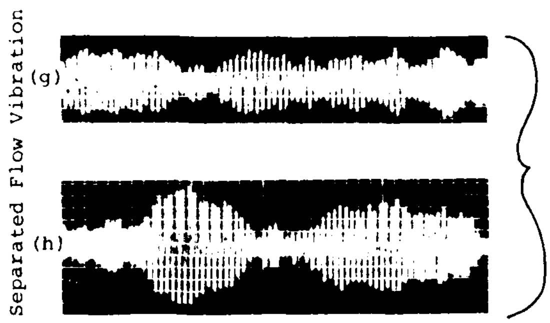




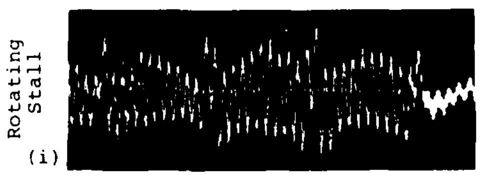
Multiple modes with a large difference in frequencies responding simultaneously. In waveform (e), the amplitude of the higher frequency is one-fourth that of the lower frequency. It clearly appears as the higher frequency response superimposed on the lower frequency response. In waveform (f) the amplitude of the higher frequency is four times that of the lower frequency.

Table 1 (con't): Strain Gage Signal Waveform Characteristics and Interpretations

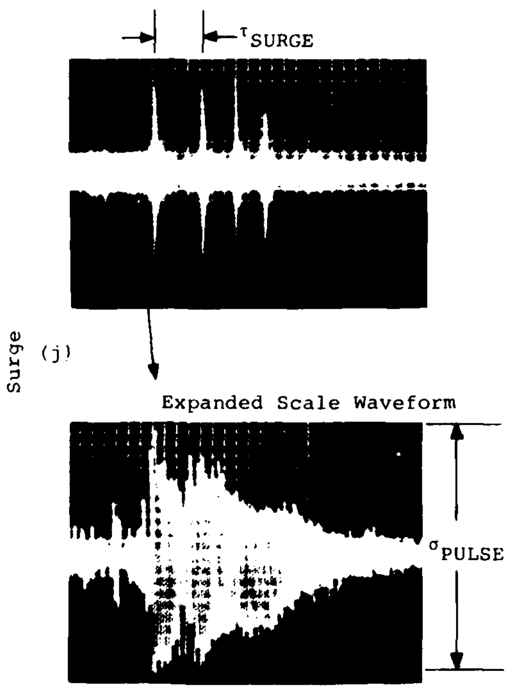
INDUCED FLOW VIBRATIONS



These two waveforms show randomly modulating amplitudes resulting from slightly turbulent or separated flow in the upper waveform to violent turbulence or separation in the lower.



A waveform typical of rotating stall where the blade amplitude suddenly increases as the blade encounters each stall cell and decays.

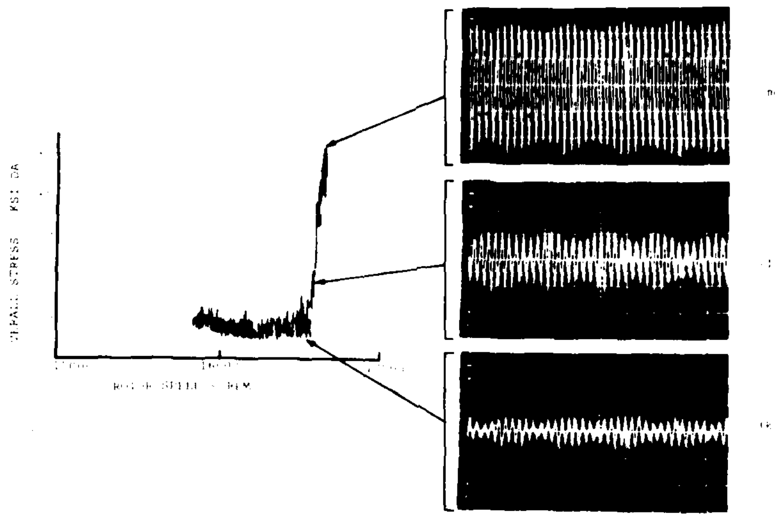


Waveform characteristic of a typical 4-pulse repeating surge ("machine-gun" surge) with surge frequency of 13 CPS (τ_{SURGE}=.075 sec). The maximum overstress ratio ($\tau_{PULSE}/\tau_{ENDURANCE}$) approached 1.6 at 95 N/$\sqrt{\theta}$. From the expanded pulse waveforms, average cascade damping is .07 log decrement. Response frequency is usually the fundamental flexural mode.

NOTE: The above waveform examples are also typical of progressive characteristics during cascade migration towards stall.

Table 1 (cont'd):

SELF EXCITED VIBRATION

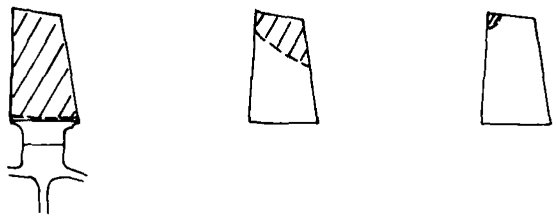


Typical waveform characteristics of self-excited vibration during cascade migration into instability regime. Initiation is usually characterized by separated flow vibration progressing rapidly to sinusoidal waveform amplitude at the airfoil's natural frequency.

Table 2

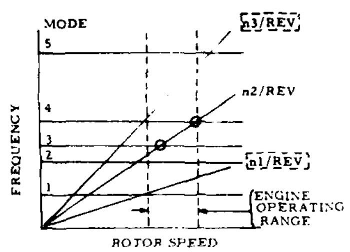
METHODOLOGY FOR MINIMIZING RESONANT RESPONSE

METHOD	EXAMPLE OF IMPLEMENTATION
• FREQUENCY CONTROL	- BLADE TAPER - SHROUDING, AIRFOIL ATTACHMENT - CHORD LENGTH, SOLIDITY - ASPECT RATIO, RADIUS RATIO - UTILIZE PROVEN FINITE ELEMENT ANALYSIS TECHNIQUES
• STIMULUS CONTROL	- SELECTION OF NUMBER VANES/BLADES - STATOR SCHEDULE ADJUSTMENTS - DISTORTION LEVELS/PATTERNS - BLEED PORT SPACING/QUANTITY - INDEXING/MAPPING TO AVOID REINFORCEMENT & SINGULAR HARMONICS
• DAMPING	- MECHANICAL DAMPERS - STATOR SHROUD RINGS
• STRUCTURAL STRENGTH	- "BEEFING-UP" BLADE/VANES TO REDUCE VIBRATORY AND/OR STEADY STATE STRESSES - REMOVE/REDUCE STRESS CONCENTRATIONS - MATERIAL SELECTION FOR INCREASED HCF CAPABILITY

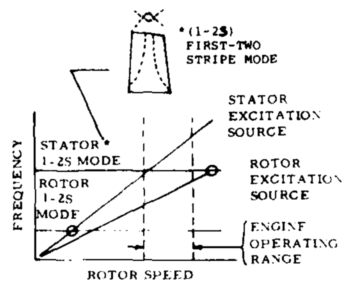


FAILURE:	• LOSS OF ENTIRE AIRFOIL AND/OR: AIRFOIL DOVETAIL AND/OR DISC DOVETAIL	• LOSS OF OUTER PORTION OF AIRFOIL	• LOSS OF SMALL AIRFOIL TIP FRAGMENT.
MODE:	• FUNDAMENTAL BENDING (AXIAL OR TANGENTIAL) AND TORSION	• SECOND TORSION • HI ORDER BENDING	• PANEL (STRIPE) MODES • *HI ORDER TORSION*
IMPLICATION AND IMPACT:	• UNBALANCE • SIGNIFICANT VEHICLE FOD • CONTAINMENT • *HIGH REPLACEMENT/* REPAIR	• SIGNIFICANT REPLACEMENT/REPAIR	• MINOR BENCHING • MINIMAL REPLACEMENT

Figure 1 - Forced Vibration - Damage Severity Criteria/Considerations



STIMULUS SELECTION



RELATIVE ROTOR/STATOR ASPECT RATIO SELECTION

SIGNIFICANCE:

- TAKE ADVANTAGE OF MODE SPACING TO SELECT STIMULUS.
- RESONANCE AVOIDANCE WITHIN OPERATING RANGE (*1F , n1 or n3 WERE CHOSEN* AS NUMBER OF VANES).

SIGNIFICANCE:

- LOW ASPECT RATIO ROTOR (FEW BLADES, LOW 1-2S MODE FREQUENCY) AND HIGH ASPECT RATIO STATOR (MANY VANES, HIGH 1-2S MODE FREQUENCY) COMBINE FOR OPTIMUM *1-2S MODE PLACEMENT FOR BOTH.*

Figure 2 - Schemes for Frequency Tuning

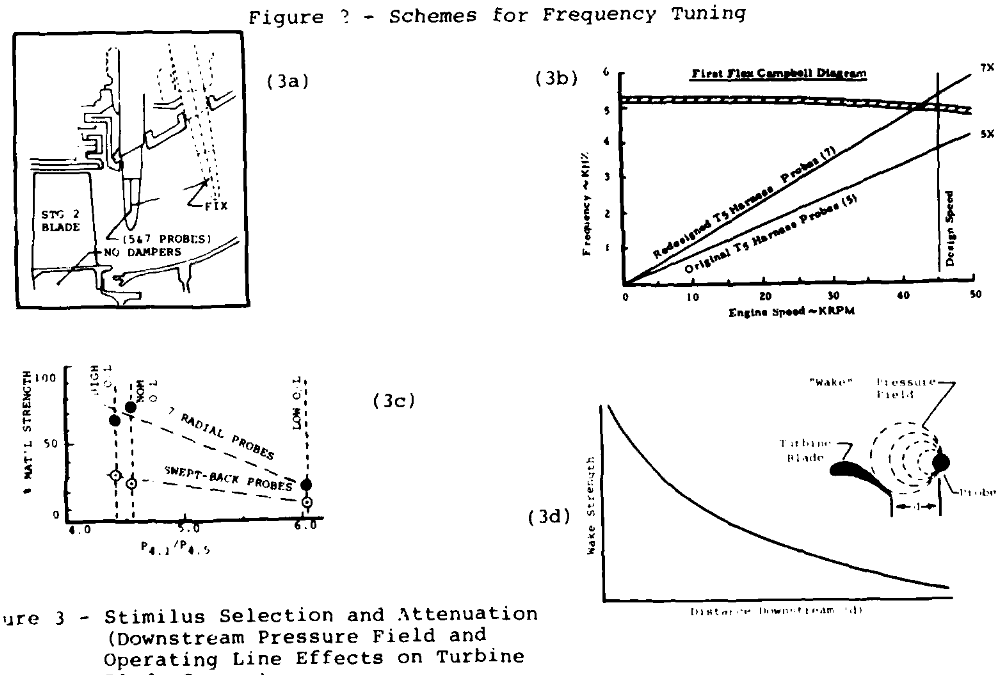


Figure 3 - Stimilus Selection and Attenuation (Downstream Pressure Field and Operating Line Effects on Turbine Blade Stress)

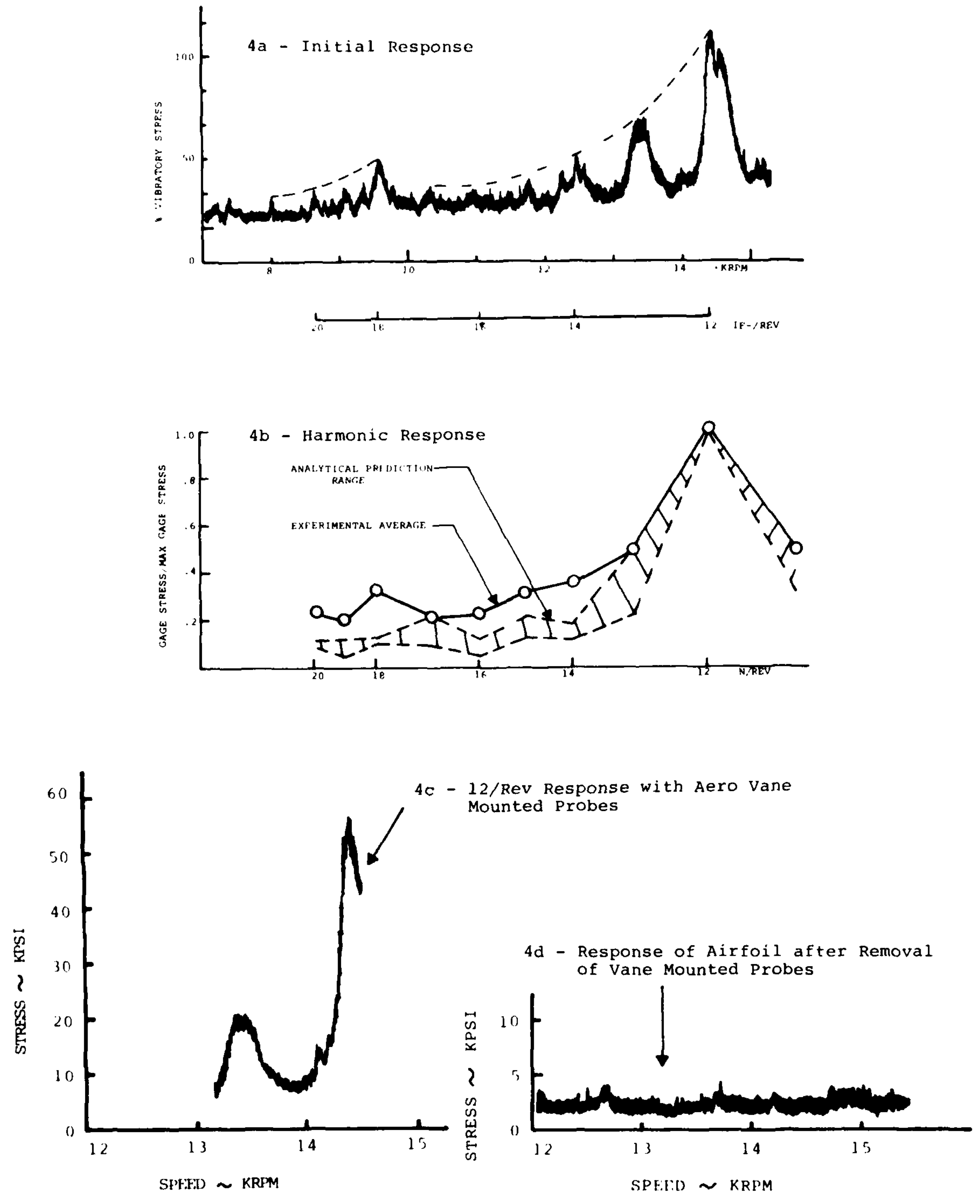


Figure 4 - Harmonic Content - Internal Distortion from Aerodynamic Vane-Mounted Probes

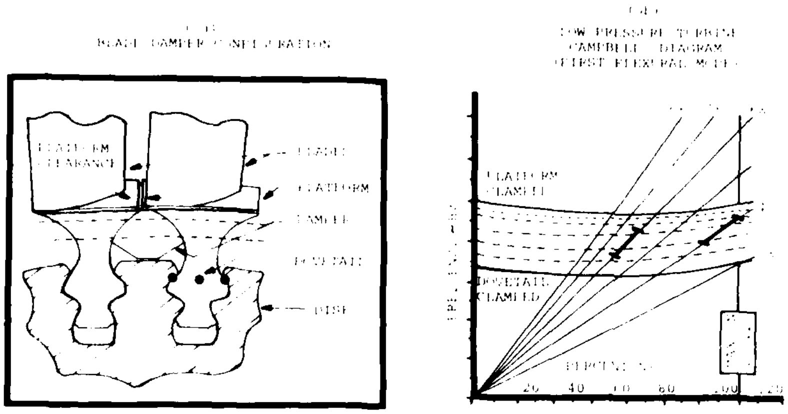
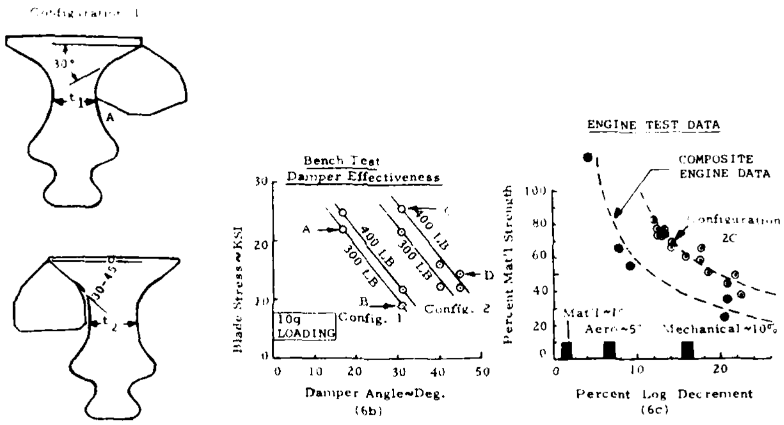


Figure 5 - Mechanical Damper Design - Turbine Interblade Seal and Damper



A. Initial Development Engine
B. Modified Development Engine
C. Initial Production Engine
D. Qualified Production Engine

Figure 6 - Turbine Mechanical Damper Effectiveness
(First Flexural Mode - 4/Rev Response)

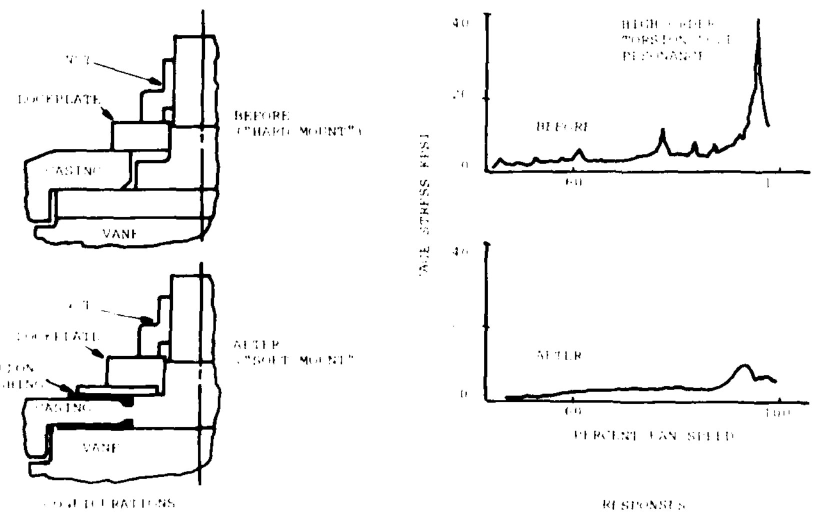


Figure 7 - Damping Effectiveness - Fan Variable Stator

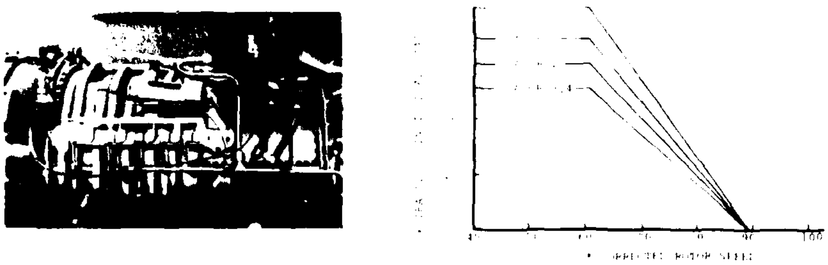


Figure 8 - Typical Variable Geometry (V.G.) System and Operation

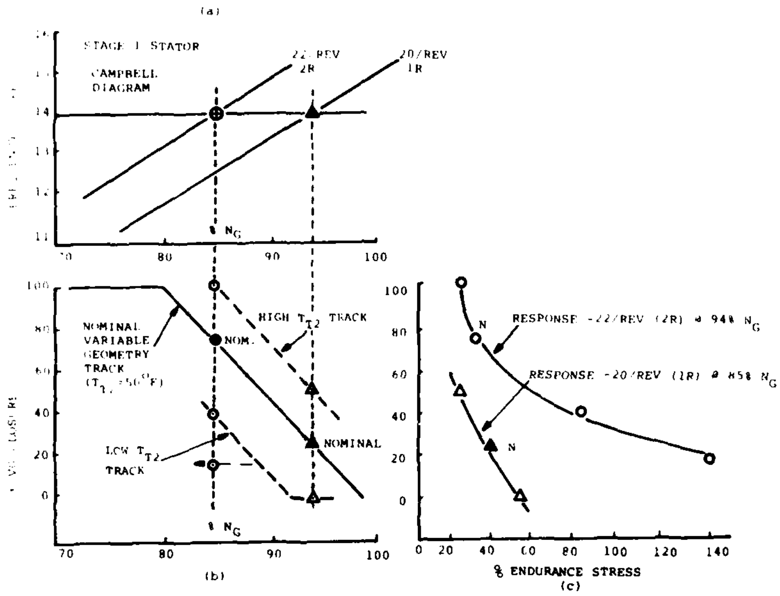


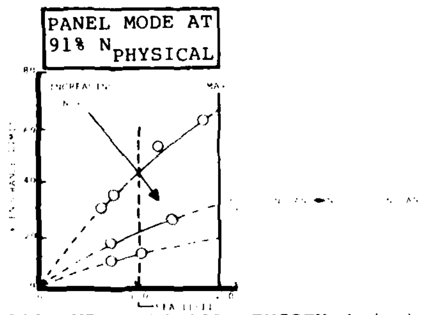
Figure 9 - Resonant Response Sensitivity to Variable Geometry System

1. EARLY COMPONENT TESTING TO EXTREMES OF INLET PRESSURE/TEMPERATURE RANGES OF FLIGHT REGIME REQUIRED FOR RESONANCE ASSESSMENT.

CHARACTERISTIC	DEPENDENCY
• RESONANT SPEED	MODE, N PHYSICAL...(N_P)
• VIBRATORY FORCE	FLUID DENSITY*...(ρ)
• DAMPING	MODE SHAPE, DENSITY
• STIMULUS (V.G. POSITION)	CORRECTED SPEED ($N/\sqrt{\theta}$)

*...FOR A GIVEN STAGE LOADING AND STIMULUS

2. TYPICAL EXAMPLE OF RESONANT RESPONSF OVER FLIGHT MAP RFGIME. (PANEL MODE RESONANCE WITH I.G.V. PASSING FREQUENCY)



RELATIVE INLET AIR DENSITY (ρ/ρ_o)

Figure 10 - Flight Regime (P_{T2}-T_{T2}) Effects

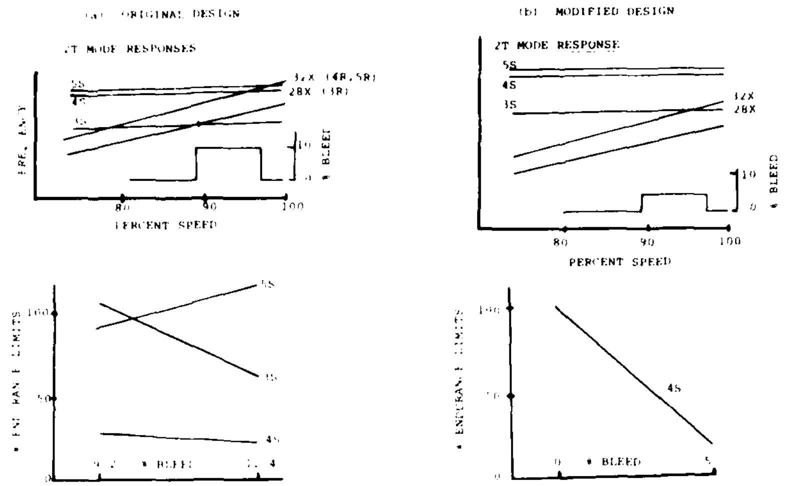
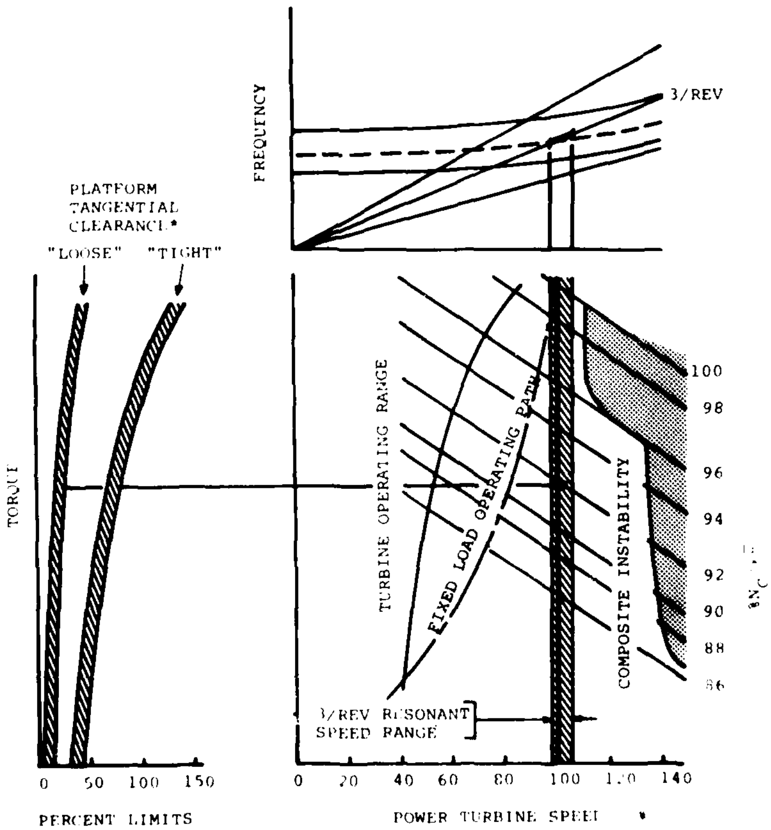


Figure 11 - Compressor Interstage Bleed Effects

(a) Campbell Diagram



(b) RESONANT STRESS - TORQUE DEPENDENCY

*REFERENCE: Figure 5a

Figure 12 - Torque Dependency of Resonant F...
Power Turbines of Turbo-Shaft E...

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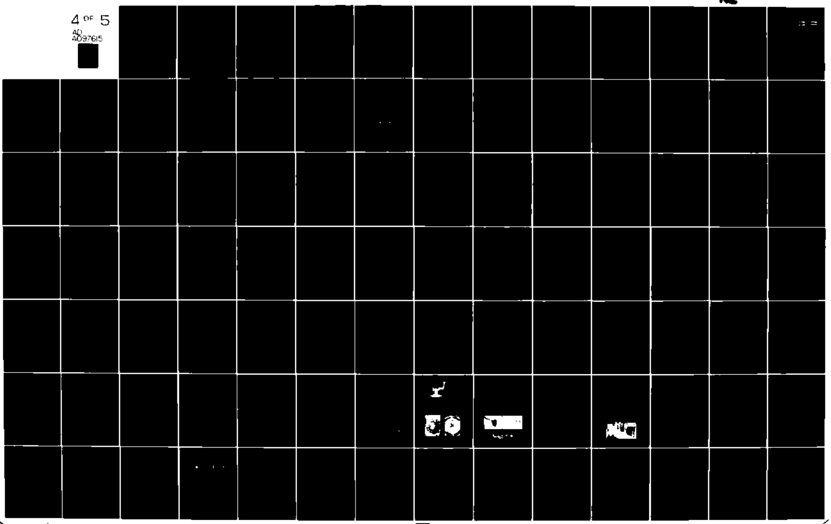


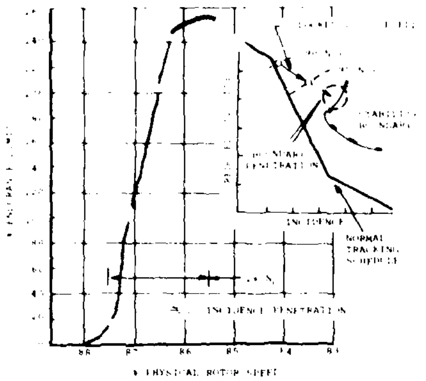


Figure 13 - Self Excited Vibration Response Severity During Boundary Penetration

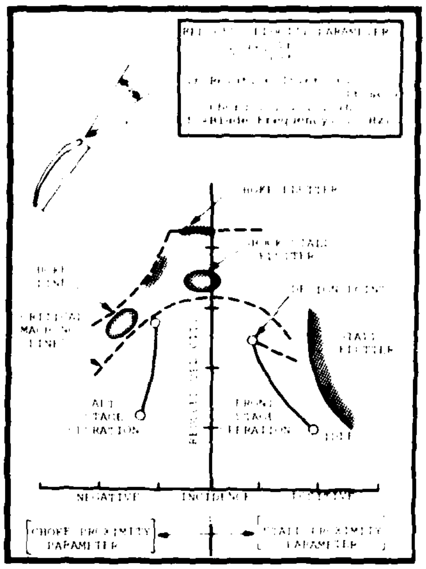


Figure 14 - Idealized Instability Map

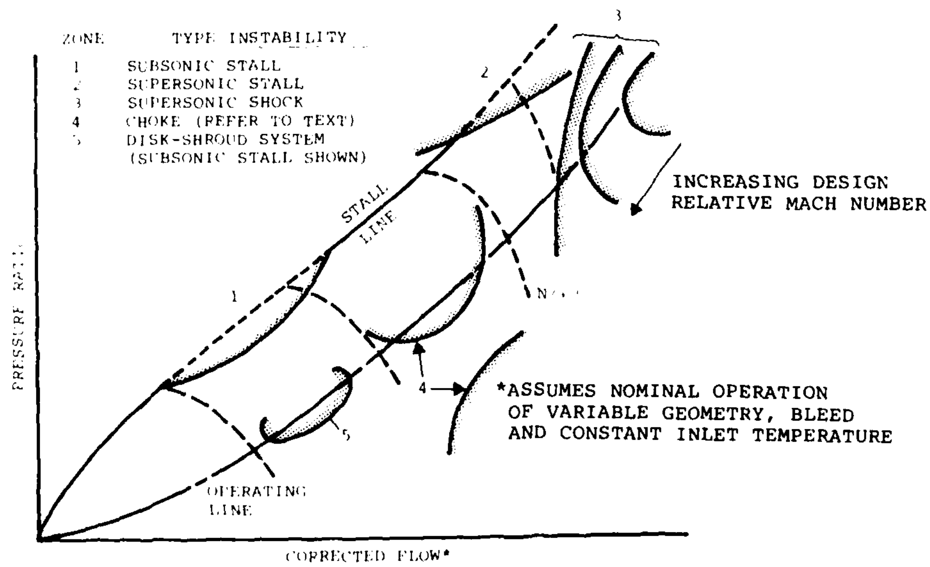


Figure 15 - Compressor Map Representation (Composite Stability Boundaries)

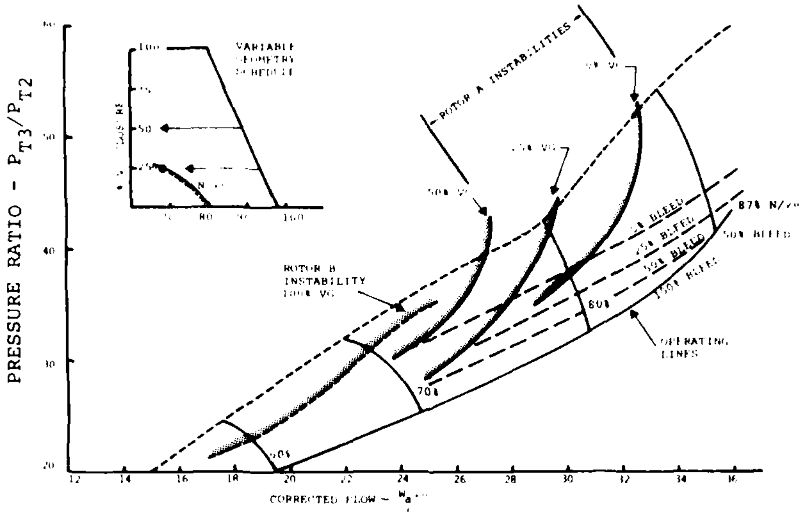
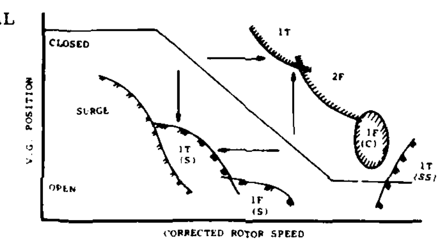


Figure 16 - Compressor Stability Boundary Map for Various V.G. Positions

OPERATING LINE: NOMINAL
INLET TEMP: CONSTANT



(C) Choke Instability
(S) Stall Instability
(SS) Supersonic Shock Instability

Figure 17 - Variable Geometry Map

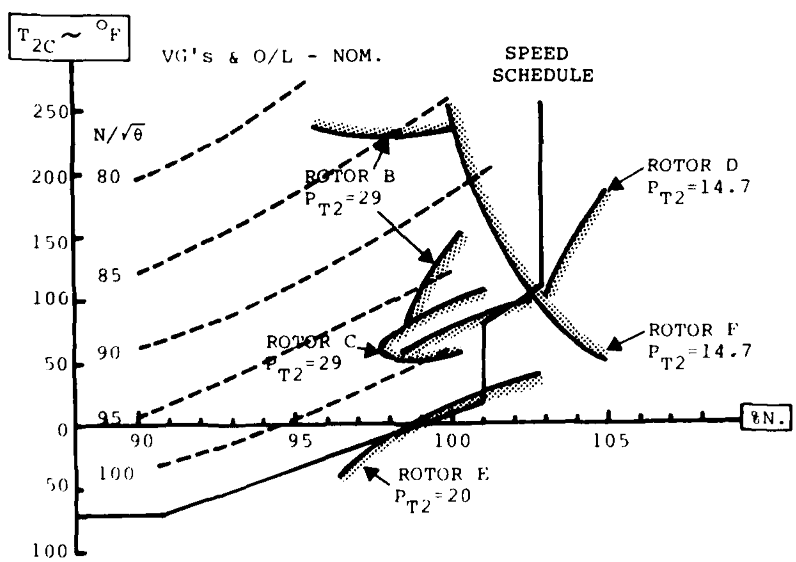
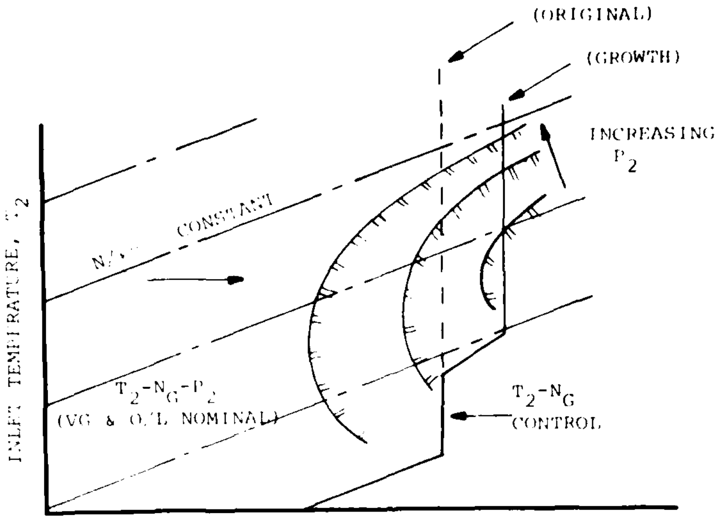
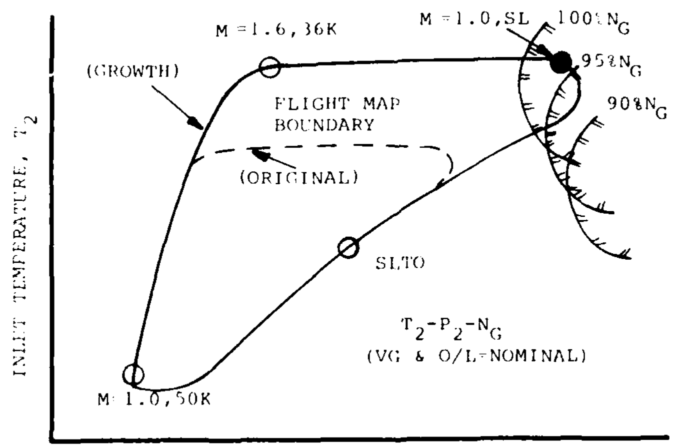


Figure 18 - Temperature-Speed-Pressure Map
(T_{T2}-N_G-P_{T2})



(a) N_G-T_{T2}-P_{T2} MAPPING



(b) FLIGHT MAP

Figure 19 - Flight Map Evaluation

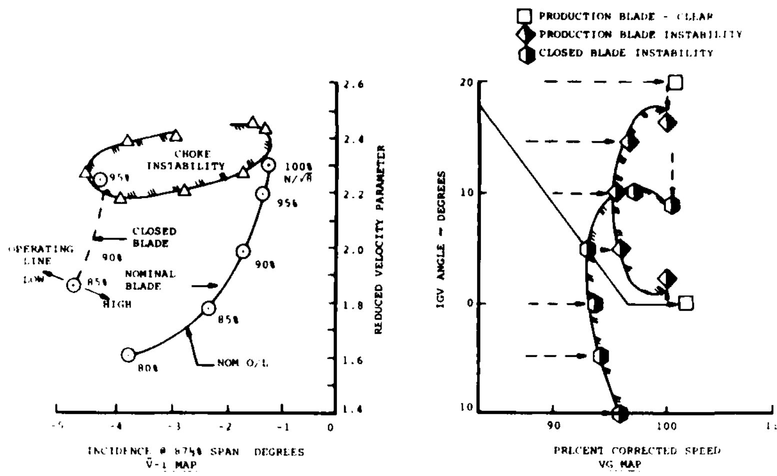


Figure 20a

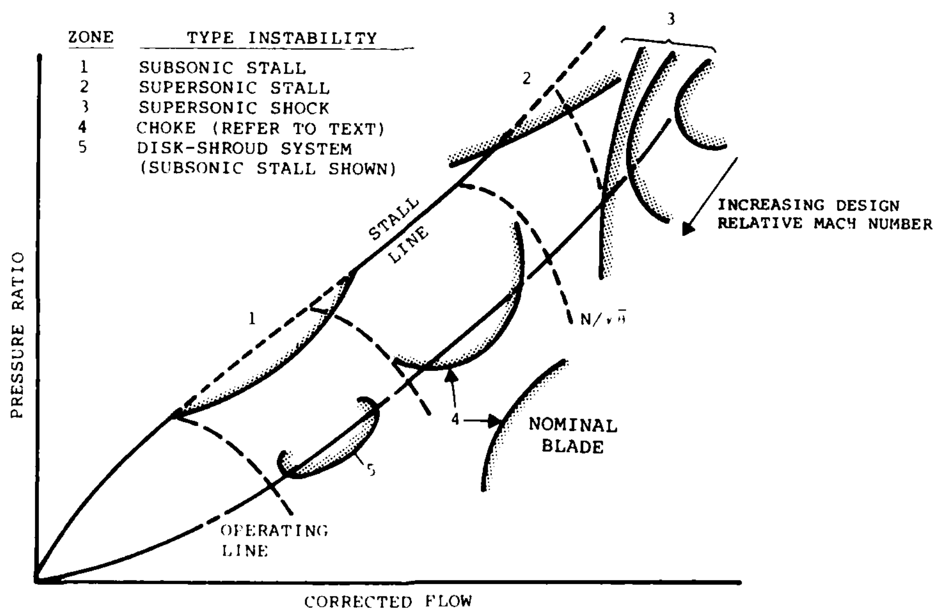


Figure 20b

Figure 20 - Compressor Map Representation Anomaly

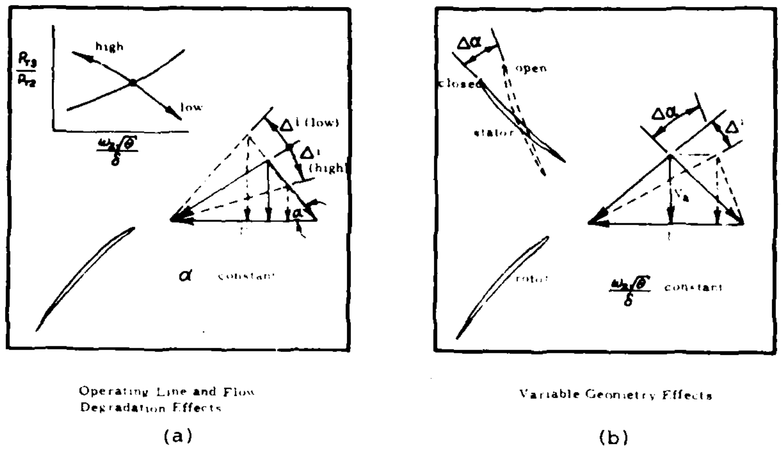


Figure 21 - Examples of Incidence Migration Mechanisms

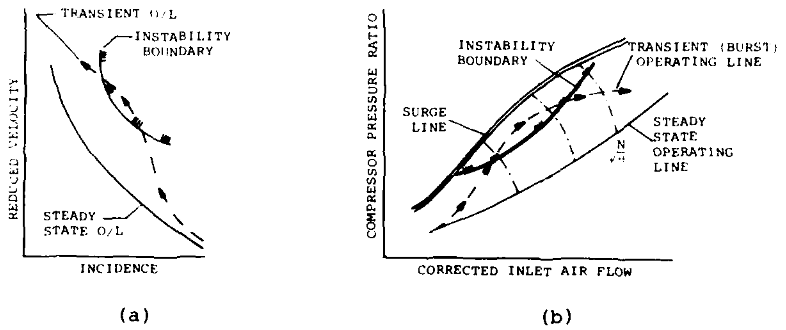


Figure 22 - Example of High Operating Line Effects
(Transient Burst)

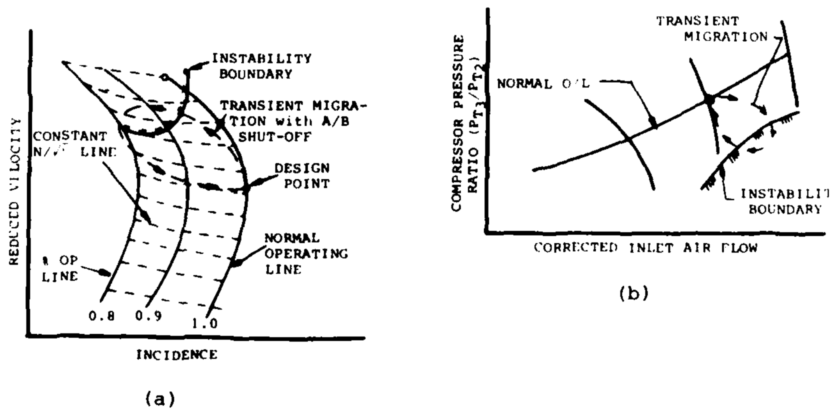


Figure 23 - Example of Low Operating Line Effects
(Afterburner Shutoff with A_8 Lag)

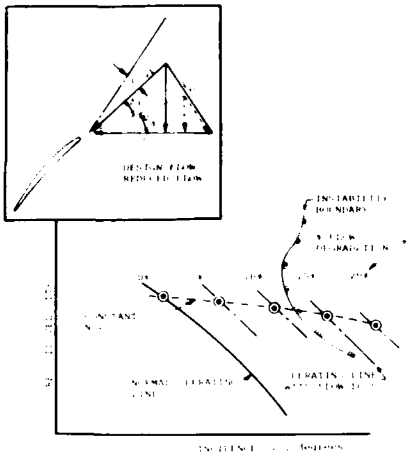


Figure 24 - Flow Degradation
Migration Mechanism

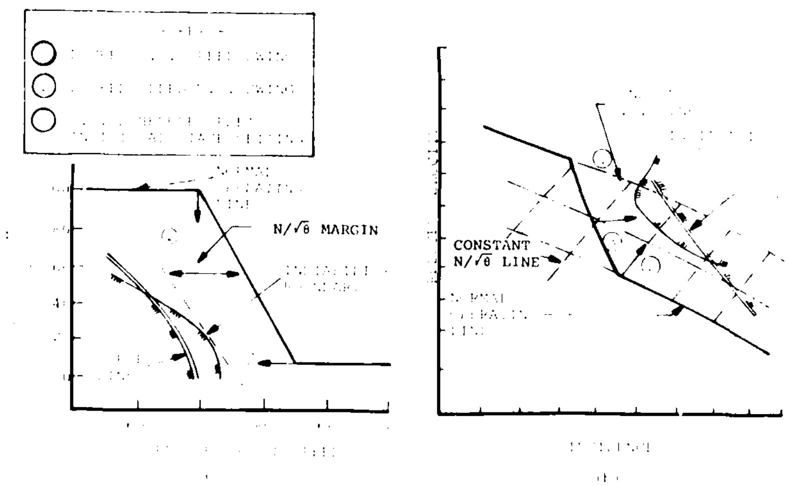


Figure 25 - Variable Geometry Mapping Techniques

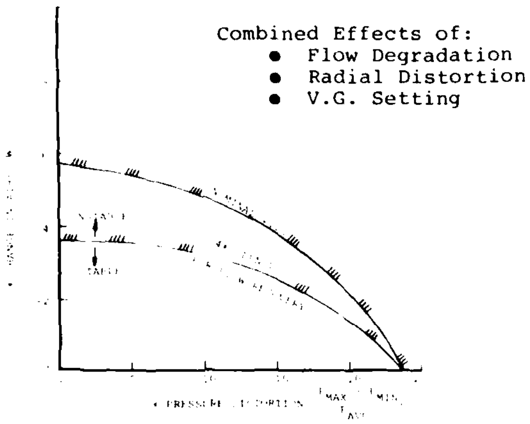
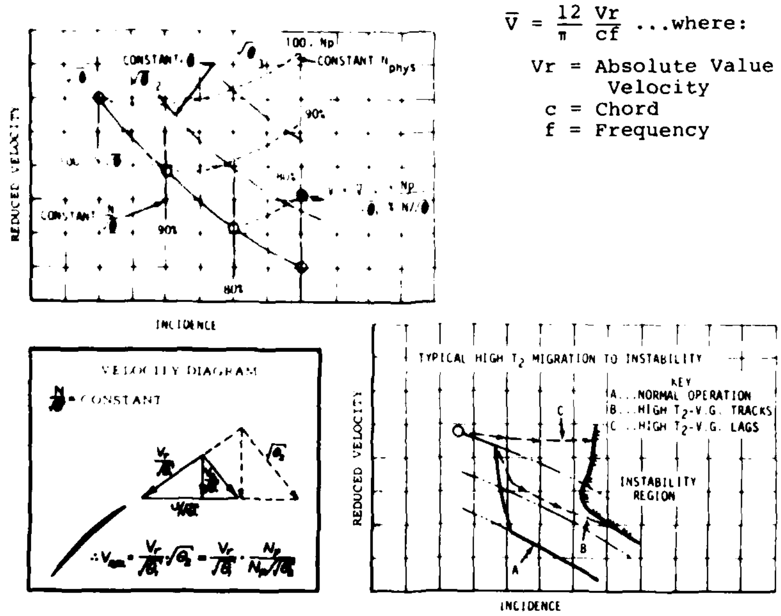


Figure 26 - Typical Instability Margin Loss



$$\bar{V} = \frac{12}{\pi}\,\frac{Vr}{cf} \text{ ...where:}$$

Vr = Absolute Value of Relative Velocity
c = Chord
f = Frequency

Figure 27 - Elevated T_2 Migration Mechanism

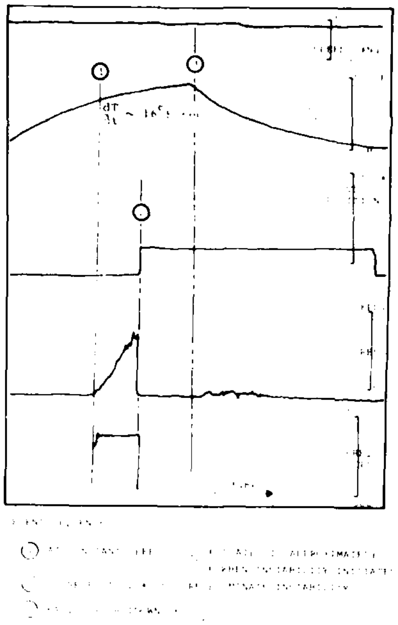


Figure 28 - Experimental Verification of Elevated T_{T2} Migration Mechanism and V.G. Effectiveness

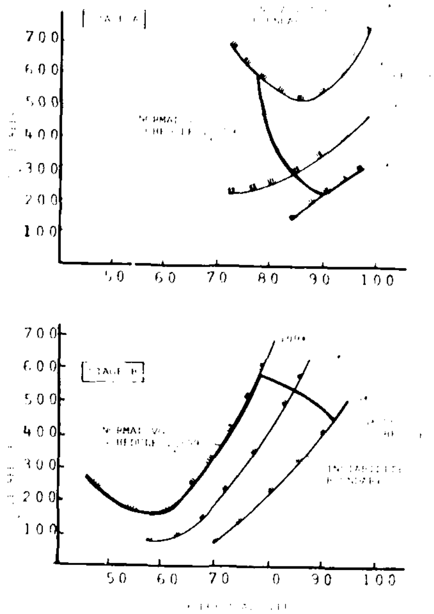


Figure 29 - Elevated T_{T2} Stability Margin vs. V.G. Schedule

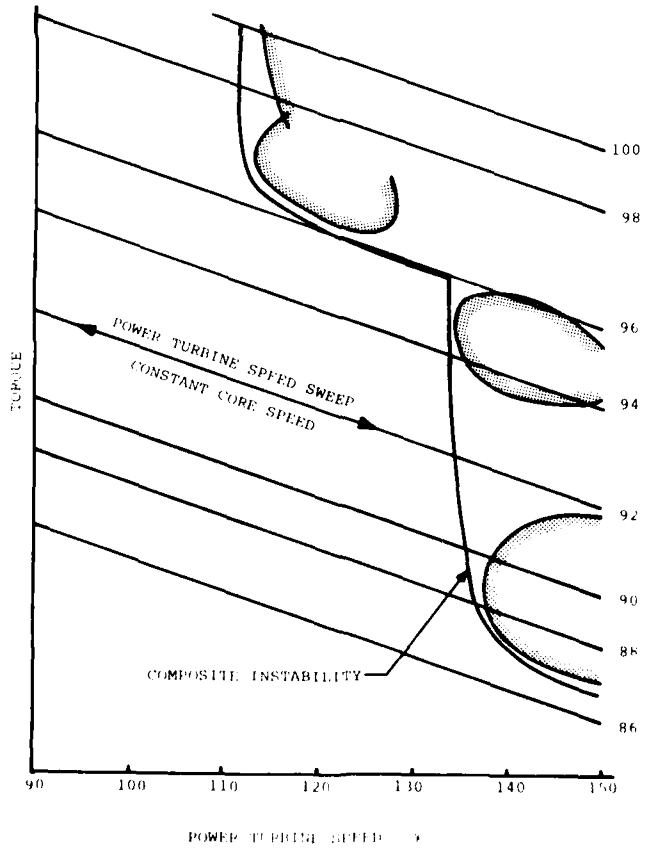


Figure 30 - Power Turbine Blade Torsion Instability

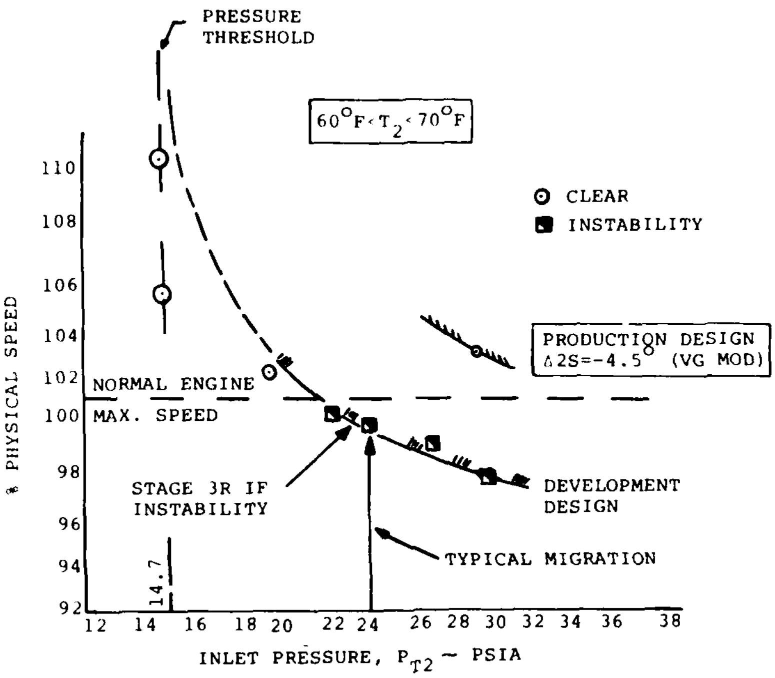


Figure 31 - Instability Sensitivities (Pressure; V.G. Effectiveness)

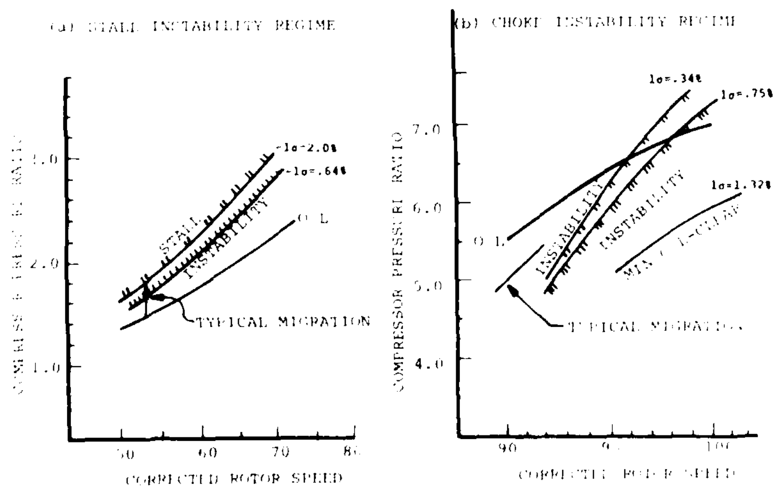
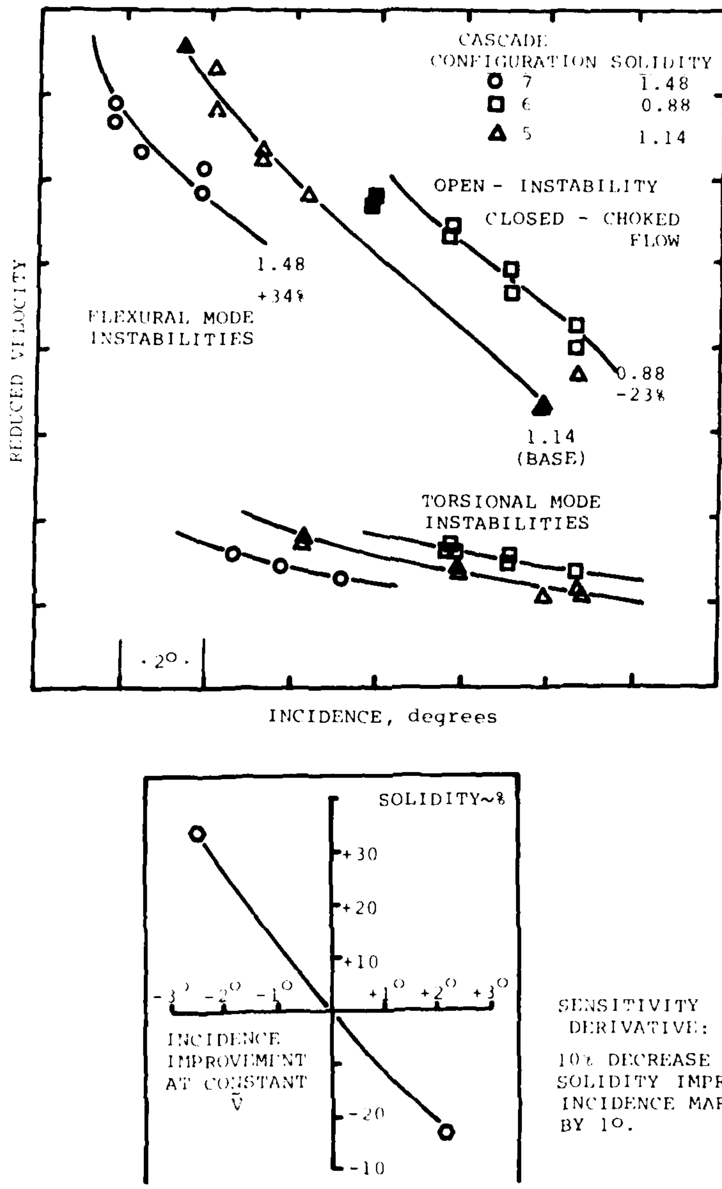


Figure 32 - Instability Sensitivity to Cascade Frequency Deviation
(1 Standard Deviation $=1\sigma$)



SENSITIVITY DERIVATIVE:

10% DECREASE IN SOLIDITY IMPROVES INCIDENCE MARGIN BY 1°.

Figure 33 - Instability Sensitivity to Cascade Solidity - Choke Regime

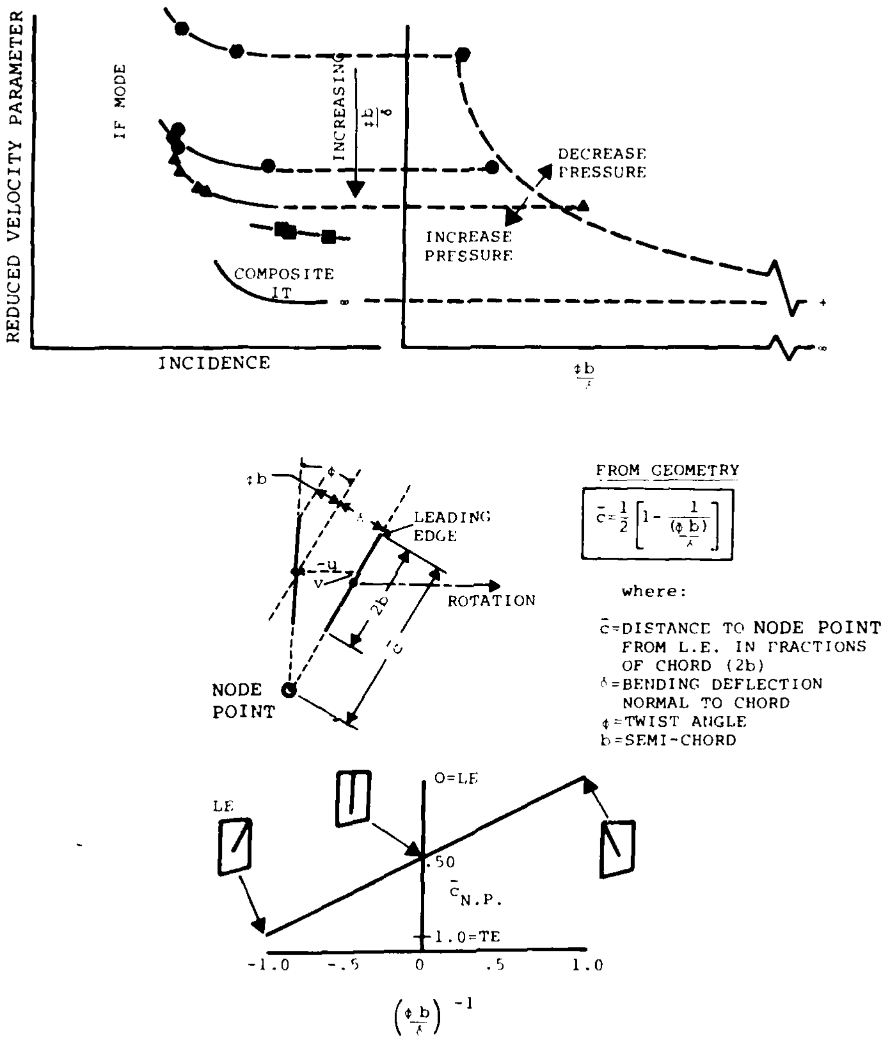


Figure 34 - Instability Sensitivity to Twist-Bend Coupling
(Cantilevered Blading)

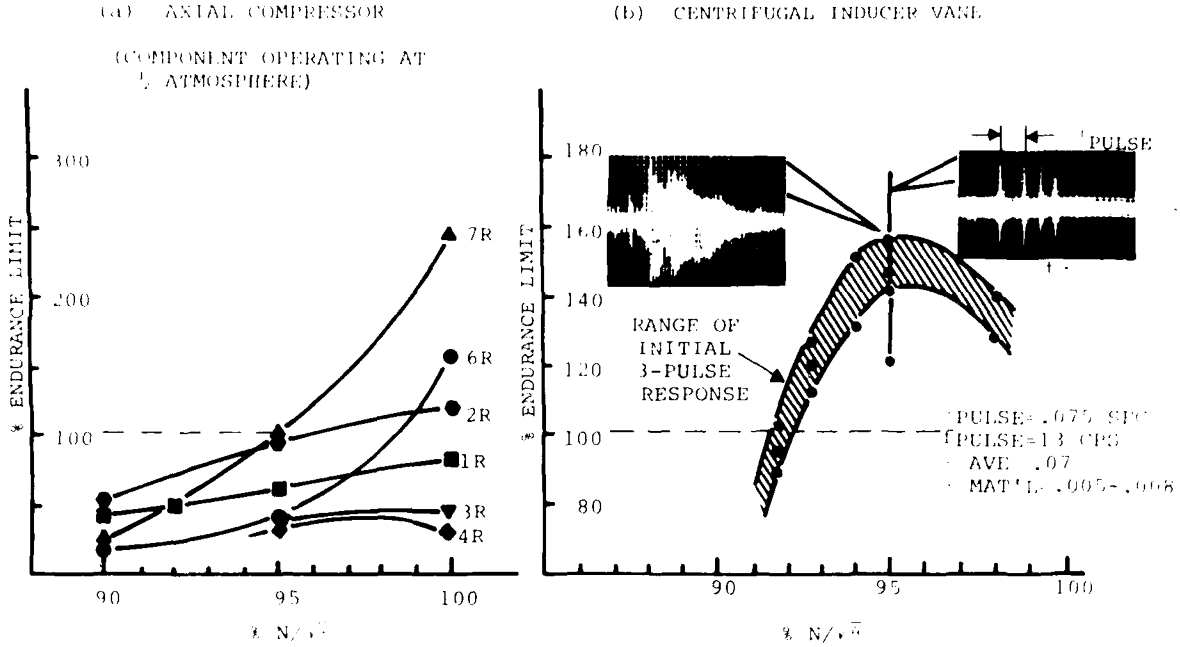


Figure 35 - Surge Pulse Vibratory Stress Severity - Typical Examples

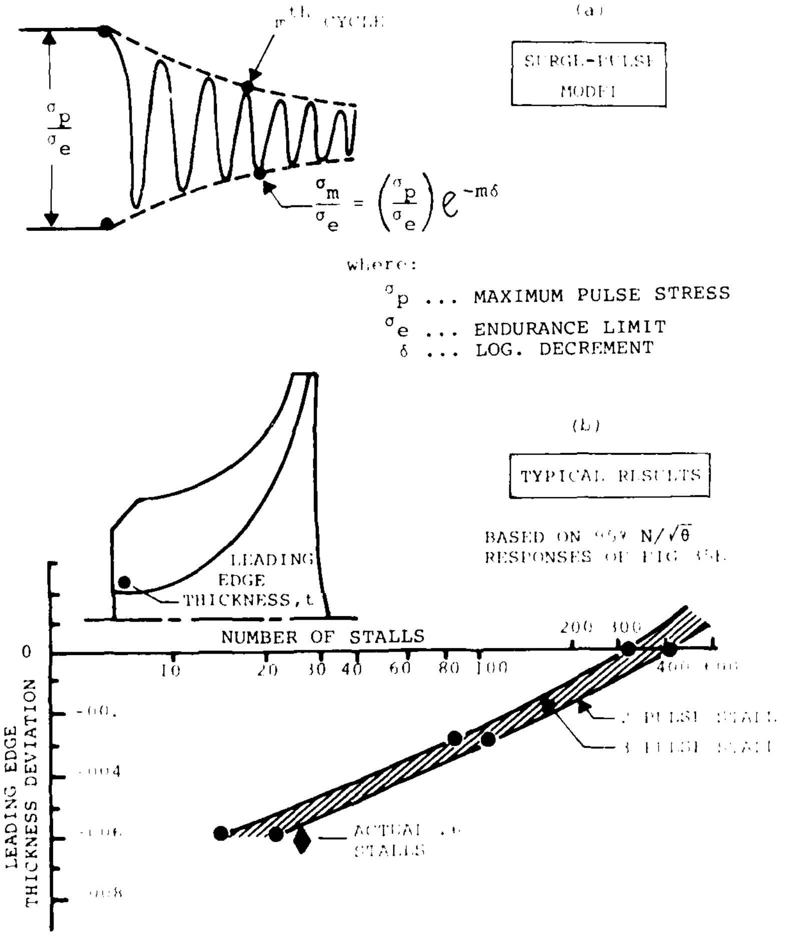


Figure 36 - Overstress Capability

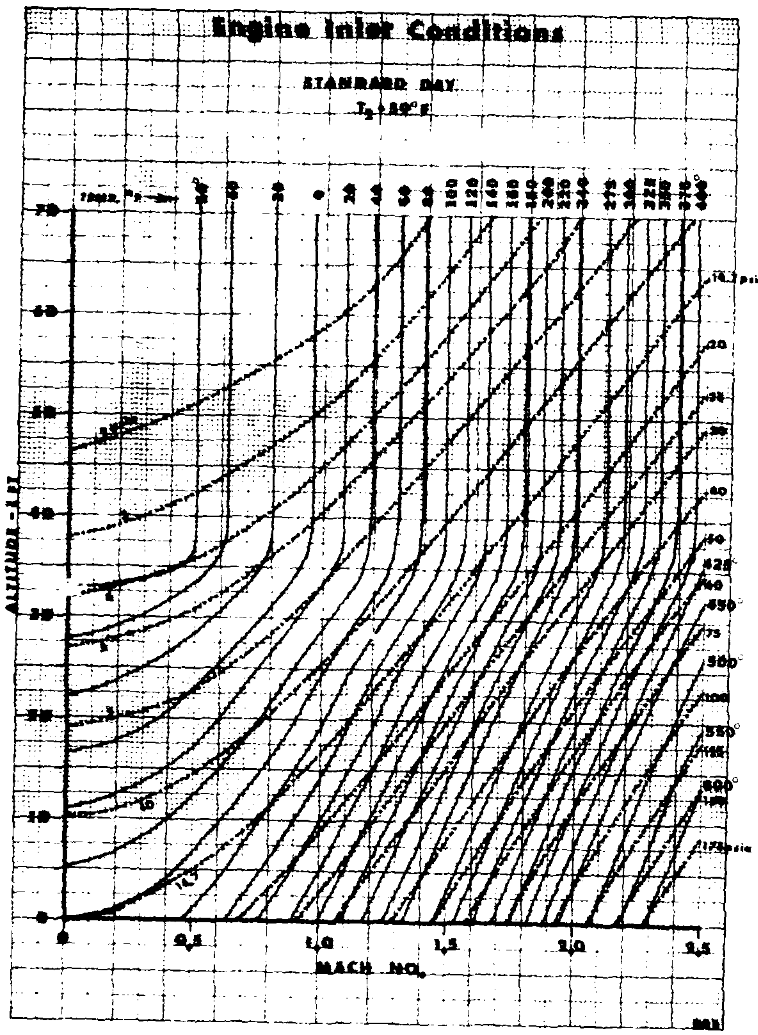


Figure A-1

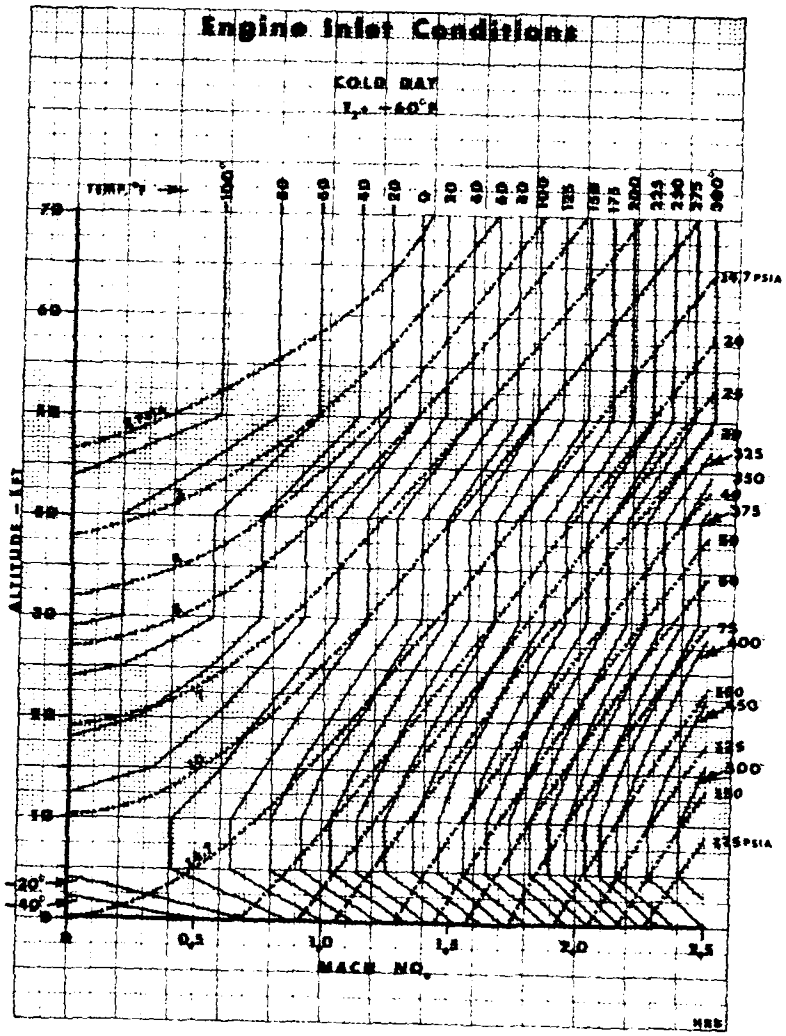


Figure A-2

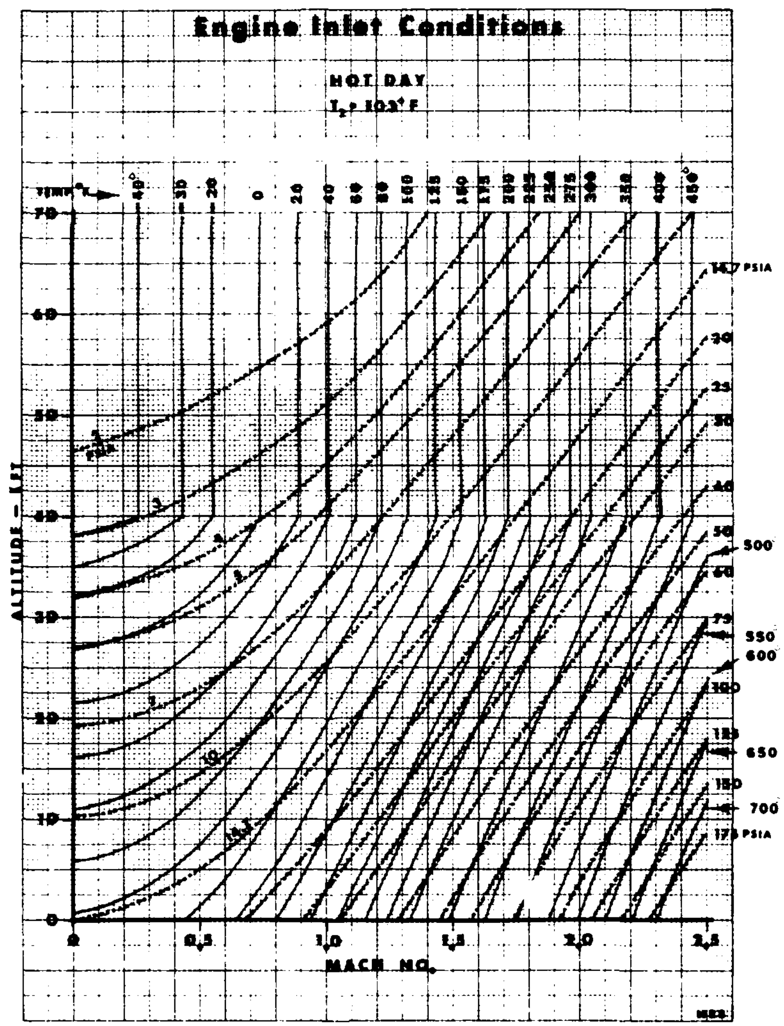

Engine Inlet Conditions
HOT DAY
T_2 = 103°F
TEMP °F
-40°
-35
-20
0
20
40
60
80
100
125
150
175
200
225
250
275
300
350
400
450°
14.7 PSIA
20
25
30
40
50
500
60
75
550
600
100
125
650
150
700
175 PSIA
ALTITUDE — KFT
70
60
50
40
30
20
10
0
0.5
1.0
1.5
2.0
2.5
MACH NO.

Figure A-3

BENCH TESTING OF A VECTORED THRUST ENGINE

R H BLAKE
MANAGER - TEST OPERATIONS

ROLLS-ROYCE LIMITED
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The Rolls-Royce Pegasus engine with its unique four exhaust nozzle thrust vectoring demanded special installations and test equipment to enable testing of Development and Production engines to be carried out close to an industrial complex without any outside influence.

The paper describes the history of plant development to accommodate the test programmes, including the need for thrust vector measurement and exhaust gas collectors to allow nozzle swivelling without hot gas re-ingestion.

The evaluation of plant effects on engine performance and the philosophy adopted for simplified Production testing in the horizontal thrust mode only are described.

1 HISTORY AND ENGINE DESIGN EVOLUTION

The significant milestones in the history of the Pegasus vectoring thrust engine, designed specifically for a single-engined VTOL aircraft, as shown in Figure 1, date back into the 1950's with the first engine run in 1959. Flying began in the following year with tethered hovering. Conventional flight trials followed and in-flight transition was accomplished in 1961.

The Kestrel aircraft was developed directly from the prototype work with an improved Pegasus and service trials were carried out with a special tripartite squadron by three nations (UK, USA and West Germany), where the operational practicability of the vectoring principle was proved. The Harrier entered service in 1969 as an operational aircraft with the RAF and has subsequently joined the US Marines and Spanish Naval Forces.

Progressive development has continued, from the original Pegasus 1 engine in 1959, to the Pegasus 103 and 104 in service today in the Harrier and Sea Harrier. Thrust has increased by almost 140% and the thrust-to-weight ratio by 60% without any significant change in dimensions.

The obvious need for the centre of gravity of the aircraft to be coincidental with the centre of thrust of the engine demanded close collaboration between the airframe and engine manufacturer. Thus, the engine design and its development had to relate very closely to any aircraft changes, and vice versa.

When in hover, stabilising jets using engine bleed air are used to correct for asymmetric aerodynamic loads and for manoeuvring, as shown in Figure 2.

2 SPECIAL PLANT REQUIREMENTS

From the aforementioned, it is clear that special test plant and techniques were being evolved along with the initial engine trials. The first thrust vectoring was carried out on an open-air plant with the nozzles swivelling from horizontally rear to vertically up to avoid hot gas ingestion. Forward and downward thrusts were measured but the latter proved difficult to obtain with equal accuracy because of the additional load of the engine test frame and thrust cradle (12,000 lb) on the thrust capsules, ie in order to measure 9,000 lb thrust vertically to 0.25% accuracy demanded a capsule measuring 21,000 lb with ± 22 lb accuracy, ie 0.1%, which was not then available.

It also demanded a different build standard of engine with respect to the nozzle rotation to that of the aircraft installation.

The need to test Development engines close to the manufacturing and engineering base and to accelerate the programme demanded an enclosed plant capable of the full engine range of conditions at all times without environmental, weather and noise restrictions. Accurate measurement of thrust in horizontal and vertical planes, which also enabled evaluation of the thrust centre position, was essential.

The requirements were further complicated by a stringent endurance schedule. A typical cycle is shown in Figure 3a. This illustrates the numerous cycles required to test and demonstrate the life for an engine of this type of application. This naturally gave problems of life of the collector duct system which necessitated modifications to prevent cracking, but the basic design has not been changed.

Details of the limited life ratings necessary for the engine operation and the complexity of the Parts 1, and 6 of the endurance test are shown in Figures 3b and 3c. In order to record the engine conditions throughout these parts, a special continuous recorder was necessary.

3 GENERAL DESCRIPTION OF PLANT

A very large plant, shown in Figure 4, was built and commissioned in 1962, and a second in 1964 (which also had the capability to test vertically-mounted lift engines, if the need arose), which incorporated exhaust collector ducts to enable thrust vectoring from horizontal to vertically down without any hot gas ingestion into the engine intake. The design of these had been previously proven on the open-air plant and only minor modifications were found necessary to achieve the requirements.

4 EXHAUST GAS COLLECTION SYSTEM

There is a collector duct for each nozzle exhaust. Four are arranged in pairs to exhaust into separate Cullum detuners for each side. A simplified cross-section is shown in Figure 5.

Because the pressure loss of the collector duct varies depending on horizontal or vertical entry gas flow, the entrainment ratio and the nature of the airflow around the engine changes. It has been difficult, therefore, to evaluate true free-field performance in the vertical thrust mode in a test plant. No true free-field facility exists where this could be measured for comparison. It would be very difficult to do so without re-ingestion effects being highly probable. However, with an aircraft of known weight hovering at an altitude where re-ingestion and ground effects are not possible (at least 80 ft) and with intake and other aircraft effects, a relationship has been evaluated for correlation of test bed measured thrust with aircraft installed thrust.

Performance tests with an aircraft intake fitted in the test bed allow a close comparison to be made. The main difference can be accounted for by the stabilising bleed which is needed in hovering mode and which can be assessed reasonably accurately. Cross-calibration with open-air testing in the horizontal thrust mode has enabled test cell effects to be evaluated.

There is also a different thrust between horizontal and vertical nozzle positions and the engine running conditions vary slightly due to the different aerodynamics through the nozzles resulting in a change of exhaust pressure and angle or splay.

5 THRUST MEASUREMENT SYSTEM

The thrust measuring system is also unique to this type of test plant. As shown in Figure 6, the test cradle into which the engine is mounted from below, is suspended on three links from above; at the bottom of each is a thrust capsule. These are not all vertically in line with the front and rear nozzle centres when in the vertical position, front and rear nozzle thrusts are calculated from a moments' equation. One link is centrally mounted at the rear and one each side at the front. This permits the individual measurement of front and rear vertical thrust to allow calculation of the position of the centre of thrust, which, as previously illustrated, is critical to aircraft stability in hovering flight.

In order to restrain the cradle when in the vertical thrust condition, a front link is employed which allows the small amount of vertical freedom necessary with minimum hysteresis effects. The horizontal thrust measuring capsules are at the points of attachment of these two front links - one on each side.

The thrust capsules are calibrated in situ with the complete engine and plant interface connections installed. Links are installed to allow a hydraulic jack-up system to simulate the engine thrust loading on the cradle, each force being measured with master standard thrust capsules installed between the jack and the link.

The thrust capsules have been the Elliott load cell type which have a Wheatstone Bridge measurement system. We are currently changing to a Bofors shear-type load cell which has the same type of electrical measuring system but gives improved accuracy.

In order to simplify the measurements to the minimum necessary, where two thrust meters need to be summated, ie the horizontal and front vertical thrust measurements, the two capsules are connected in parallel with a common voltage supply, as illustrated in Figure 7. The outputs from the middle of the Wheatstone Bridges are also connected in parallel to the measuring system.

Because of the static vertical load of approximately 12,000 lb, the thrust measuring systems are set to zero, prior to applying loads, by an electrical adjustment.

No problems of calibration linearity have been experienced over the whole range despite the fact that the direction of loads on the measuring capsules change over at approximately 12,000 lb total vertical thrust.

6 PRODUCTION TESTING PLANTS

The Production and overhaul test plants are naturally very much simplified from the Development plants explained above - the principle differences being that testing is in the horizontal thrust mode only. The exhaust collection systems in use at Rolls-Royce are shown in Figure 8. Other types are in use at customer overhaul test bases.

As previously stated, the important thrust measurement is the vertical thrust and an evaluation of thrust centre. A cross-calibration between a Development plant in vertical thrust and the Production plant (horizontal) is carried out, which gives a simple thrust correction factor to be applied. Also, a calibration of the nozzle pressure ratios against vertical thrusts, front and rear, is obtained as shown in Figure 9. Thus, nozzle pressure ratios measured on the Production plant allow evaluation of thrust centre. The summation of thrusts calculated from pressure ratios allows a check to be made on

the total thrust measurement.

7 FUTURE DEVELOPMENTS

Future development of the Pegasus or similar vectored thrust engines may well have a reheat-type boost system in the fan exhaust to the front nozzles, commonly known as Plenum Chamber Burning (PCB). This will undoubtedly give new problems in the collector ducts, particularly in the vertical mode, necessitating water-cooling and mechanical improvements to withstand the greater thermal cyclic and shock loading, corrosion and erosion.

Some experience was gained with PCB into the collector ducts in the mid 1960's with a BS 100 engine, a much larger engine than the Pegasus, but the project was cancelled after 197 hours of testing, of which 10 hours was with PCB.

Within the next two years, we will be installing a full on-line data acquisition and processing system with computer test control possible.

The test plants can be modified to test an engine similar to the Pegasus but with a single vectoring rear nozzle, space exists in the plants for a collector duct and separate central detuner for the exhaust, as is evident from Figure 4.

Date JULY '80 Chart No. 1

SIGNIFICANT DATES IN THE DEVELOPMENT OF VECTORED THRUST

FIRST RUN OF PEG.1	SEPT. 1959
FIRST HOVER OF P1127	OCT. 1960
FIRST FLIGHT OF P1127	MAR. 1961
FIRST TRANSITION OF P1127	SEPT. 1961
TRIPARTITE KESTREL SQD. FORMED	MAY 1965
HARRIER ENTRY INTO SERVICE	APR. 1969

PEG1	THRUST	9000 Lbf
	THRUST-TO-WEIGHT RATIO	4·34
PEG 103	THRUST	21500 Lbf
	THRUST-TO-WEIGHT RATIO	6·94

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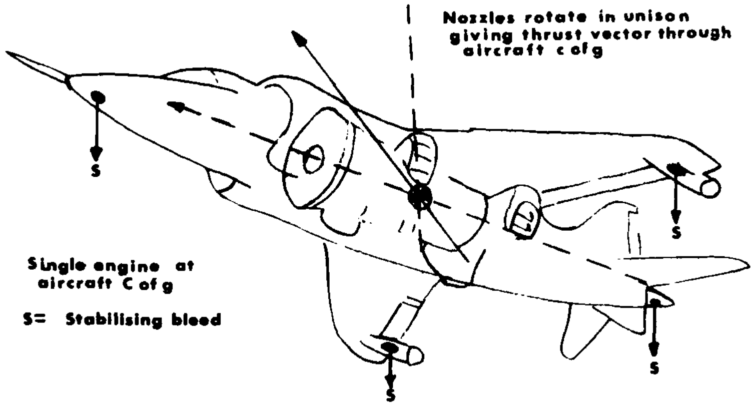
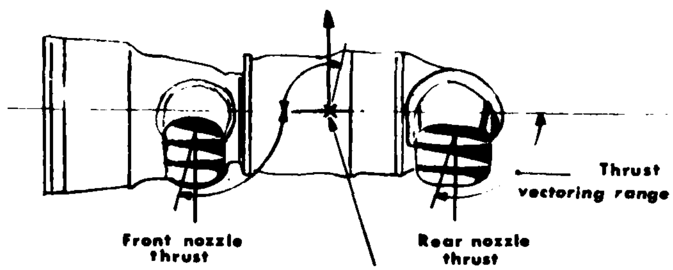
Date JULY '80

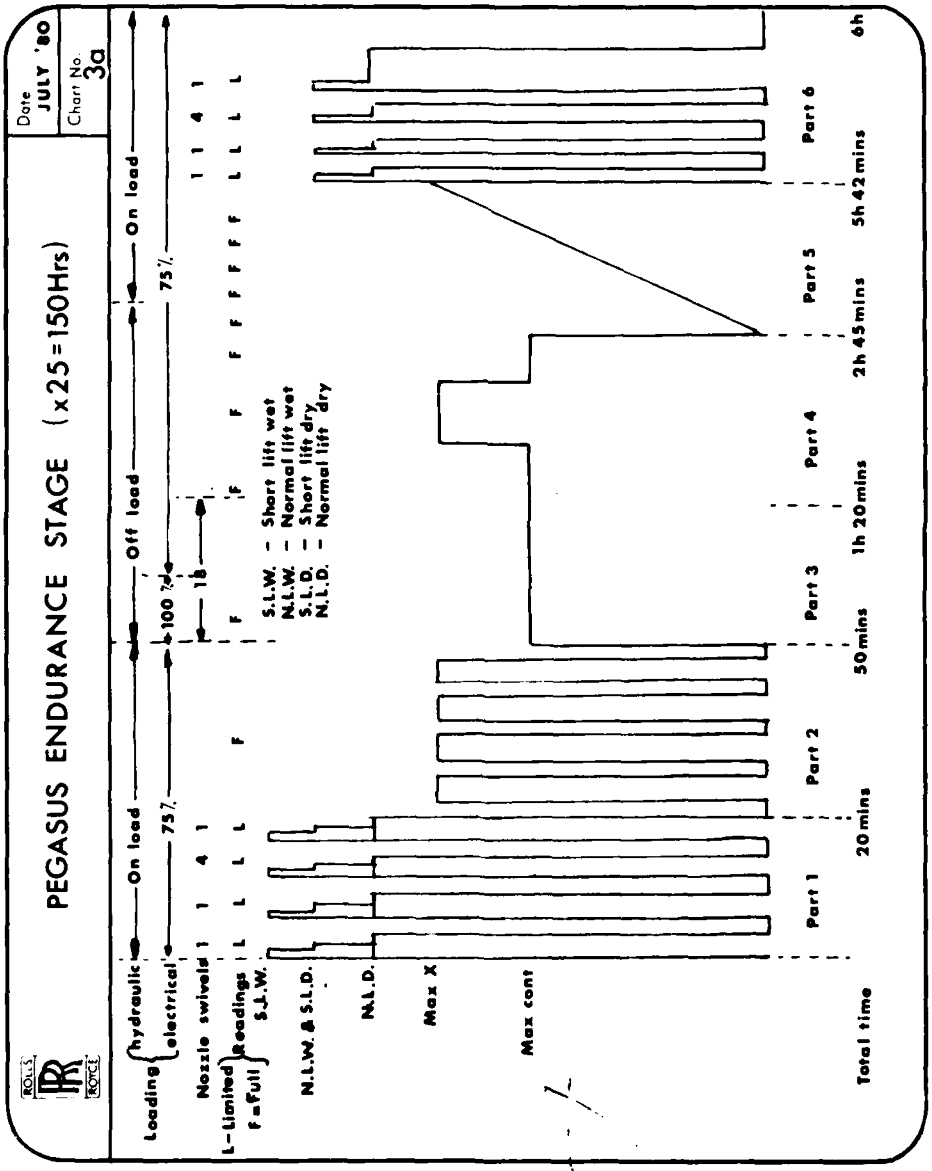
Chart No. 2

THE VECTORED-THRUST PRINCIPLE





THE VECTORED-THRUST POWERPLANT


PEGASUS ENDURANCE STAGE (x25=150Hrs)
Date JULY '80
Chart No. 3a
ROLLS ROYCE
Loading hydraulic
electrical
On load
Off load
On load
75%
100%
75%
Nozzle swivels
18
L-Limited F=Full Readings
S.L.W.
N.L.W. & S.L.D.
N.L.D.
Max X
Max cont
S.L.W. - Short lift wet
N.L.W. - Normal lift wet
S.L.D. - Short lift dry
N.L.D. - Normal lift dry
Part 1
Part 2
Part 3
Part 4
Part 5
Part 6
Total time
20 mins
50 mins
1h 20 mins
2h 45 mins
5h 42 mins
6h

Date JULY '80 Chart No. 3b

ENGINE RATINGS

Normal Lift Dry (N.L.D.)	:	T_j determined at rated S.O.T. with bleed
Short Lift Dry (S.L.D.)	:	Increased speed & thrust at T_j(NLD) +18°c & same bleed
Normal Lift Wet (N.L.W.)	:	Increased speed & thrust at T_j(NLD) +10°c with water injection & same bleed
Short Lift Wet (S.L.W.)	:	Increased speed & thrust at T_j(NLD) +23°c with water injection & same bleed

ROLLS ROYCE

Date JULY '80

Chart No 3C

ENDURANCE LIFT RUNNING SCHEDULE

Part 1 is made up of 4 cycles each of :-

a) Slam accel. from idle to S.L.W. with nozzle swivel from horizontal to vertical

b) 15 secs at S.L.W.

c) 75 secs at N.L.W.

d) 60 secs at N.L.D.

e) Slam decel. to idle with nozzle swivel to horizontal

f) 150 secs at idle

Part 6 is made up of:-

a) Slam accel. from idle to S.L.D. with nozzle swivel from horizontal to vertical

b) 15 secs at S.L.D.

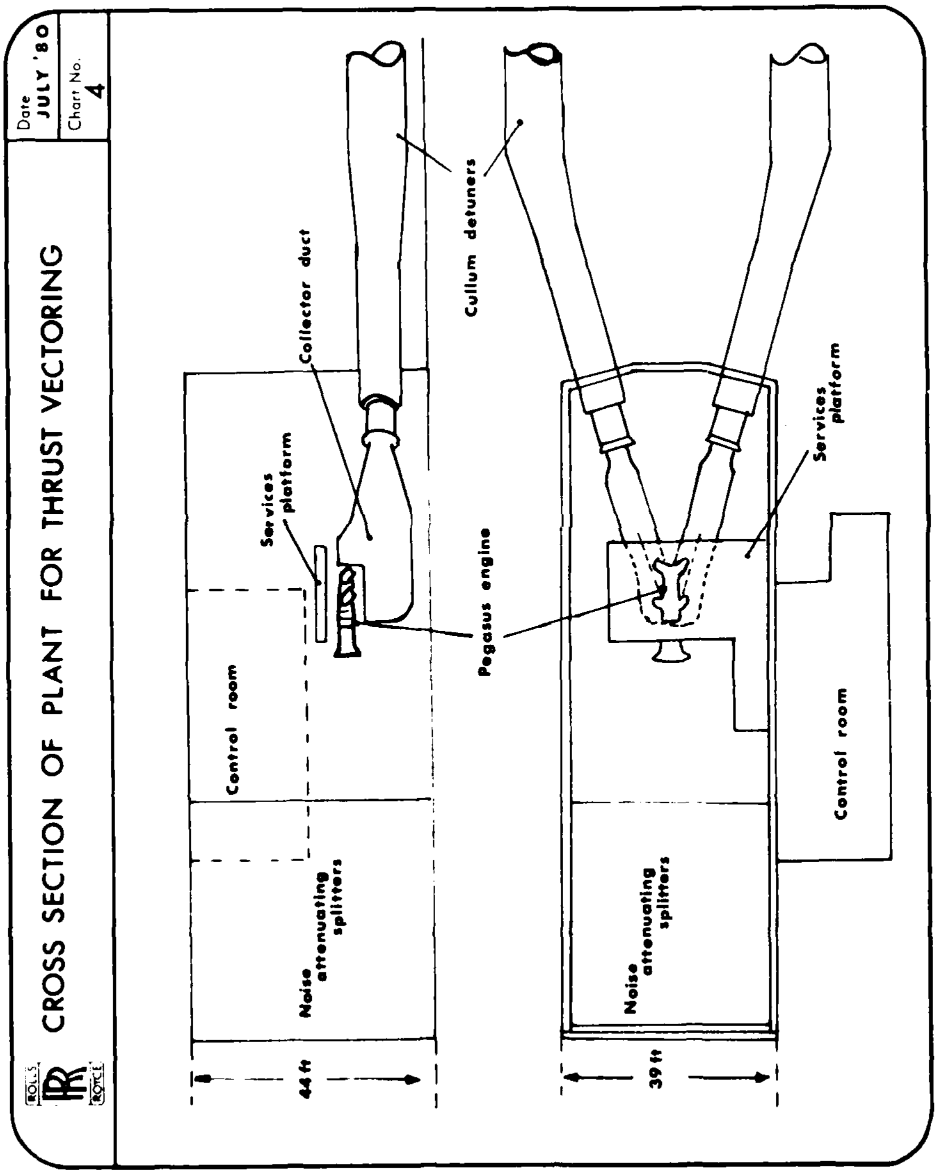
c) 40 secs (60, 100 & 200 secs on successive cycles) at N.L.D.

d) Slam decel. to idle with nozzle swivel to horizontal

e) 150 secs at idle

Note:- Continuous recordings of limited performance parameters are made for all running above idle

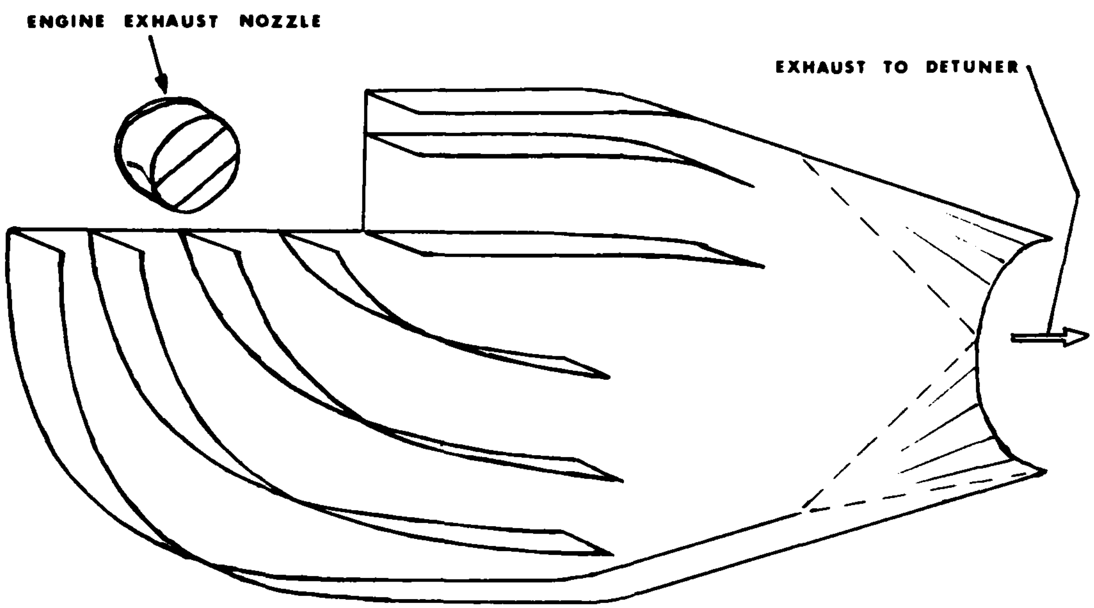
ROLLS ROYCE

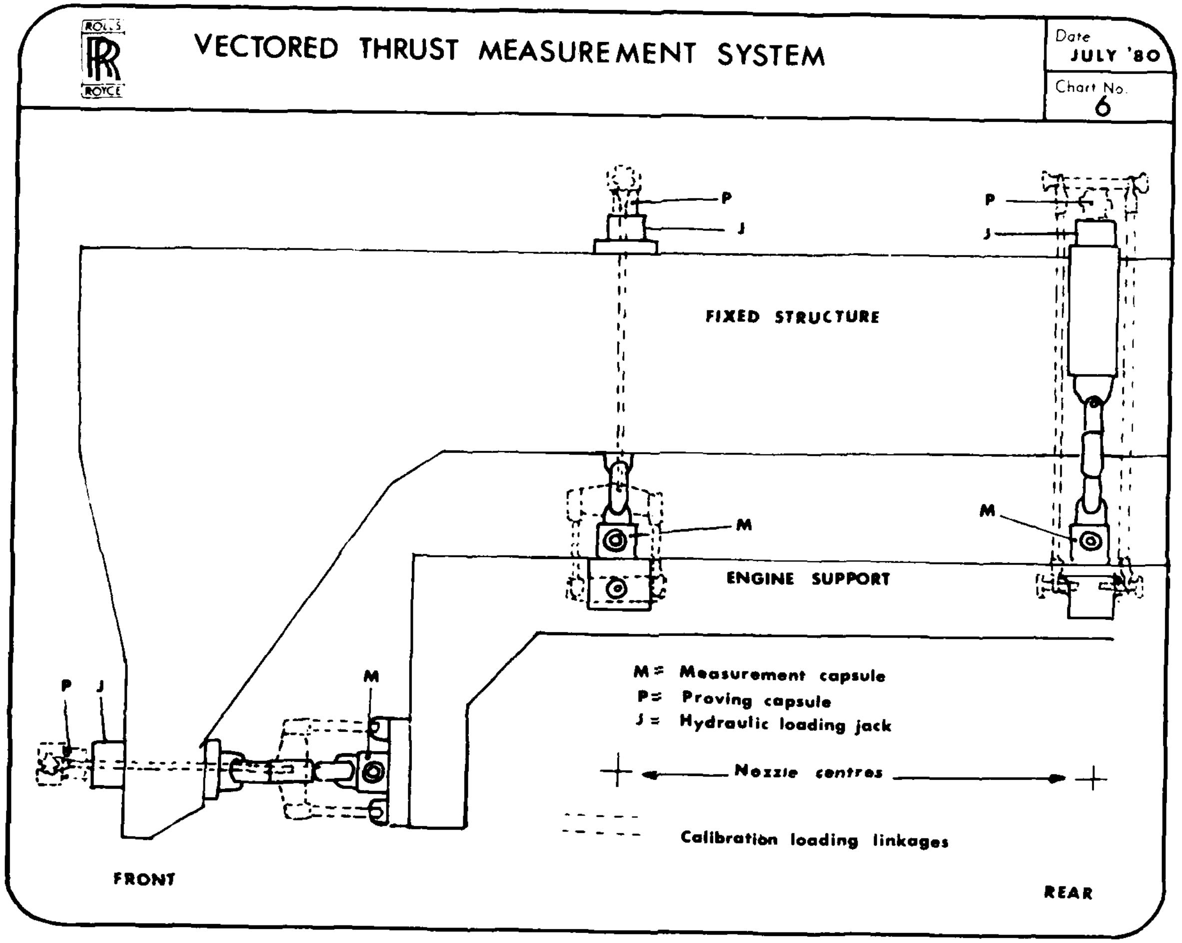

CROSS SECTION OF PLANT FOR THRUST VECTORING
Date JULY '80
Chart No. 4
ROLLS ROYCE
Collector duct
Cullum detuners
Services platform
Pegasus engine
Control room
Noise attenuating splitters
44 ft
Services platform
Control room
Noise attenuating splitters
39 ft

SECTION THROUGH A COLLECTOR DUCT

Date JULY '80

Chart No. 5





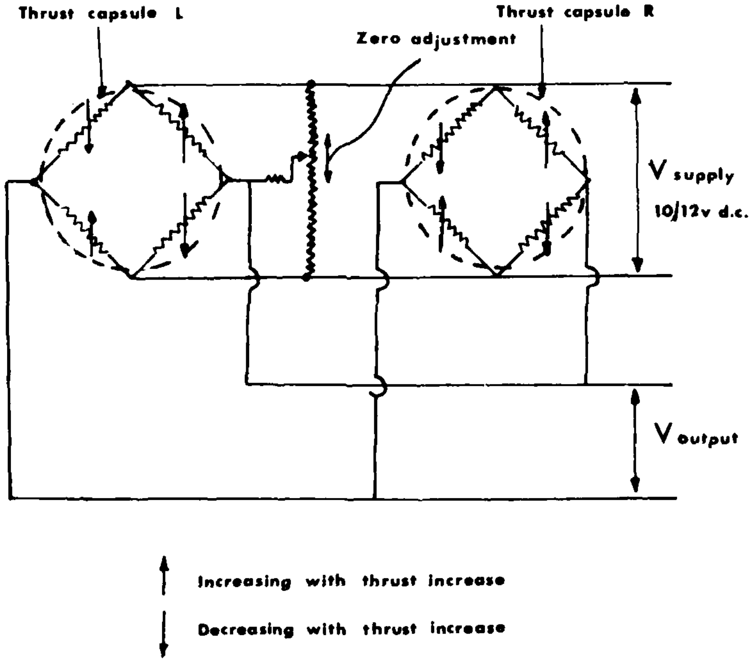
ROLLS ROYCE
VECTORED THRUST MEASUREMENT SYSTEM
Date JULY '80
Chart No. 6
P
J
P
J
FIXED STRUCTURE
M
M
ENGINE SUPPORT
P
J
M
M = Measurement capsule
P = Proving capsule
J = Hydraulic loading jack
Nozzle centres
Calibration loading linkages
FRONT
REAR

25-12

Date JULY '80

Chart No. 7

THRUSTMETER WIRING ARRANGEMENT FOR SUMMATING LOADS

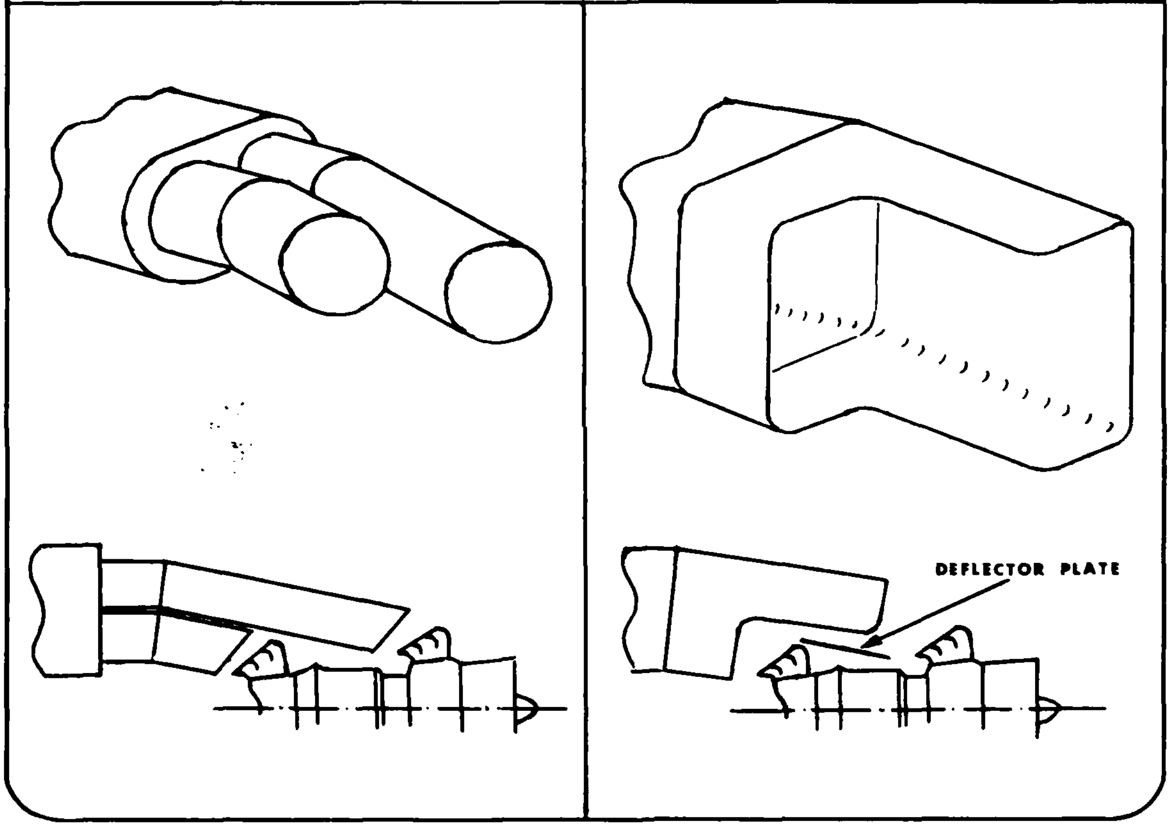


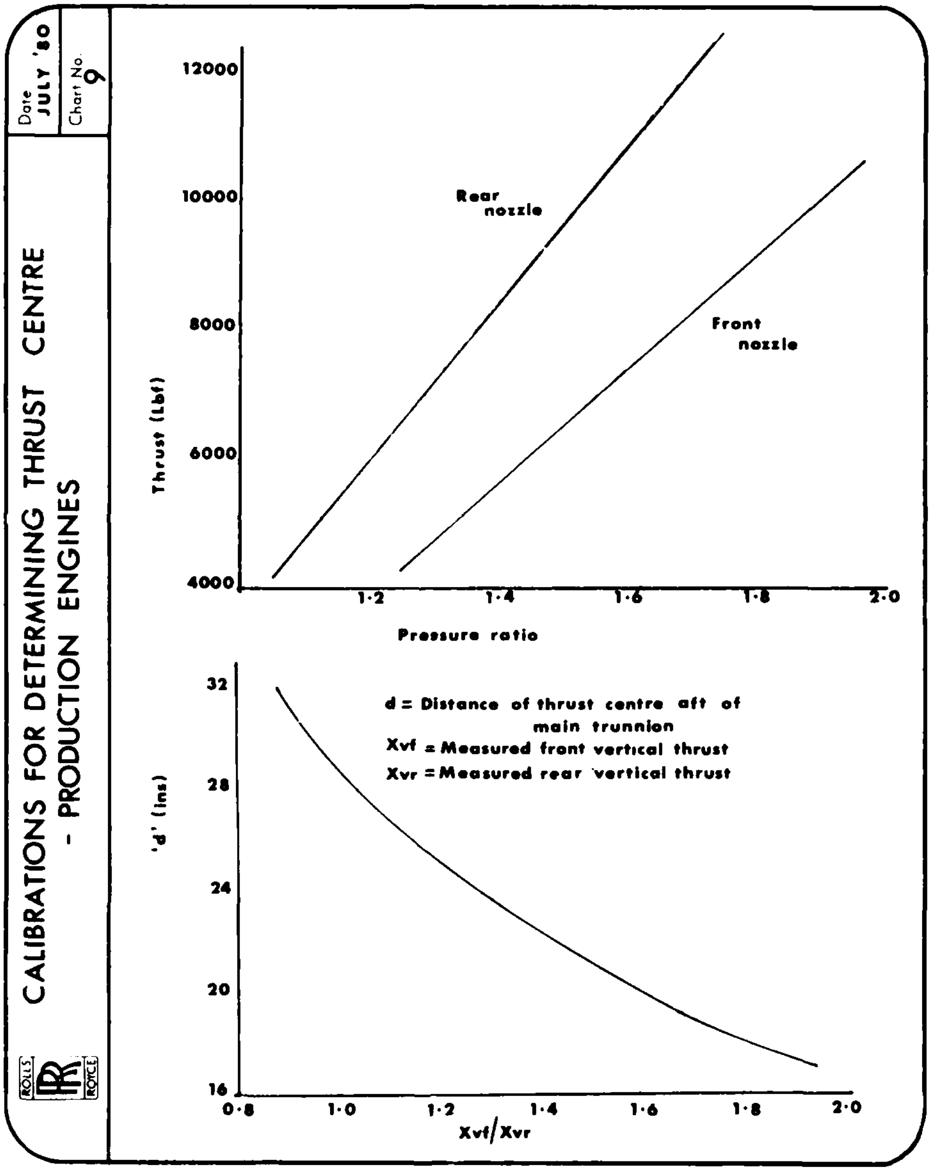
ROLLS ROYCE

ALTERNATIVE ARRANGEMENTS OF COLLECTOR DUCTS FOR PRODUCTION TESTING, HORIZONTAL ONLY.

Date JULY '80

Chart No. 8




CALIBRATIONS FOR DETERMINING THRUST CENTRE
- PRODUCTION ENGINES
Date JULY '80
Chart No. 9
ROLLS ROYCE
Thrust (Lbf)
12000
10000
8000
6000
4000
Rear nozzle
Front nozzle
1·2
1·4
1·6
1·8
2·0
Pressure ratio
d = Distance of thrust centre aft of main trunnion
Xvf = Measured front vertical thrust
Xvr = Measured rear vertical thrust
'd' (ins)
32
28
24
20
16
0·8
1·0
1·2
1·4
1·6
1·8
2·0
Xvf/Xvr

DISCUSSION

D.K.Hennecke, MTU, Ge

How much bleed air do you require for the stabilizing jets? Do you simulate it in your tests, and, if you do, doesn't it interfere with your thrust measurements?

Author's Reply

Bleed air is taken off during the endurance testing, cycling between various flows up to 12 lb/sec. with very short transients to 15 lb/sec., which exceeds 10% of the core engine flow. The ducting of this bleed air passes from the thrust cradle to the fixed plant through a flexible joint, a thrust spoiler can also be fitted to the exhaust. Thus, whenever testing with bleed is carried out, during endurance or performance testing, the thrust of the bleed is not included.

PERFORMANCE ASSESMENT OF AN ADVANCED REHEATED TURBO FAN ENGINE

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SUMMARY

During the test phase of an - in this case two engined - combat aircraft extensive evaluation work has to be done especially if this aircraft is powered by a newly developed engine. As far as engine performance is concerned various approaches to determine thrust-in-flight and other appropriate parameters are discussed. The degree of specialization or simplification of some methods points to the applicability, i.e. the quick estimation of actual take off thrust with regard to safety aspects in case of single engine climb out requires only a simple option, which produces acceptable results even if the engine is of lower instrumentation standard. On the other hand, inflight thrust for engine performance assessment and aircraft drag analysis is calculated by an ambitious computer programme using test data of engines with higher instrumentation standard. Eventually the influence of ATF testing (Altitude Test Facility testing) on the accuracy is described.

INTRODUCTION

During the development of a military turbofan engine from its test bed stage (development engines) via the early flight cleared phase of the final production engine standard two complementary test philosophies are followed up: the engine manufacturer is endeavoured to improve engine performance in terms of thrust and s.f.c. as well as engine and reheat handling by improving the component efficiencies and altering control laws (fuel supply, nozzle area control), since these engine qualities are subject to minimum requirements of the customer and user. The engine user on the other hand is interested to know how the engine behaves in both, handling and performance during flight rather than on test beds. As handling can be covered by Yes/No tests, performance evaluation requires extensive use of flight test data. Engine thrust for example is to be calculated as accurate as possible to fit up the individual lift-drag polars for given wing sweep positions, flap and slat positions and the great variety of external stores which the aircraft can be equipped with. This implies of course a steady improvement of the computer programmes as the amount of data increases, which in turn enables the user to assess the engine itself in detail. The following paragraphs deal with the problems and knowledges concerned in the engine performance assessment.

NOMENCLATURE

α	=	Angle of Attack
Aj, A7	=	Nozzle Exit Area
ATF	=	Altitude Test Facility
BL	=	Bleed
C_G	=	Gross Thrust Coefficient
EHV	=	Effective Heating Value
EPR	=	Engine Pressure Ratio
FAP-METHOD	=	Gross Thrust-Area-Pressure Method
FG	=	Gross Thrust
FM-METHOD	=	Gross Thrust-Mass-Flow-Temperature Method
FN	=	Net Thrust
FNIN	=	Installed Net Thrust
FPR	=	Fan Pressure Ratio
HPT	=	True Pressure Altitude

NOMENCLATURE cont'd

Ma	=	Mach Number
MFM	=	Main Fuel Flow
MFR	=	Reheat Fuel Mass Flow
MFT	=	Total Fuel Mass Flow
M1	=	Air Mass Flow at Engine Entry
M6	=	Gas Flow at Nozzle Entry
M7	=	Gas Flow at Nozzle Exit
$NH/\sqrt{\theta}$	=	Aerodynamic High Pressure Spool Speed
$NL/\sqrt{\theta}$	=	Aerodynamic Low Pressure Spool Speed
NPR	=	Nozzle Pressure Ratio
PTO	=	Free Stream Total Pressure
PSOSL	=	Standard Static Pressure at Sea Level
Pamb	=	Ambient Pressure
PT1	=	Total Pressure at Engine Entry
PT2LB	=	Total Pressure By-pass Entry
PS2LB	=	Static Pressure By-pass Entry
PT4LB	=	Total Pressure By-pass Exit
PS4LB	=	Static Pressure By-pass Exit
PT4	=	Total Pressure LP Turbine Exit
PS4	=	Static Pressure LP Turbine Exit
PMIX	=	Mixed Total Pressure at Reheat Entry
PS6	=	Static Jet Pipe Pressure upstream Nozzle Entry
$PT7 = P_j$	=	Total Pressure at Nozzle Exit
ΔP	=	$PMIX - P_j$
POT	=	Power Off Take
SLTB	=	Sea Level Test Bed
SFC	=	Specific Fuel Consumption
SOT	=	Turbine Stator Outlet Temperature
R	=	Gas Constant
TIF	=	Thrust-In-Flight
TT1=TTO	=	Free Stream Total Temperature
TSOSL	=	Standard Static Temperature at Sea Level
TT3	=	Turbine Stator Inlet Temperature
TT5	=	Mixed Jet Pipe (=R/H-) Entry Temeprature
TT7 = TT6	=	Total Temperature at Nozzle Exit
$TR_{cap.}$	=	Reheat Capability Temperature
$M1\sqrt{TT1}/PT1$	=	Corrected Air Flow at Engine Entry (= WAT)
$(M\sqrt{T}/P)_3$	=	Turbine Capacity

NOMENCLATURE cont'd

$(M\sqrt{T}/P)_5 = \phi 5$	=	Corrected Gas Flow at Jet Pipe Entry
$\eta_{REH.}$	=	Reheat Efficiency
$\eta_{R_{cap}}$	=	Maximum Reheat Efficience
Θ	=	TT1/TSOSL
δ	=	PT1/PSOSL
γ	=	Ratio of Specific Heats

1. THRUST EVALUATION OPTIONS

Depending on the task at hand e.g. engine performance, aircraft drag analysis, safety aspects during take off several, more or less sophisticated approaches to thrust determination can be pursued. The engine performance module presented in this paper consists of one main and some secondary options.

Engine performance and aircraft drag analysis require, for example, as main option the "thrust derived $-P_j$ method" in connection with the so-called nozzle calibration to obtain gross thrust and air mass flow, respectively. Since these two most important engine parameters are determined at the same engine station, the nozzle exit plane, it is labelled as a "linked method" so as to distinguish it from other methods. Any pressure distortion induced by the intake and any temperature profile disturbances are eliminated as much as possible. Therefore, measurements made farthest downstream of the engine will usually give better results than those carried out just behind the fan.

If on the other hand the gross thrust is obtained by nozzle parameters as mentioned above and the airflow is calculated from a fan map, the method is denoted as being "unlinked". The accuracy one can expect is lower because independent measurements are subject to individual tolerances which, in turn require individual mathematical approaches (see also para III). This happens in principle to some of the secondary options provided as back up methods.

These options have been selected out of a total of about 32 ways of computing basic parameters required to obtain the net thrust by combining gross thrust and airflow subroutines. The selection was done with respect to the highest accuracy achievable together with the utmost simplicity in the computational methods.

One of these secondary options fulfills the requirement of easy and quick estimation of actual installed thrust just prior to take off.

The following table shows the subroutines combinations of which lead to the 12 net thrust options.

Massflow	Gross Thrust	TT6 Calcul.
B1 fan-character. B2 by-pass calibr. + heat balance B3 by-pass calibr. + heat balance + turbine capacity B4 nozzle calibr. + heat balance	A1 FAP-method A2 FM-method A3 nozzle pressure methods	C1 engine heat balance requ. for those options which use A2 or A3 in combination with B3 or B4.

For the detailed flow charts see fig. 2 to 6. The air mass flow subroutines B1, B2, B3 and B4 are combined with the gross thrust subroutines A1, A2 and A3, thus giving 12 options to calculate gross and net thrust by use of flight data. A distinction will be made between linked and unlinked methods. For engine station identifiers see Fig. 1.

Option 1 (see Fig. 2) is an unlinked method. Airflow is obtained by use of a certain form of the fan characteristic (B1).

Since this kind of fan map is a function of the engine pressure ratio EPR = PMIX/PT1 or PS6/PT1 the intake pressure recovery map has to be included in the calculation loop

(airflow M1 is a function of PT1 which in turn depends on M1 via the intake recovery characteristic). Then gross and net thrust can easily be determined by use of the FAP graph A1. For the definition of PMIX see Fig. 9

That graph A1 is generated from sea level test bed data. The expansion to higher pressure ratios $PMIX/P_{amb}$ or $PS6/P_{amb}$ can be done by AFT calibration. This graph is valid for the uninstalled as well as for the installed engine, i.e. power of takes, bleed air extractions and aircraft intake effects result in an appropriate pressure ratio change, thus always yielding the correct gross thrust. For the reheated engine, lines of constant R/H-nozzle area A_j can be plotted and used same as in the dry case.

The intake recovery characteristics η_R = f (WAT, α , Ma) usually represent model data which may be confirmed or improved through flight test. A 46-probe rake is sufficient for this task but the number of test conditions at which stabilized data must be gathered is rather high.

The fan characteristic B1 is the result of SLTB- and ATF data. This special form was selected to provide a unique line for any operating condition. Experience shows that the scatter encountered is just about acceptable for this characteristic to be used as a back up subroutine.

Option 2

(Fig. 3, unlinked method) Subroutine B2 does not require intake recovery curves because the airflow M1 is obtained in a straight forward manner from measured engine parameters, a by-pass calibration curve and the overall engine heat balance. Gross and net thrust is calculated as before with option 1. The advantage of the by-pass calibration according to subroutine B2 or B3 is, that the by-pass ratio can easily be determined.

Option 3

(Fig. 4, unlinked method) requires the turbine capacity $(M\sqrt{T}/P)_3$ because the heat balance covers only the combustion chamber. The turbine stator inlet temperature TT3 is found by an iteration process which in turn leads to the unknown engine core flow. Total airflow M1 is obtained by adding the core flow and by-pass flow, which has been determined from the calibration curve same as in option 2. The advantage of this subroutine B3 relative to the simple B2 subroutine is the possibility one has to calculate the HP-turbine rotor entry temperature SOT from TT3. The turbine capacity $M3\sqrt{TT3}/PT3$and the individual compressor bleed factors have to be supplied by the engine manufacturer. Especially the turbine capacity should be very accurate because a variation of 1 % entails an unacceptable stator inlet temperature deviation.

Option 4

(Combination B4-C1-A1, linked method) This combination of nozzle calibration (subroutine B4) and heat balance provides a preferable method by which the airflow is determined. The calibration curve "flow function v.s. pressure ratio" is derived from SLTB data and extended by use of ATF test results. The unknown airflow M1 and the mean jet pipe or nozzle entry temperature for either dry or reheated operation is calculated by an iterative process on the engine heat balance (Subroutine C1). Gross and net thrust are obtained as described above.

Option 5, 6, 7 (unlinked) and 8 (linked)

(Fig. 5, option 5 presented as example.) The major difference between options 1/5, 2/6, 3/7 and 4/8 is the use of a second type of gross thrust function $FG/(M6\cdot\sqrt{TT6})$ vs (pressure ratio) instead of $FG/(Aj\cdot Pamb)$. This function A2 is as reliable as A1 in terms of data scatter. Since this characteristic implies the unknown nozzle entry temperature TT6, the subroutine C1 has to be incorporated into options 5, 6 and 7. The heat balance C1 is already required, as described for option 4, for the flow function of the nozzle calibration B4 and, therefore immediately available for the gross thrust subroutine A2. For the definition of the engine heatbalance see Fig. 9.

Option 9, 10, 11 (unlinked) and 12 (linked)

(Fig. 6, option 12 presented as example.) Calculation of air mass flow M1 is performed as in options 1/5, 2/6, 3/7 and 4/8, however, the gross thrust subroutine A3 provides a more ambitious approach to thrust via the nozzle pressure ratio NPR. Engine parameters to be measured are fuel flow, jet pipe pressure, nozzle area and bleed air flow. The iteration on the heat balance jet pipe pressure, nozzle area and bleed air flow. The iteration on the heat balance leads to the nozzle temperature required to determine the NPR. Since the measured nozzle area is a geometric rather than an aerodynamic area and the total pressure PT7 = PT6 does not include any losses, the calculated ideal gross thrust has to be corrected with the so-called thrust coefficient C_G. This coefficient is function of NPR and therefore a typical result of ATF testing. During the early test phases an altitude effect became apparent expressing itself through distinct C_G-lines.

It can be shown that there is no altitude effect, if a temperature correction is applied to the measured nozzle area, which takes thermal expansion of the nozzle petals into account. The C_G -data then collapse within the usual scatter.

These secondary options have been used as well for back up investigations as for special purposes. Data and results are presented in para II.

The flow chart for the "main option" is shown in Fig. 7. After having stored the data obtained during flight the loop starts with an initial TT5-input.

Nozzle exit pressure is calculated from measured area weighted PMIX by use of the ΔP-calibration curve which is based on ATF data. With the static thermodynamic parameters in the nozzle exit plane and discharge coefficients obtained from model tests the aerodynamic nozzle area A_7 is determined and compared with the measured nozzle area A_j. This comparison will usually show that both areas differ after this first step. Therefore, the whole calculation is restarted after having corrected the calculated nozzle area A_7 by a ΔA_j term. This ΔA_j value is obtained from another calibration curve $\Delta A_j/A_j = f\ (\phi\ 5)$.

If $A7-\Delta A_j$ and A_j are identical after the n^{th} loop, gross and net thrust and various engine parameters can be determined. If the engine was in reheated operation, the calculation of the static parameters in the nozzle exit plane including the reheat nozzle area requires the determination of the reheat temperature and the fundamental pressure loss due to heating. In this case the third calibration curve for the reheat efficiency η_{Reheat} = f (R/H-temperature TT7) is used. Again this characteristic is the result of SLTB and ATF testing. The required max. capability curves for TT7 and reheat efficiency have been evaluated from component test beds. The subsequent calculational steps are then the same as above.

If the instrumentation standard of the test engine does not include the rakes to measure PT4 and PT4LB which give the area weighted PMIX, a calibration curve EPR vs. (MFM or $NL/\sqrt{\theta}$) would aid in establishing PMIX. These curves are the result of a great number of ATF- and flight tests with appropriately instrumented engines of the same performance standard. Due to the data scatter the accuracy in thrust is about 1 to 2 % worse than that achievable by a "heavily" instrumented engine.

The subsequent calculation is basically identical to the above described pattern, but due to the fact that PT1 is reuqired so as to determine PMIX from EPR, it is necessary to incorporate PT1 into the calculation loop because it depends on airflow (intake recovery!) which improves after each loop until $A_7-\Delta A_j = A_j$, as already mentioned.

With the subcritical spillage-, intake bleed- and afterbody interference drag characteristics applied to the net thrust the so-called "installed net thrust" FNIN can be estimated. Due to the bookkeeping method used only thrust dependent drag components have to be used to distinguish between engine thrust and aircraft drag.

Within the scope of engine performance it is customary to establish a performance map thrust vs. Machnumber with altitude, air mass flow, engine pressure ratio and s.f.c. as parameters, for example. Therefore, it is mandatory to correct the engine parameters as calculated from test data to round numbered altitudes and to a certain temperature level (e.g. ISA or ISA + 15). In the following these are named reference or standard conditions. This correction is done by a test engine performance correction programm (Fig. 8). Basically the correction is made by use of a "ratio or slope method": after having calculated the engine parameters, e.g. compressor pressure ratios, engine pressure ratio, airflow, turbine entry temperature in the above-described option 13, these parameters will then be determiend anew using appropriate gas generator characteristics for test conditions as well as for reference conditions. The corrected data then are obtained by the following equation:

$$\text{corrected data} = \text{test data} \times \frac{\text{gas generator data, refer.cond.}}{\text{gas generator data, test cond.}}$$

Thus the final results as gross thrust, net thrust, airflow, s.f.c corrected to standard or any other reference condition are then calculated in exactly the same manner as they have been computed in the main option for test conditions.

2. GENERAL PRESENTATION OF RESULTS

In the following, some graphs will be shown and briefly discussed as far as calibration curves are concerned as used in the thrust options. In addition, also some flight test results pertaining to engine performance are presented.

A convenient generalized presentation of these results, with only a small scatter, are plots of gross thrust functions $FG/(P_{amb} \cdot A_j)$ vs. P/P_{amb} (= FAP, subroutine A1) or $FG/(M6 \cdot \sqrt{TT6})$ vs. P/P_{amb} (= FM, subroutine A2). It should be pointed out, that gross

thrust functions calculated with PMIX be plotted against the independent pressure ratio $PS6/P_{amb}$ and if calculated with PS6, be plotted against $PMIX/P_{amb}$.

Fig. 10 shows the FAP calibration curve established using ATF data, i.e. using without exception, only measured engine and ambient parameters. The mean curve through these data appears again on Fig. 11. The data points along this line are the results of 10 test flights. In this case the gross thrust was, of course calculated utilizing the main propulsion programme according to Fig. 7. Both characteristics show small scatter, which means that this is a reliable method.

Since all the FAP-data, i.e. uninstalled pass-off data from SL Test Bed, installed acceptance test data from the aircraft with or without power and/or bleed extraction and intake influences, as well as ATF test data collapse, for a certain A_j, onto a unique line, it is obvious that this simple method should be used for gross thrust calculations just prior to take off. Especially if overload take offs at high ambient temperatures have to be performed, the measurement of just PMIX or PS 6, Pamb and A_j allows the on-line determination of gross thrust and thus the decision whether a single engine climb out is feasible should the higher thrust engine fail at the critical point during the take off run.

Similar results can be obtained if the data are plotted as in Fig. 12 and 13. The former again represents ATF data with PS 6, PMIX, M6 and FG being measured values. The mixed temperature TT6 is calculated by use of the engine heat balance. Fig. 13 contains the mean PMIX/Pamb-curve of Fig. 12 and calculated gross thrust data which are based on in-flight measurements. Again, in both figures the scatter is small. A deviation of in-flight data from the ATF calibration curve is not discernible. In Fig. 14 the fan characteristic according to subroutine B1 is plotted with ATF data. The results of the calculations using in-flight data are satisfactory in comparison with the nozzle calibration method.

The by-pass calibration curve as required in subroutines B2 and B3 is presented in Fig. 15. The results have been obtained by use of ATF data with the B2-subroutine itself running in a reverse mode. Since the data scatter in this case is about $\pm$ 5 % the final results obtainable with option 2, 6 and 10 are just about acceptable. A quite similar picture can be produced with the B3-subroutine which contains a different heat balance. The scatter is slightly worse. Therefore, options 3, 7 and 11 as well as 2, 6 and 10 are used only as back up methods or for turbine entry temperature and by-pass ratio determination, respectively, Ref. para I.

Good results can be obtained with the nozzle calibration curve of subroutine B4. If the data points are interpreted as having scatter (this happened with some engines) the uncertainty is $\pm$ 1.5 %. However, in this case (Fig. 16) an altitude effect can be observed rather than a scatter. If this fact is taken into consideration the uncertainty is reduced to $\pm$ 0.4 % for the worst data points.

Finally the most important calibration curves required for the thrust derived P_j method are discussed. The jet pipe pressure loss for dry engine operation $\Delta P = PMIX - P_j$ has to be known in order to calculate gross thrust and mass flow from $P_j = PT7$ via the flight measured PMIX. The analysis of ATF data showed no significant difference between the area and momentum weighted PMIX, but the area weighting method does not require the determination of the by-pass ratio. It is therefore reasonable to use the simpler and consistent area weighting method. This P_j can be computed from ATF data by use of the following ideal thrust equation for convergent nozzles:

$$(1) \quad FG = A_j \cdot P_{amb} \cdot \frac{2\gamma}{\gamma - 1} \cdot \left[\left(\frac{P_j}{P_{amb}} \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right] \quad \text{(subcritical nozzle condition)}$$

$$(2) \quad FG = A_j \cdot P_{amb} \cdot \left\{ \left[2 \left(\frac{2}{\gamma + 1} \right)^{\frac{1}{\gamma - 1}} \right] \cdot \frac{P_j}{P_{amb}} - 1 \right\} \quad \text{(supercritical)}$$

Since all parameters except P_j have been measured, the latter can be determined and thus ΔP too (Fig. 17). The ratio of true specific heats, γ, is calculated within the iteration process, which implies the determination of the required mean gas temperature at the nozzle exit TT7 by heat balance.

This, in turn, allows the calculation of $\Delta A_j = A_{j\,measured} - A_{j\,calcul.}$ (Fig. 18) from the isentropic expression for non-dimensional mass flow at the nozzle exit:

$$(3) \quad \left(\frac{M \cdot \sqrt{T}}{P_j \cdot A} \right)_7 = \left\{ \frac{2 \cdot \gamma}{R \cdot (\gamma - 1)} \cdot \left[\left(\frac{P_j}{P_{amb}} \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right] \right\}^{1/2} \cdot \left(\frac{P_j}{P_{amb}} \right)^{-\frac{\gamma + 1}{2\gamma}}$$

A7 can now be determined and compared with the measured A_j. The nozzle gas flow M7 is obtained by adding the fuel flow to and subtracting any bleed mass flows from the engine

inlet airflow M1 measured with the aid of a venturi facility in the ATF. Both, $\Delta P/PMIX$ and $\Delta A/A_j$, are plotted vs. the flow function at station "5", i.e. at reheat entry, as $M5 \cdot \sqrt{TT5}/PMIX$.

For the reheated engine the reheat efficiency calibration curve (Fig 19) is required from ATF data by the reverse use of option 13 with all necessary ATF engine parameters being input parameters. The main difference to the dry engine operation as far as the pressure loss between stations "MIX = 5" and "J = 7" is concerned is the fundamental pressure loss due to heating. While calculating $P_{jREHEAT}$, the dry ΔP-characteristic is assumed to be unchanged, and therefore, used as in the non-reheated case. The capability terms in Fig. 19 are functions of the by-pass exit conditions and thus established through component testing.

Once series of test flights with a sufficient number of suitable aircraft manoeuvres have been completed, preferably steady levels, but also level accelerations and climbs with constant Machnumber, at maximum dry (MAXD) or maximum reheat (MAXR) power, engine flight performance maps can be established. An example for such a performance map may be a presentation of thrust vs. machnumber and altitude with lines of constant airflow, s.f.c. and others superposed. This is possible for the reheated and non-reheated engine at the appropriate maximum power. For comparison each may be plotted using either predicted or test data. The test data should be corrected for round numbered altitudes and ISA conditions.

Other ways of presenting engine performance can easily be imagined, of course, for example FNIN vs (MFM, Ma, HPT) or SFC vs (FNIN, NH, Ma, HPT).

Some remarks on the performance of convergent nozzles will conclude this section. In general, nozzle performance is expressed in terms of nozzle coefficients, which take account of actual flow effects such as (1):

- Three dimensional nature of the flow in the nozzle
- non-uniformity of pressure and temperature profiles in the planes of measurement
- coverage of pressure and temperature probes, which will not necessarily give representative mean values
- local flow direction deviations
- dissociation at high temperatures
- pressure losses due to friction and due to facilities as flame holders between plane of measurement and nozzle

The following nozzle coefficients, as defined at the nozzle throat, are in common use:

- discharge coeff. $C_D = \dfrac{M_{act} \cdot \sqrt{T_t}}{A_{act} \cdot P_t} \Big/ \dfrac{M_{id} \cdot \sqrt{T_t}}{A_{act} \cdot P_t} = \dfrac{M_{act}}{M_{id}}$

or $$C_D = \frac{M_{act} \cdot \sqrt{T_t}}{A_{act} \cdot P_t} \Big/ \frac{M_{act} \cdot \sqrt{T_t}}{A_{id} \cdot P_t} = \frac{A_{ideal}}{A_{actual}}$$

both for the same NPR. The resulting numerical values of C_D are idential,

- thrust coeff. $C_X = \dfrac{F_{G,act}}{M_{act} \cdot \sqrt{T_t}} \Big/ \dfrac{F_{G\ id}}{M_{id} \cdot \sqrt{T_t}}$ and
- thrust coeff. $C_G = \dfrac{F_{G,act}}{A_{act} \cdot P_{amb}} \Big/ \dfrac{F_{G\ id}}{A_{act} \cdot P_{amb}}$

It can easily be shown that $C_G = C_D \cdot C_X$ (for the convergent nozzle, only).

These coefficients - as function of NPR - represent typical characteristics to be determined preferably by ATF testing. The results can - as all the other calibrations - be used for thrust-in-flight purposes, keeping in mind that it is important that a completely consistent approach between calibration and application is maintained.

As at the beginning of a test programme a sufficient number of data such as full scale nozzle coefficients are usually not available, model nozzle characteristics obtained using cold air instead of hot gases are supplied by the engine manufacturer. Therefore, early calculations with test data are restricted by these shortcomings.

However, ATF testing allows for comparison between full scale and model data. Figs. 20 to 22 show ATF data related to the appropriate model data. For example C_D/C_{DMODEL} vs NPR appears to be equal to 1 for NPR $>$ NPR_{CRIT} within a scatter of $\pm$ 0.5 % (Fig. 20).

It deviates up to - 2 % for lower NPR. In general, it can be stated that the ratio of 1 is maintained and the scatter decreases to about ± 0.2 % for higher NPR's.

The C_X/C_{XMODEL}-characteristic tends to values between 0.98 and 0.99 for NPR > 1.9 (see Fig. 21). A similar picture is shown for the gross thrust coefficient ratio C_G/C_{GMODEL} vs NPR in Fig. 22.

Even if these facts should be ignored, the effect on the accuracy of the thrust-in-flight calculations would be of lower order as long as consistency - either only model data or only full scale data are used - is maintained.

3. ATF TESTING REQUIREMENTS AND ACCURACY ACHIEVABLE

Aircraft performance evaluation phases usually require engines to be either pre-flight or post-flight ATF calibrated. The calibration runs should be performed at those conditions at which the test flights will be carried out. These will normally cover great parts of the flight envelope with the guarantee points being included as a minimum. In the case of a pre-flight calibration the ATF test points will be at ISA, in the other case the mean ISA conditions, at which most of the flights have been performed, should preferably be applied. At any test condition the engine has to be stabilized within a certain R.P.M. range and - if applicable - within a certain reheat power range. The latter will chiefly cover the supersonic flight regime. Both the R.P.M. and R/H ranges are required to obtain calibration lines rather than calibration points.

The most important requirement for any calibration is that the engine remain "sealed" under any circumstances, i.e. in terms of instrumentation, fuel flow meters and components, after removal from either the ATF or the aircraft, into whichever facility it was first installed. Otherwise the consistency, as discussed in para II. is not longer maintained.

Several investigations have been made concerning accuracies (1), (3), (4) in terms of mathematical handling of linked and unlinked methods, influence coefficient estimation, error limit classification and effects of whether one or more engines were SLTB-and/or ATF-calibrated for a given performance phase.

The advantage of the linked methodology is the smaller uncertainty achievable due to the beneficial effects of non-independent errors (the mass flow is used to calculate inlet momentum as well as gross thrust). Thus errors may cancel out in the net thrust. Erros in the unlinked methodology are assumed to be independent. This leads to a root-sum-square combination. An Example may clear the situation: for FG/FN = 2, 1 % error in FG, 1 % error in inlet momentum FO

- the unlinked method results in:

$$\frac{EL(FN)}{FN} = \sqrt{2^2 x (1\ \%)^2 + (-1)^2 x (1\ \%)^2} = 2{,}24\ \%\ \text{error in FN}$$

- the linked method results in:

$$\frac{EL(FN)}{FN} = 2x1\ \% - 1x1\ \% = 1\ \%$$

error in FN, because FG and FO are linked by airflow.

The influence coefficient IC can be defined as the percentage change in the result (net thrust for example) caused by 1 % change in a input measurement (e.g. pressure, temperature or fuel flow). This can easily be done by the use of the appropriate thrust calculation programme, e.g. the main option. The results of such an investigation for two given flight conditions are presented in Fig. 23 and Fig. 24 as an example.

For a Gaussian distribution the "2-σ-Uncertainty" or "2 σ-Error Limits" of parameter x (=EL(x)) is defined under the restriction that the various parameters be independent of each other and be within the same error class.

The following equation expresses the 2-σ -uncertainty of a final result y (e.g. net thrust) in terms of error limits and influence coefficients for one or more input measurements x_i:

$$\frac{\%\ EL(y)}{y} = \sqrt{\sum_i \left\{ \left[IC(y:x_i) \right] \cdot \left[\%\ \frac{EL(x_i)}{x_i} \right] \right\}^2}$$

$$IC(y:x_i) = \frac{\partial y}{\partial x_i} \cdot \frac{x_i}{y}$$

If 2-σ_1 is the uncertainty of one engine and 2-σ_2 that of the second engine to be installed in the aircraft, the uncertainty of the installed thrust is determined by

$$2\text{-}\sigma\,(FNIN) = \sqrt{\left(\frac{2\text{-}\sigma_1}{2}\right)^2 + \left(\frac{2\text{-}\sigma_2}{2}\right)^2}$$

The 2-σ-values in the following table have been calculated on the supposition that the uncertainty of the SLTB-calibration of ΔP, ΔAj and η_{REH} is twice that of the ATF calibration and the instrumentation is unchanged, as stated above.

Altitude	5000 ft	5000 ft	20000 ft	20000 ft
Mach No.	0.45	0.7	0.9	1.4
Power	dry	dry	dry	Combat
1 ATF cal.	3,9 %	3,5 %	3,0 %	5,8 %
2 ATF cal.	2,8 %	2,5 %	2,1 %	4,1 %
1 SL cal.	6,5 %	6,2 %	5,8 %	8,0 %
2 SL cal.	4,6 %	4,4 %	4,1 %	5,7 %
1 SL + 1 ATF	3,8 %	3,6 %	3,3 %	4,9 %

4. CONCLUDING REMARKS

The results achieved during in-flight tests, ground tests with the aircraft, ATF and SLBT tests as shown in the aforementioned Figures lead to the conclusion that test techniques and evaluation methods are available, which give immediate answers to the question whether the measured or calculated gross thrust or the instrumentation is wrong. Furthermore, ambitious calculation methods in connection with sufficient calibrations allow good results in terms of non-dimensional as well as absolute values as they are required for on overall assessment. Higher accuracy is achieved if linked methods are used. Best accuracy can be obtained if at least those 2 engines of any performance standard are ATF calibrated, performance and aircraft drag shall be analysed with. This should imply the successive up-date of those parts of the software which are subject to alteration due to progress of testing.

In any case, it is most important that consistency be maintained between calibration (ATF) and application (flight test analysis) if the above mentioned evaluation methods are used, because consistency may be impacted by negligences in the software as well as in the hardware handling.

REFERENCES

(1) MIDAD Study Group
Guide to procedures for in-flight thrust measurement AGARDograph No. 237 (NGTE Note NT. 995)

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Evaluation of A/C 06 Flights 95 to 102.
MBB report FTR/0089, Nato Confidential

(3) H. Dirscherl
Accuracy Study and Selected Problems of the Flight and ATF Evaluation of Engine 0038/5
MBB report FTA/0183, Nato restricted

(4) H. Dirscherl
Accuracy Study of Thrust-in-Flight P_j and 12-Option-Program
MBB report FTA/0120, Nato restricted

FIG. 1

ENGINE STATION IDENTIFIERS

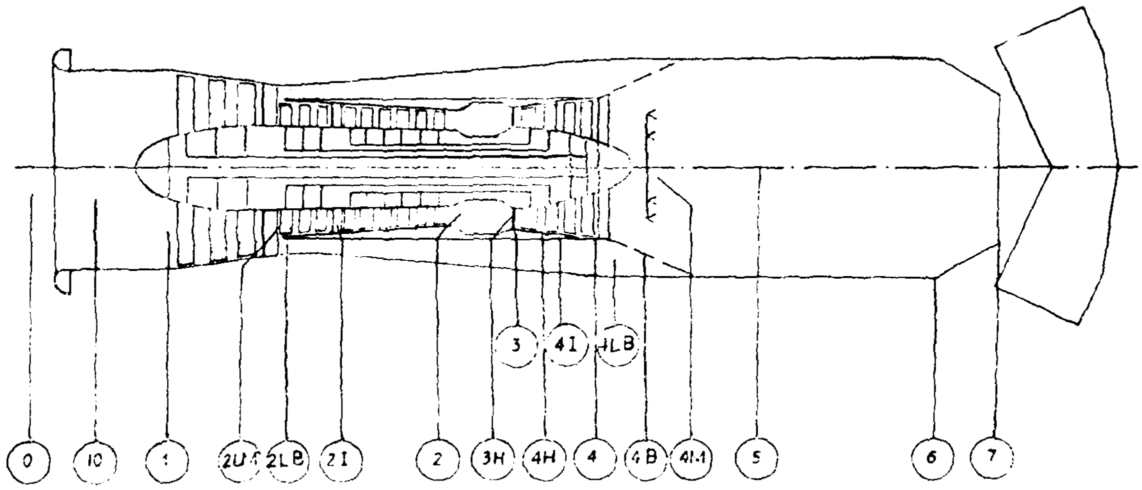
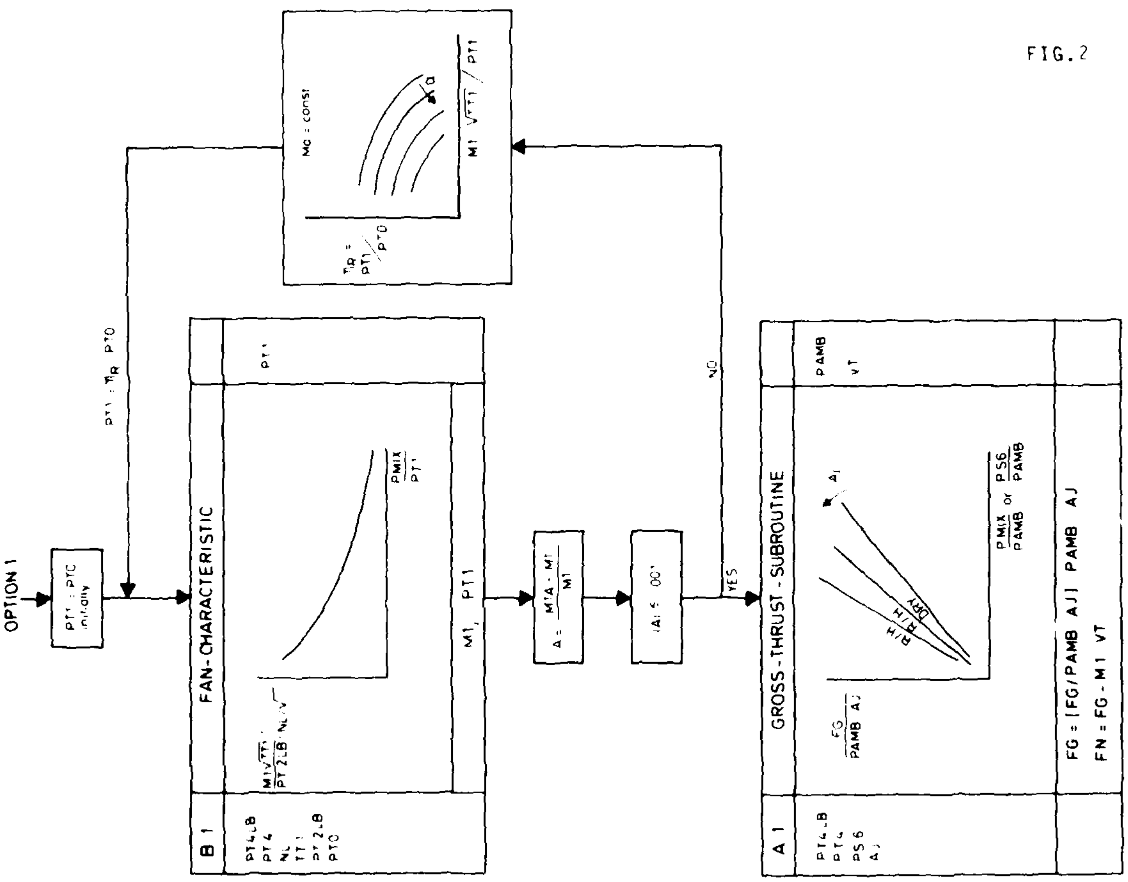


FIG. 2



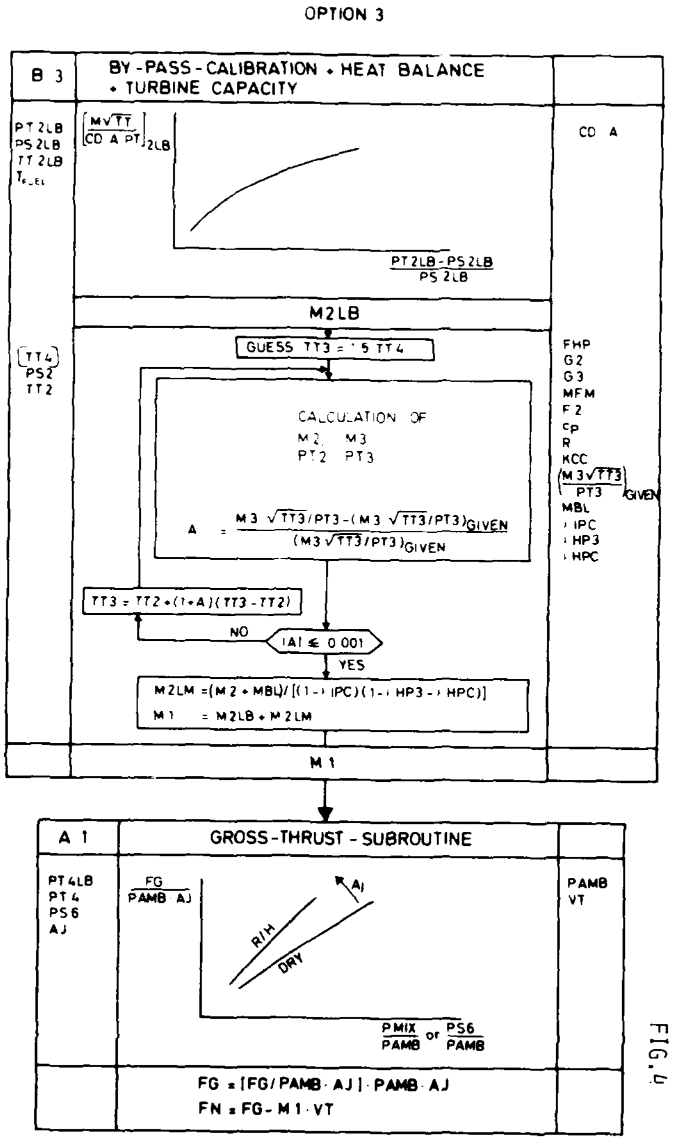

OPTION 3
B 3
BY-PASS-CALIBRATION + HEAT BALANCE + TURBINE CAPACITY
PT 2LB
PS 2LB
TT 2LB
T_{FUEL}
$\left[\frac{M\sqrt{TT}}{CD\ A\ PT}\right]_{2LB}$
$\frac{PT2LB-PS2LB}{PS\ 2LB}$
CD A
M2LB
TT4
PS2
TT2
GUESS TT3 = ·5 TT4
CALCULATION OF
M2 M3
PT2 PT3
$A = \frac{M3\ \sqrt{TT3}/PT3-(M3\ \sqrt{TT3}/PT3)_{GIVEN}}{(M3\ \sqrt{TT3}/PT3)_{GIVEN}}$
FHP
G2
G3
MFM
F2
cp
R
KCC
$\left(\frac{M3\sqrt{TT3}}{PT3}\right)_{GIVEN}$
MBL
IPC
HP3
HPC
TT3 = TT2 + (1+A)(TT3 - TT2)
NO
$|A| \leq 0.001$
YES
M2LM = (M2 + MBL)/[(1 - IPC)(1 - HP3 - HPC)]
M1 = M2LB + M2LM
M1
A 1
GROSS-THRUST-SUBROUTINE
PT4LB
PT4
PS6
AJ
$\frac{FG}{PAMB \cdot AJ}$
AJ
R/H
DRY
$\frac{PMIX}{PAMB}$ or $\frac{PS6}{PAMB}$
PAMB
VT
FG = [FG/PAMB · AJ] · PAMB · AJ
FN = FG - M1 · VT

FIG. 4

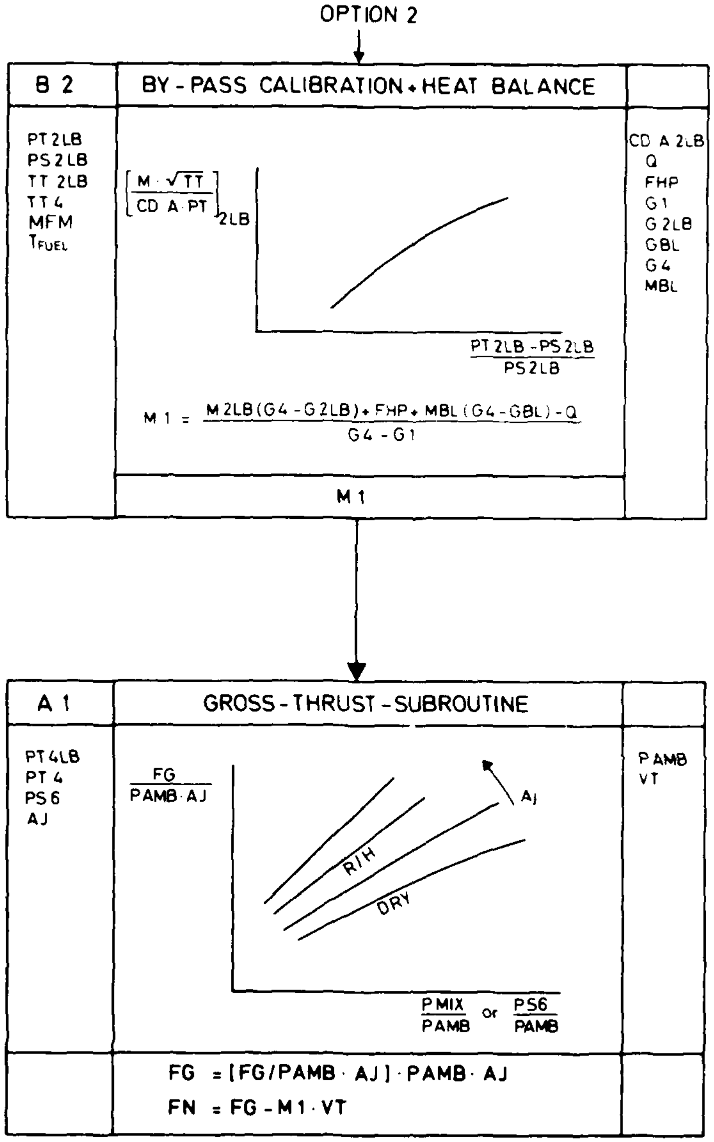

OPTION 2
B 2
BY-PASS CALIBRATION + HEAT BALANCE
PT2LB
PS2LB
TT 2LB
TT4
MFM
T_{FUEL}
$\left[\frac{M \cdot \sqrt{TT}}{CD\ A \cdot PT}\right]_{2LB}$
$\frac{PT2LB-PS2LB}{PS2LB}$
$M1 = \frac{M2LB(G4-G2LB)+FHP+MBL(G4-GBL)-Q}{G4-G1}$
CD A 2LB
Q
FHP
G1
G 2LB
GBL
G4
MBL
M 1
A 1
GROSS-THRUST-SUBROUTINE
PT4LB
PT 4
PS 6
AJ
$\frac{FG}{PAMB \cdot AJ}$
AJ
R/H
DRY
$\frac{PMIX}{PAMB}$ or $\frac{PS6}{PAMB}$
PAMB
VT
FG = [FG/PAMB · AJ] · PAMB · AJ
FN = FG - M1 · VT

FIG. 3

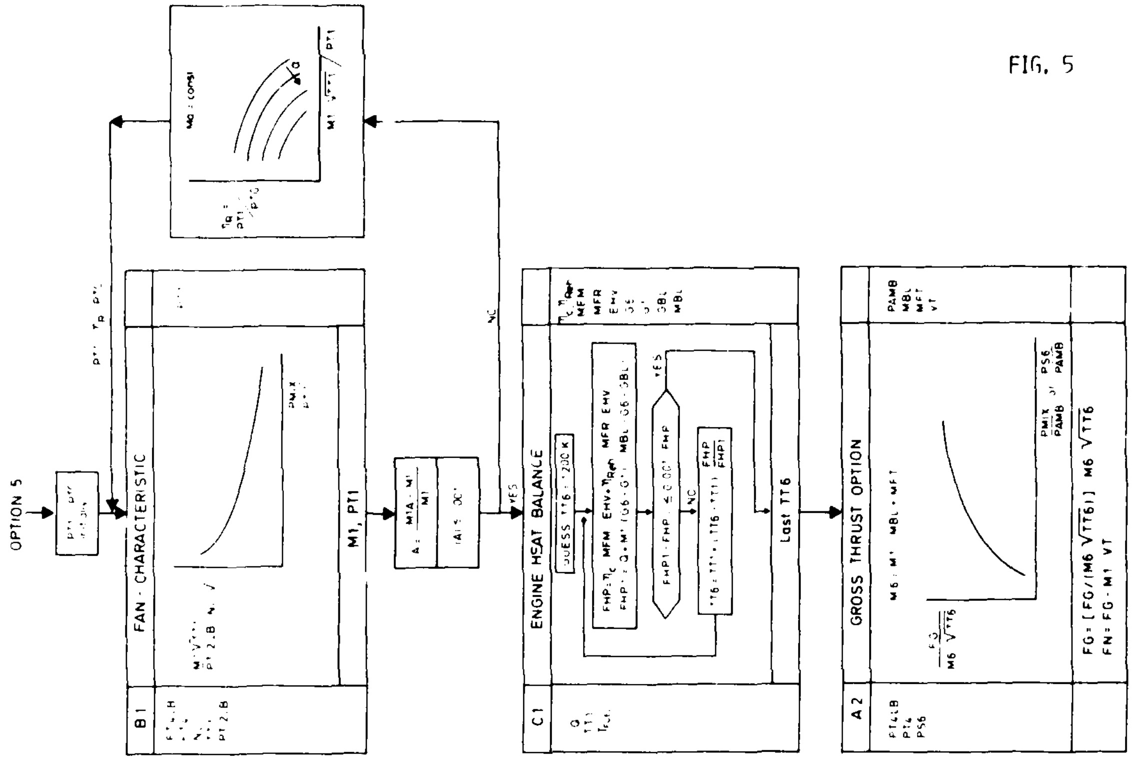

OPTION 5
FAN - CHARACTERISTIC
B1
M1, PT1
YES
NO
ENGINE HEAT BALANCE
C1
Last TT6
GROSS THRUST OPTION
A2
FG = [FG/(M6 √TT6)] M6 √TT6
FN = FG - M1 VT

FIG. 5

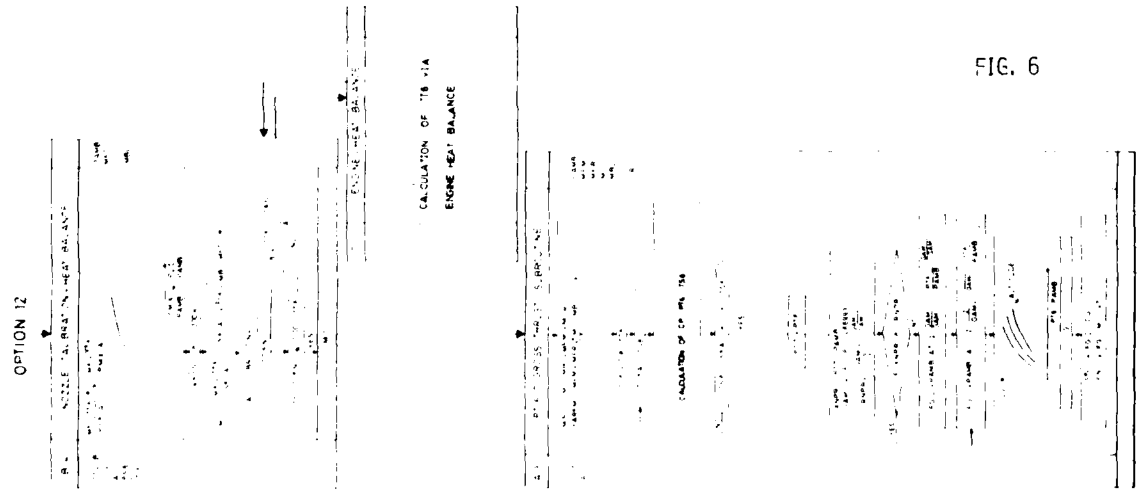

OPTION 12
CALCULATION OF TT6 VIA
ENGINE HEAT BALANCE

FIG. 6

FIG. 7

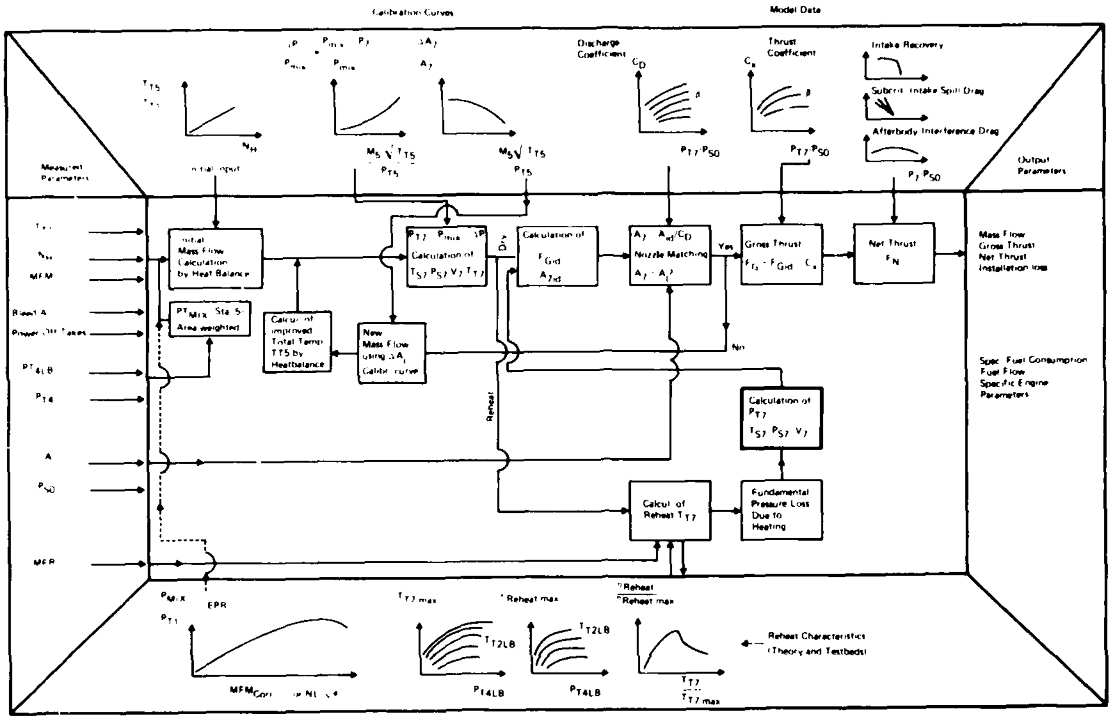


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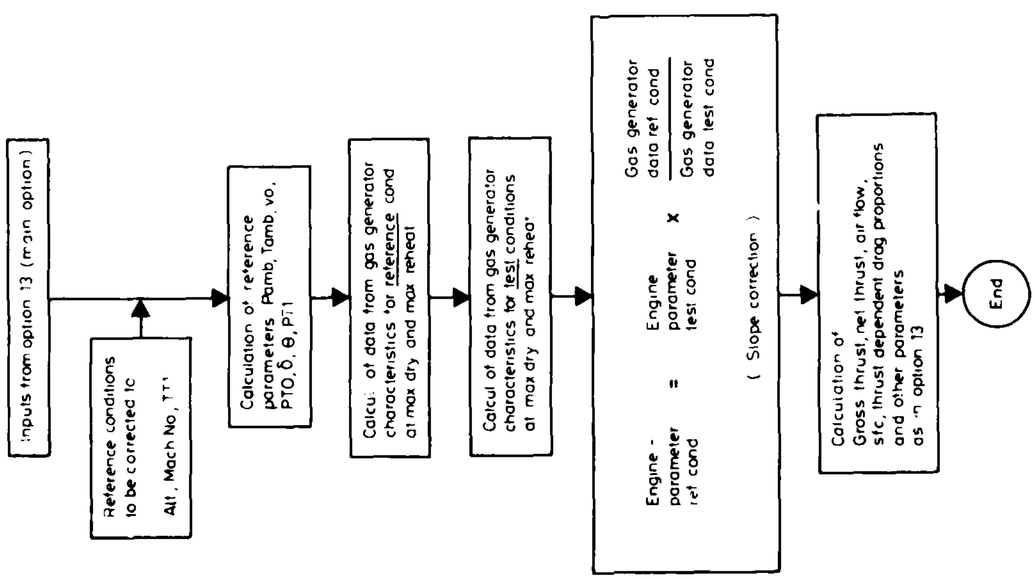
FIG. 8



TEST ENGINE PERFORMANCE CORRECTION

FIG. 9

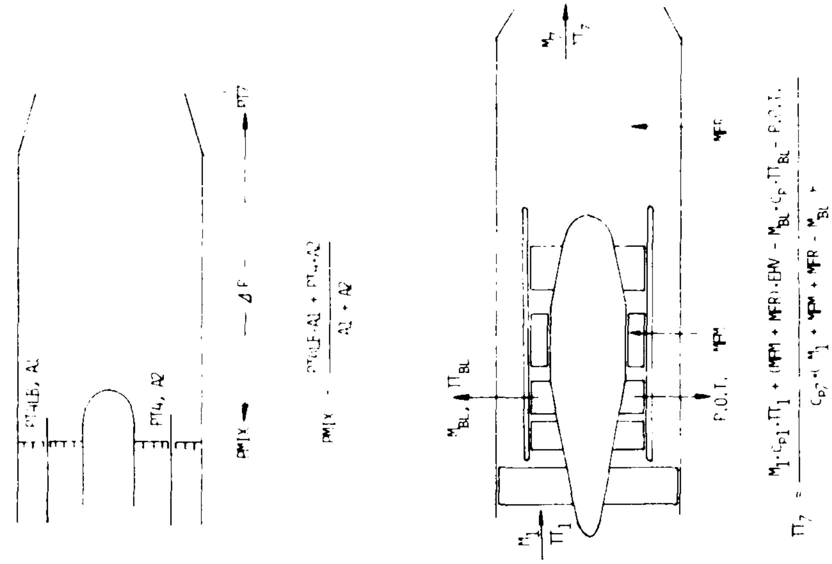
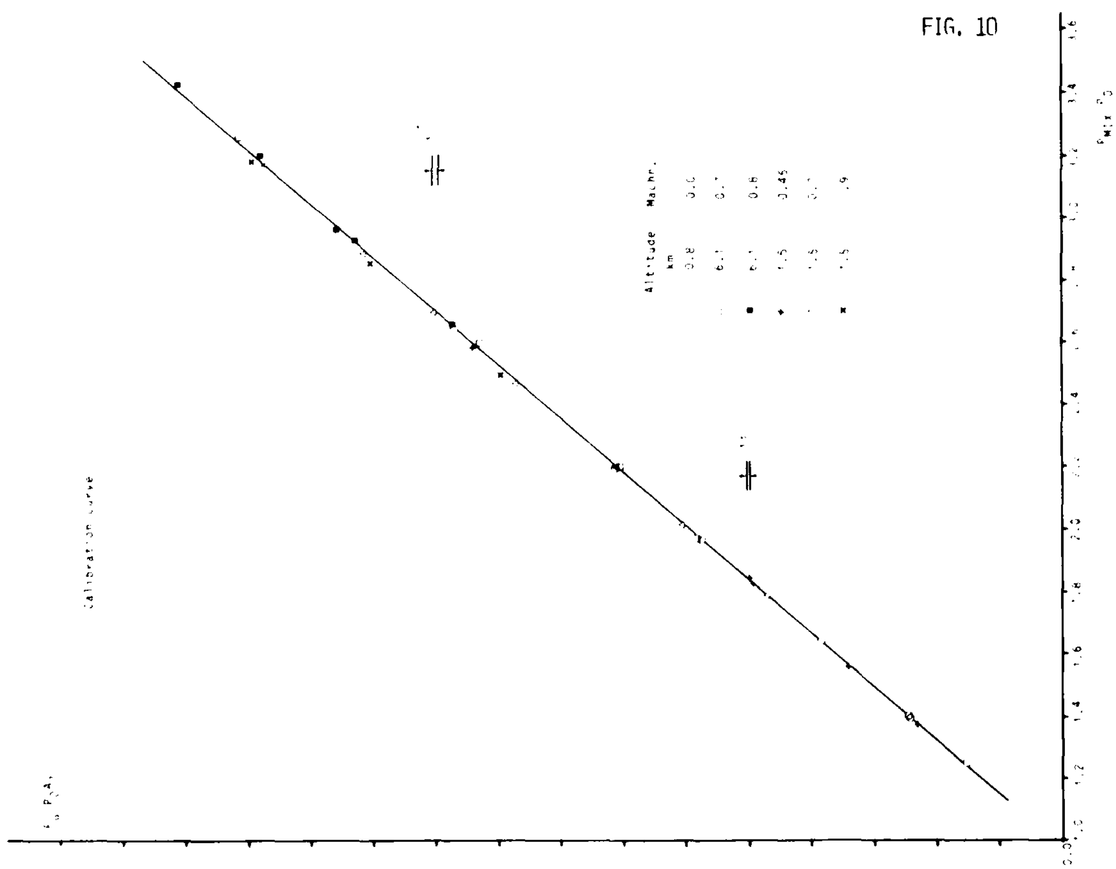


FIG. 10



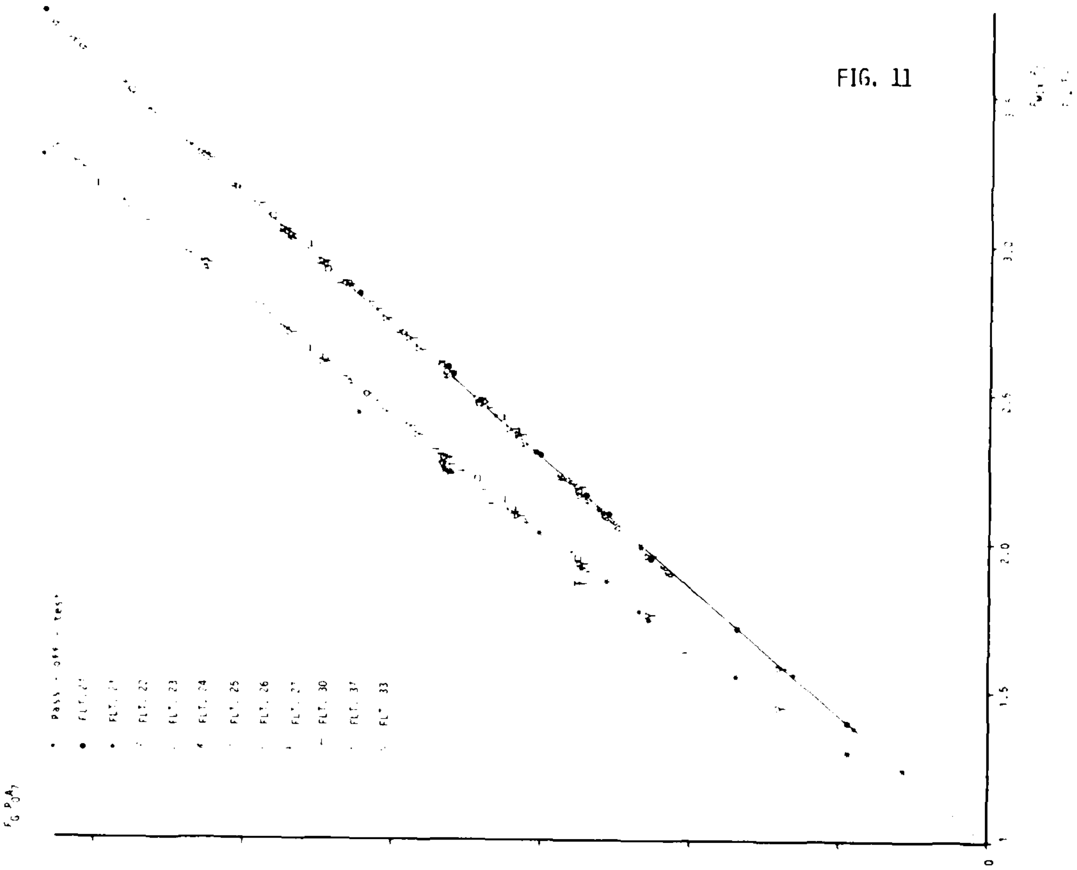


FIG. 11

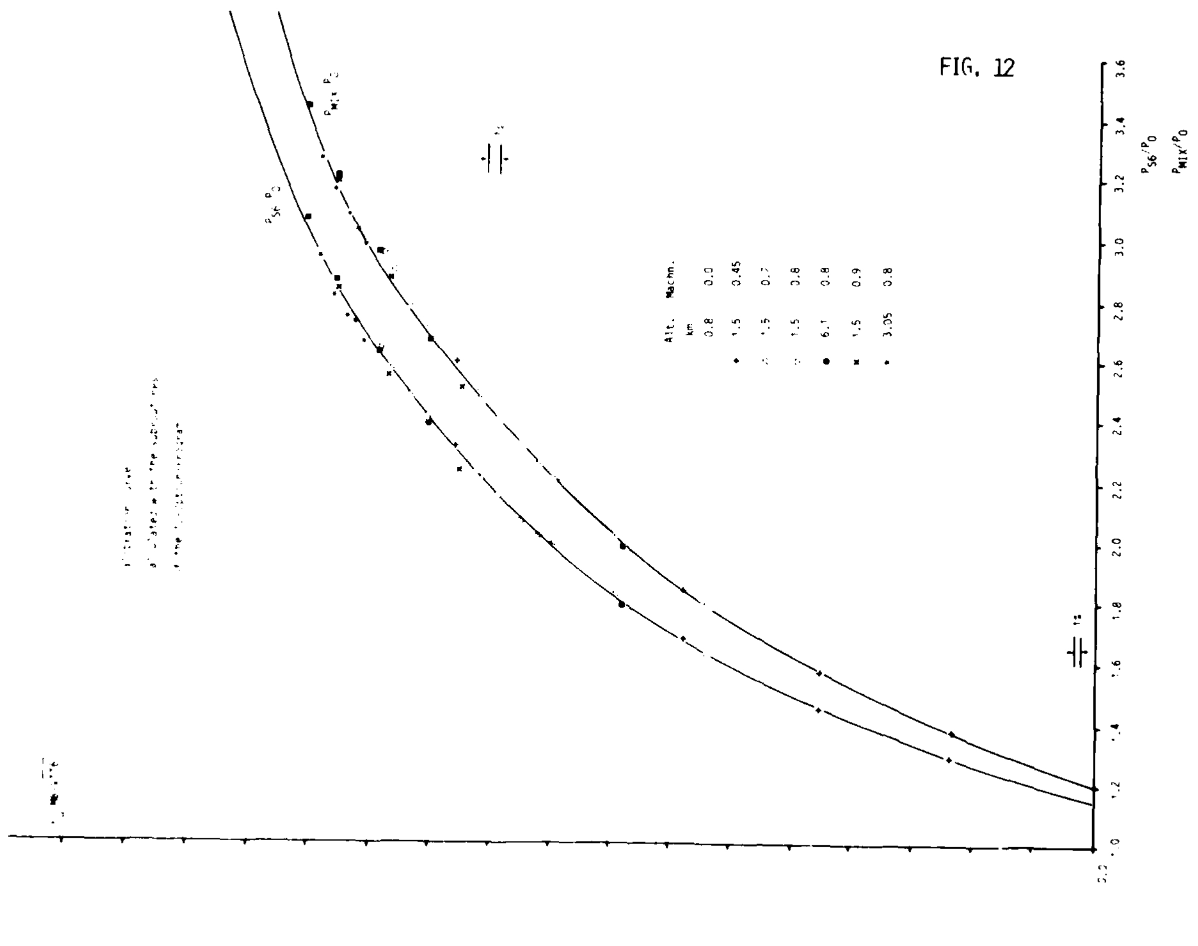


FIG. 12

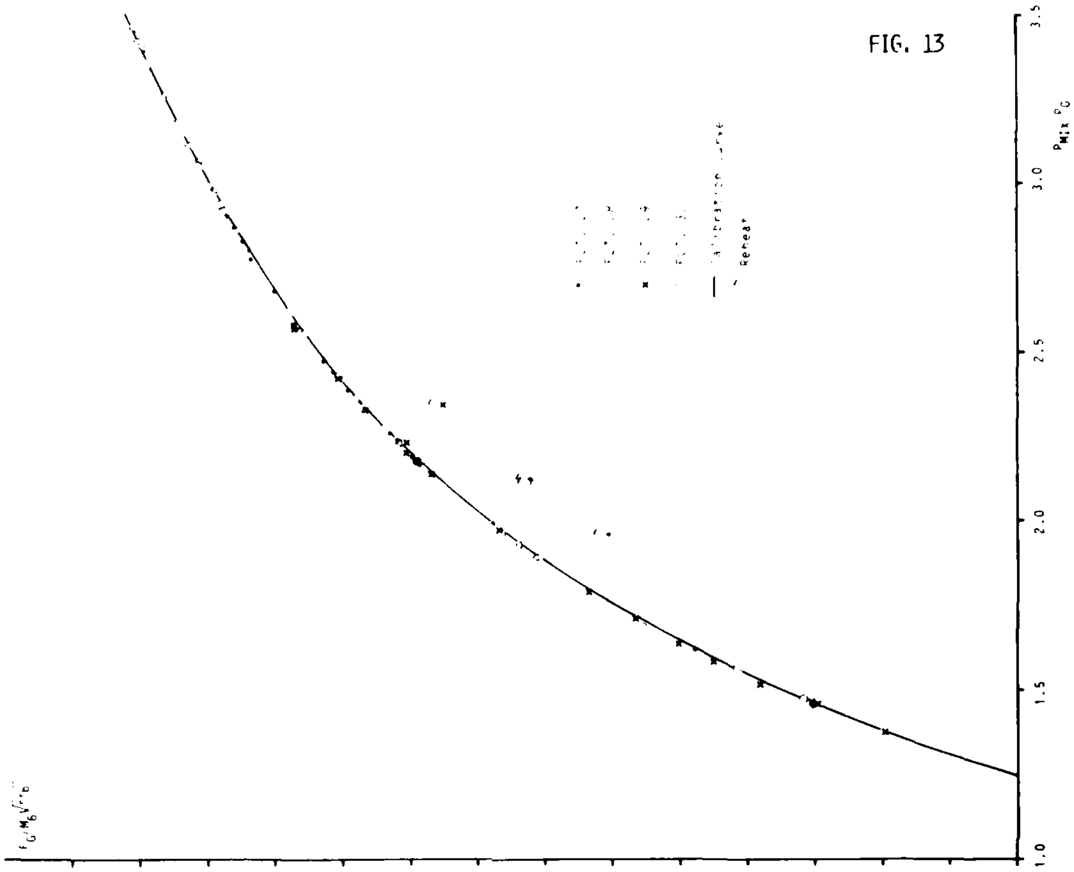

Calibration Curve
Repeat

FIG. 13

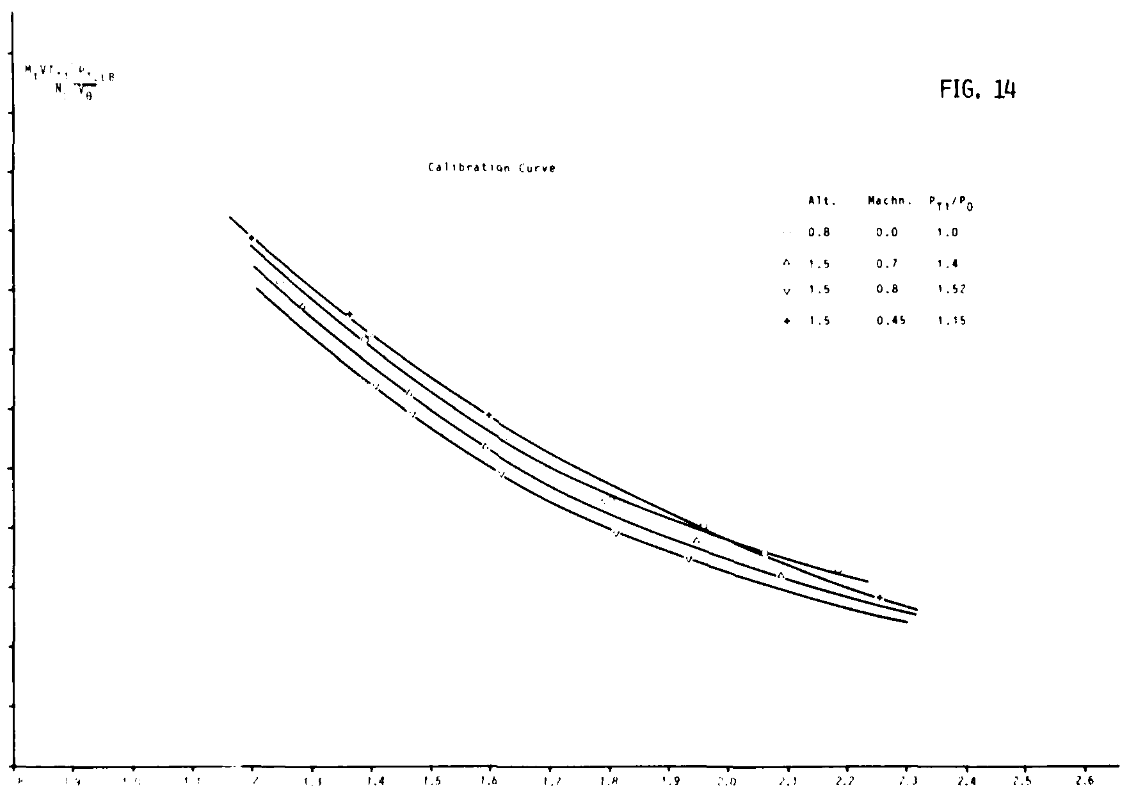

Calibration Curve
Alt. Machn. P_Tt/P_0
0.8 0.0 1.0
1.5 0.7 1.4
1.5 0.8 1.52
1.5 0.45 1.15
1.4
1.5
1.6
1.7
1.8
1.9
2.0
2.1
2.2
2.3
2.4
2.5
2.6

FIG. 14

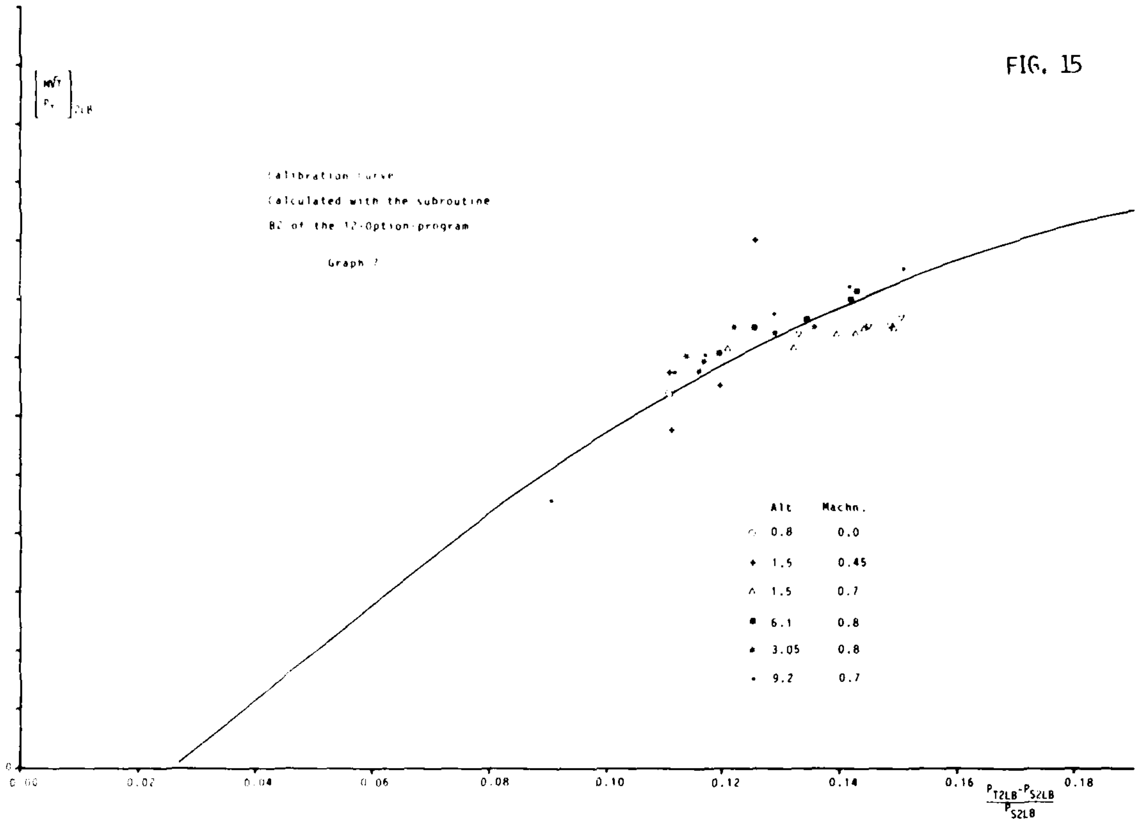

Calibration curve
Calculated with the subroutine
B2 of the 12-Option-program
Graph 7
Alt Machn.
0.8 0.0
1.5 0.45
1.5 0.7
6.1 0.8
3.05 0.8
9.2 0.7
0.00
0.02
0.04
0.06
0.08
0.10
0.12
0.14
0.16
0.18

FIG. 15

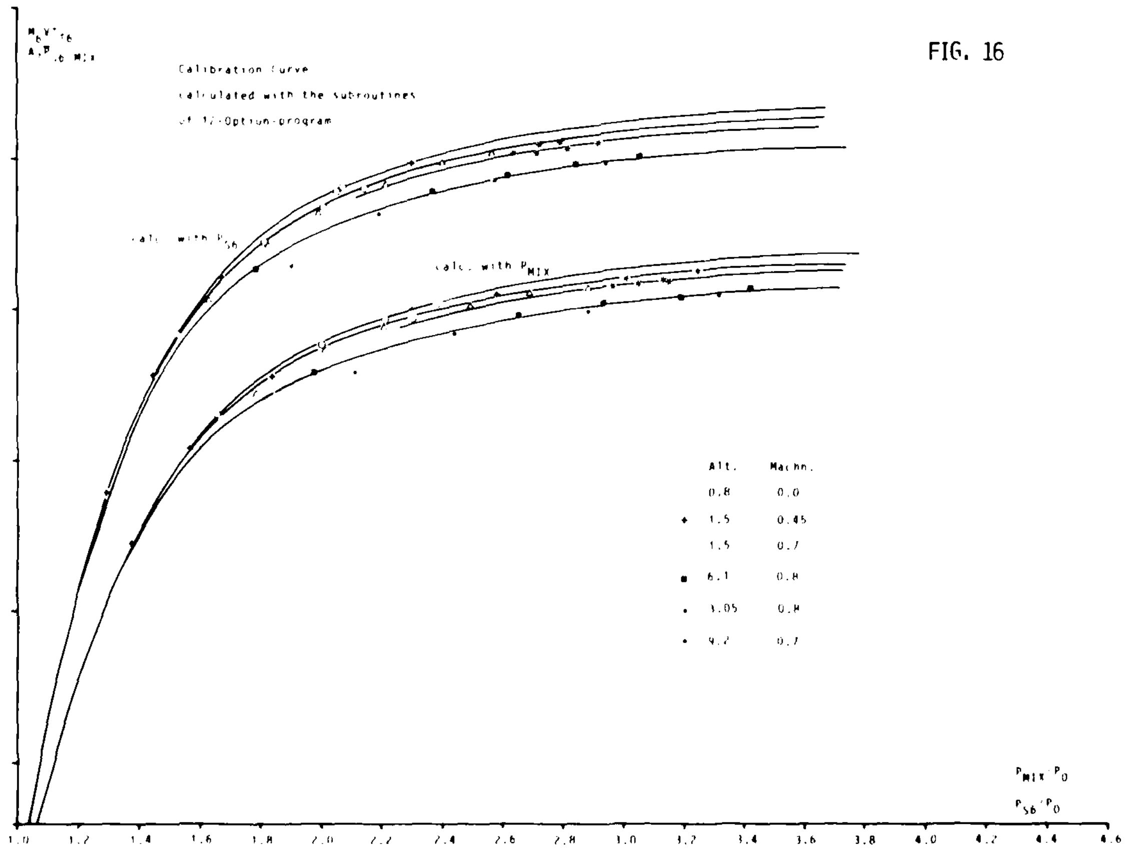
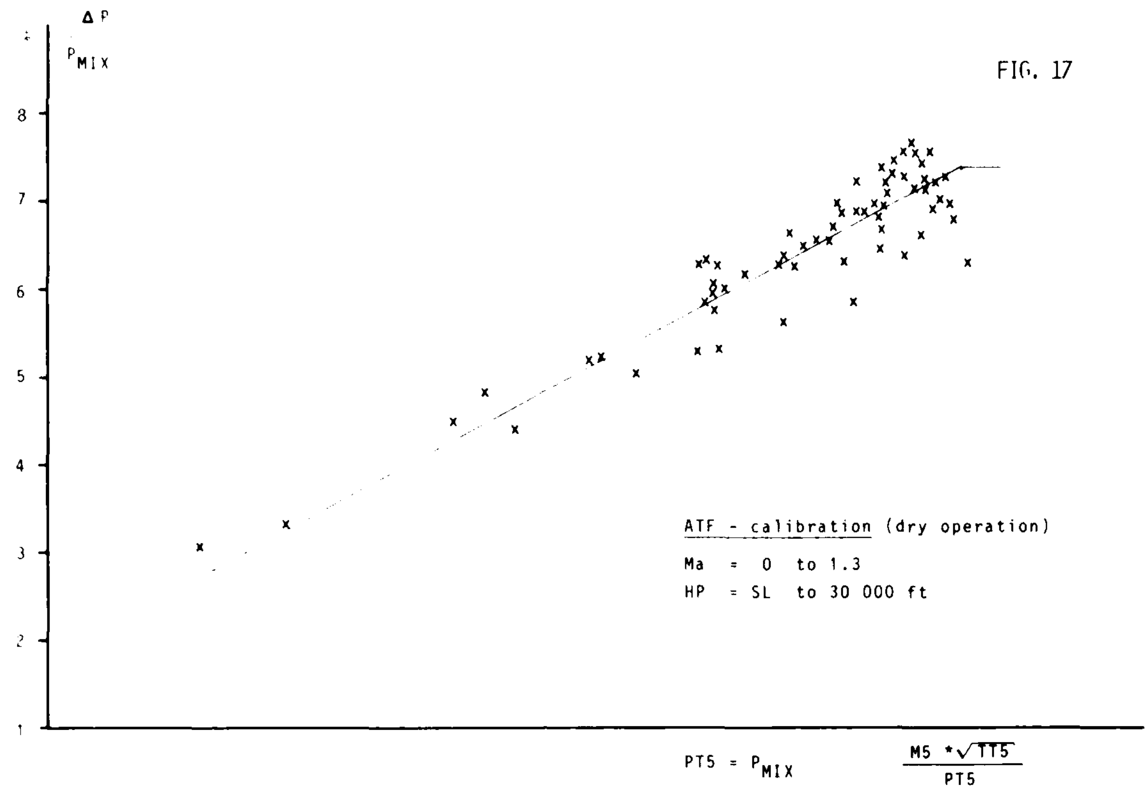

Calibration curve
calculated with the subroutines
of 12-Option-program
calc. with P_{MIX}
Alt. Machn.
0.8 0.0
1.5 0.45
1.5 0.7
6.1 0.8
3.05 0.8
9.2 0.7
1.0
1.2
1.4
1.6
1.8
2.0
2.2
2.4
2.6
2.8
3.0
3.2
3.4
3.6
3.8
4.0
4.2
4.4
4.6

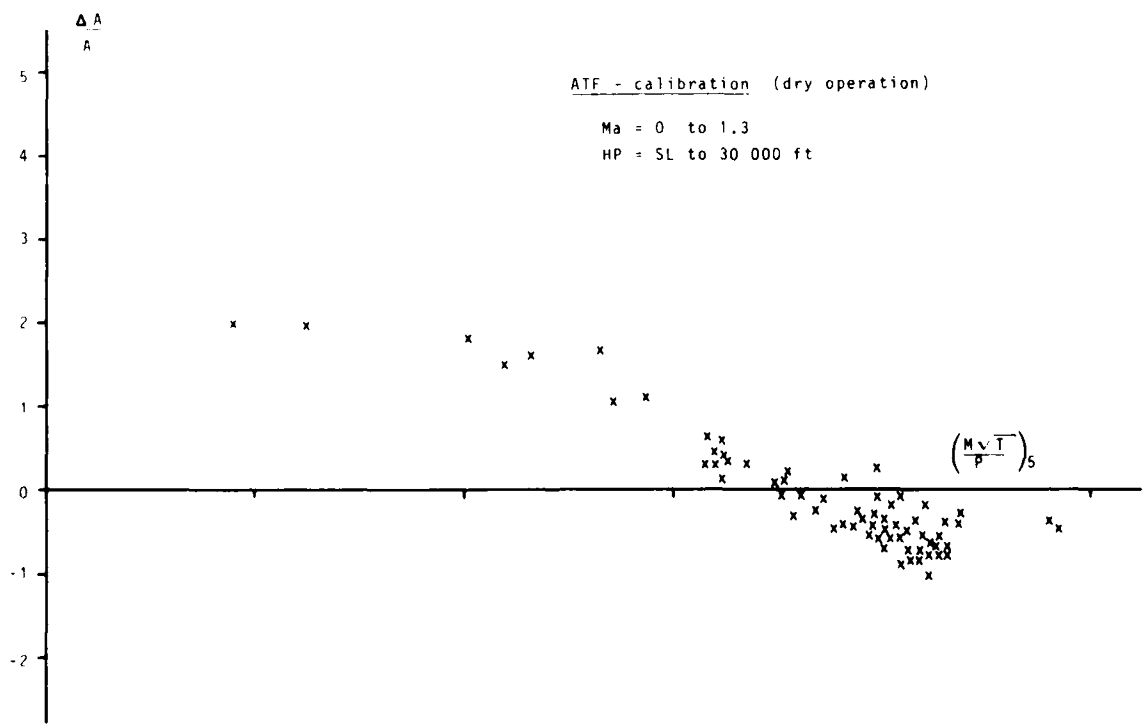
FIG. 16

FIG. 17



ΔP
P_{MIX}
8
7
6
5
4
3
2
1
ATF - calibration (dry operation)
Ma = 0 to 1.3
HP = SL to 30 000 ft
PT5 = P_{MIX}
$\frac{M5 * \sqrt{TT5}}{PT5}$

FIG. 18



$\frac{\Delta A}{A}$
5
4
3
2
1
0
-1
-2
ATF - calibration (dry operation)
Ma = 0 to 1.3
HP = SL to 30 000 ft
$\left(\frac{M\sqrt{T}}{P}\right)_5$

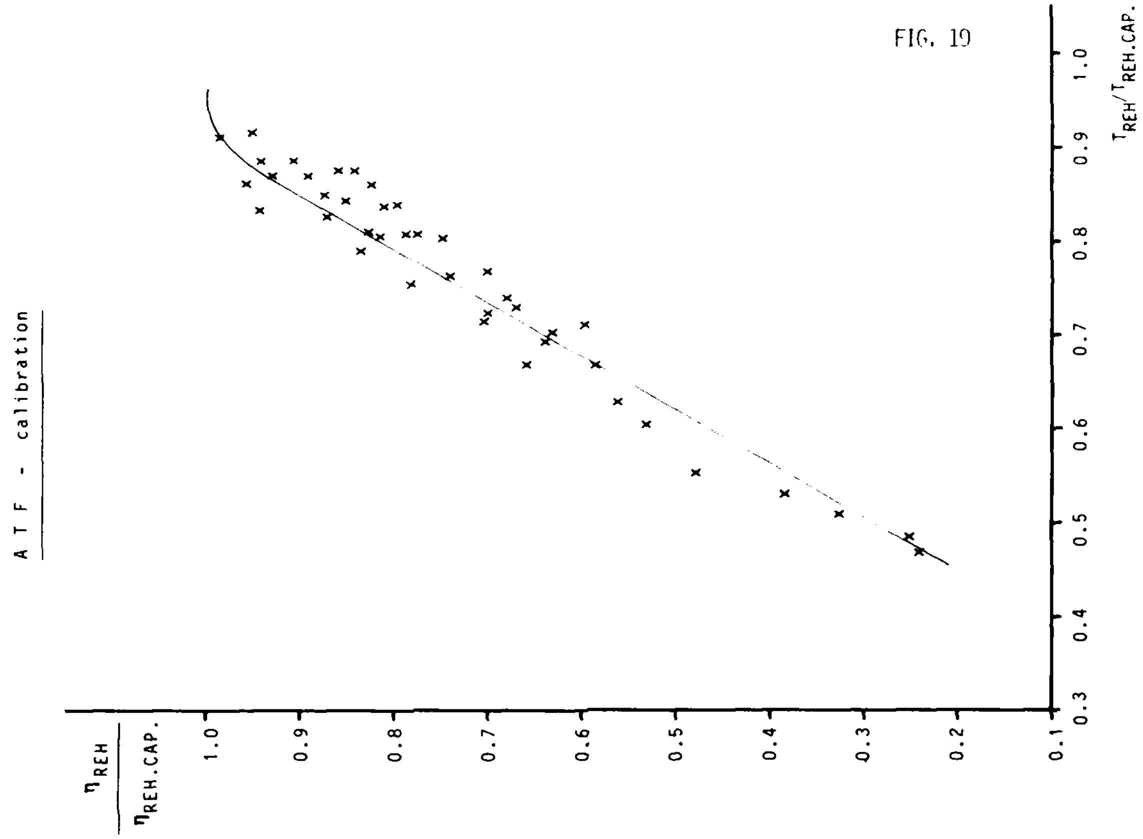

A T F - calibration
$\frac{\eta_{REH}}{\eta_{REH.CAP.}}$
1.0
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
$T_{REH}/T_{REH.CAP.}$
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0

FIG. 19

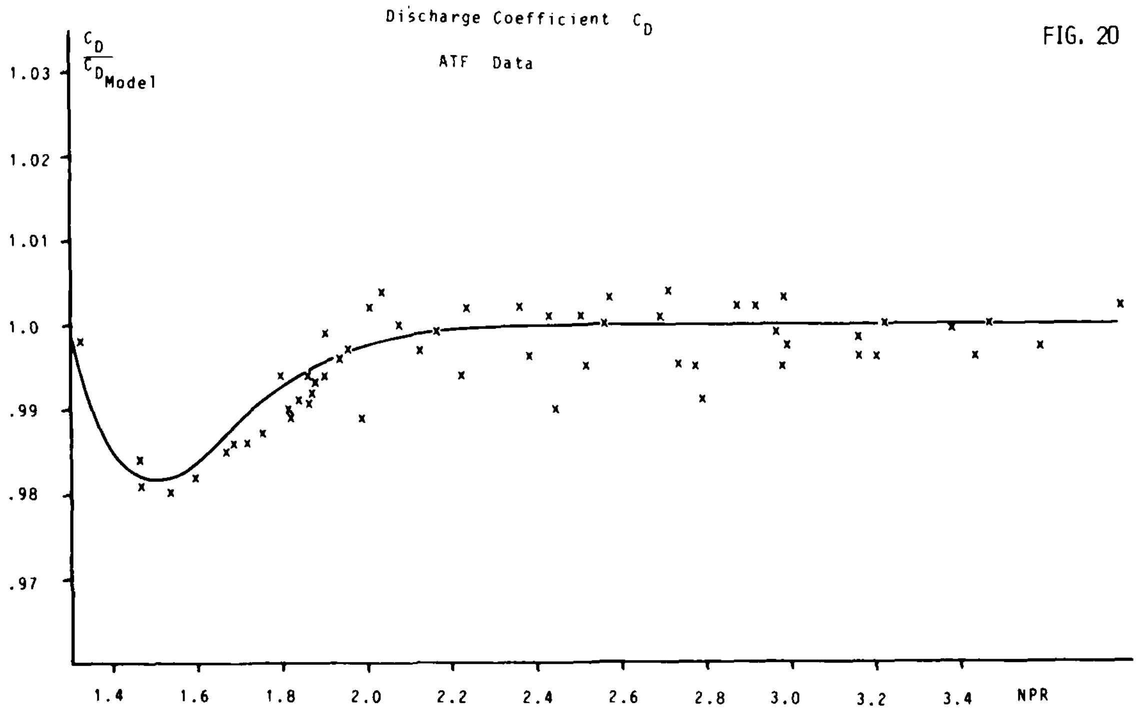

Discharge Coefficient C_D
ATF Data
$\frac{C_D}{C_{D_{Model}}}$
1.03
1.02
1.01
1.0
.99
.98
.97
1.4
1.6
1.8
2.0
2.2
2.4
2.6
2.8
3.0
3.2
3.4
NPR

FIG. 20

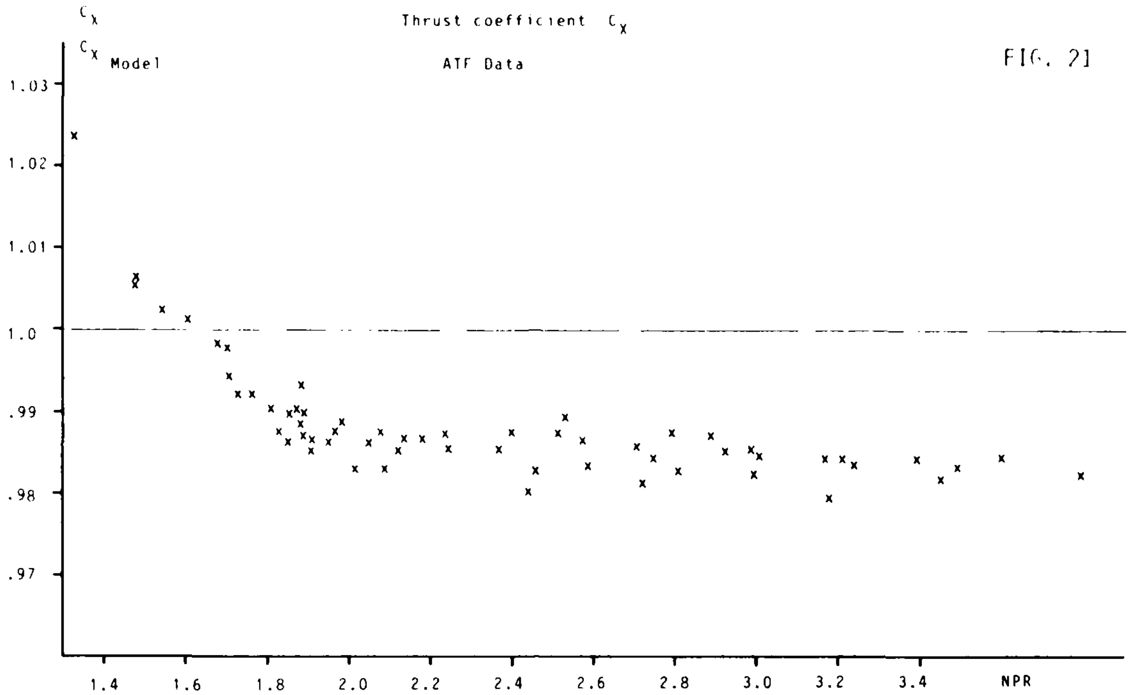

Thrust coefficient C_X
C_X
C_X Model
ATF Data
1.03
1.02
1.01
1.0
.99
.98
.97
1.4
1.6
1.8
2.0
2.2
2.4
2.6
2.8
3.0
3.2
3.4
NPR

FIG. 21

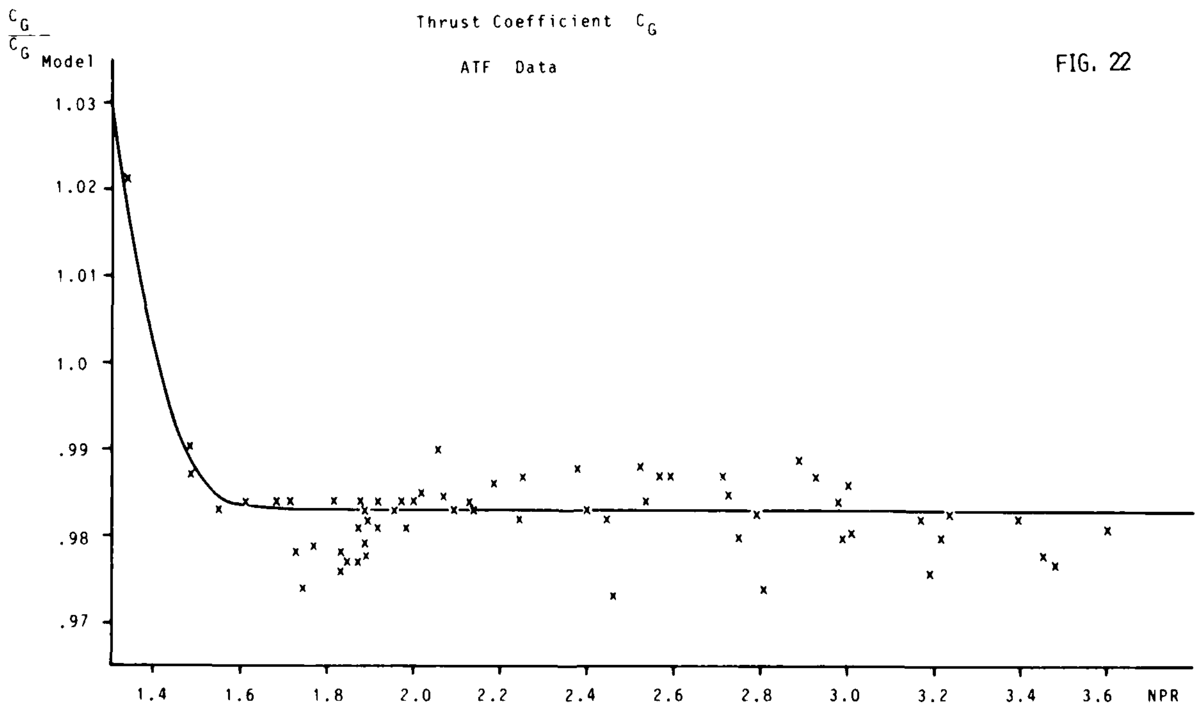

Thrust Coefficient C_G
C_G
C_G Model
ATF Data
1.03
1.02
1.01
1.0
.99
.98
.97
1.4
1.6
1.8
2.0
2.2
2.4
2.6
2.8
3.0
3.2
3.4
3.6
NPR

FIG. 22

FIG. 23

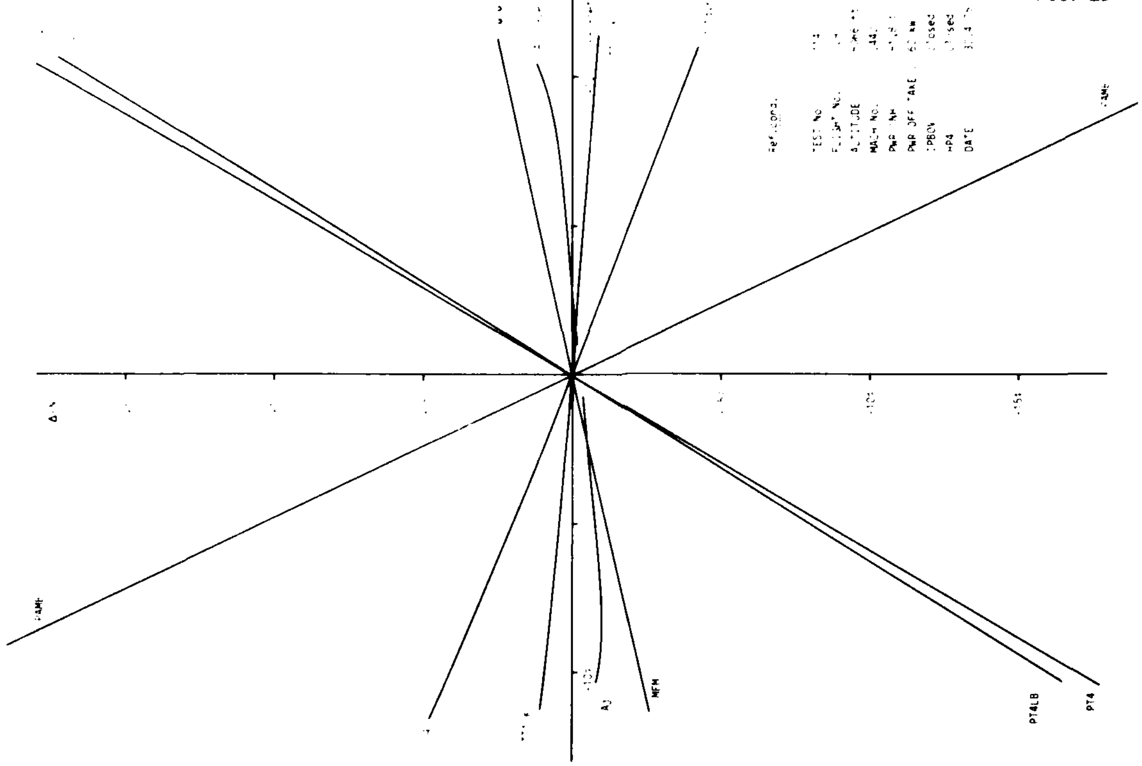
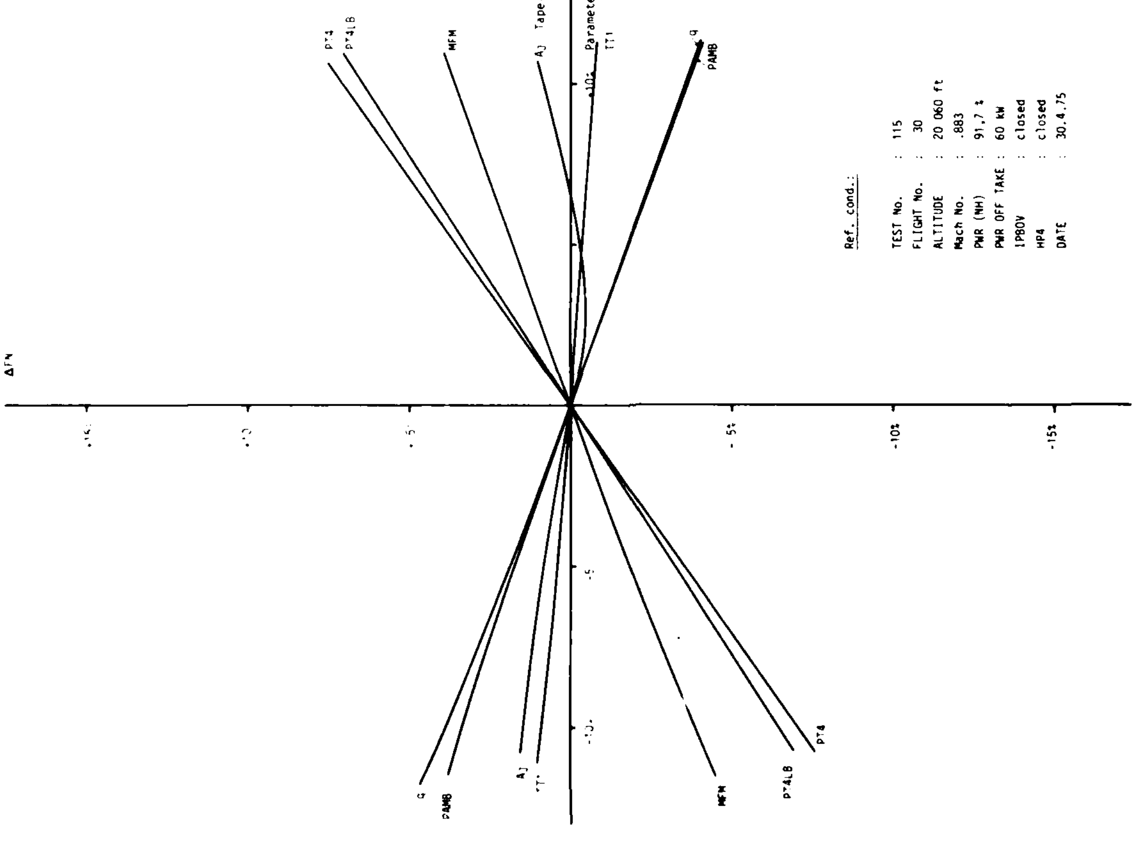

PAMB
PT4LB
PT4
MFM
AJ
ΔFN
-10%
-15%

FIG. 24


Ref. cond.:
TEST No. : 115
FLIGHT No. : 30
ALTITUDE : 20 060 ft
Mach No. : .883
PWR (NH) : 91,7 %
PWR OFF TAKE : 60 kW
IPBOV : closed
HP4 : closed
DATE : 30.4.75
PT4
PT4LB
MFM
AJ Tape
Parameter
TT1
q
PAMB
ΔFN
- 5%
-10%
-15%

DISCUSSION

Don Rudnitski, NRCE, Ca

How good does the engine calibration hold during the life of the engine? Also, if some components have to be replaced in service, is a recalibration required?

Is the calibration also a function of measured total pressure profile?

Author's Reply

Since the flight engines ATF tested so far have been pre- *OR* post-calibrated, rather than pre- *AND* post-calibrated, the answer can be given only indirectly, referring to the thrust evaluation method presented. By calibrating the nozzle and measuring down stream of all components eventually to be replaced, all effects of deterioration due to aging are automatically included. Thus the thrust may change, but not the calibration itself.

If components are changed, a careful comparison of the original SL calibration (pass off test) with the new one will give an answer whether a new ATF calibration is required.

No, because all pressure probes are ganged, thus giving a mean pressure.

LES ESSAIS EN VOL DE MOTEURS D'AVIONS DE COMBAT - METHODES DE VALIDATION EMPLOYEES PAR LE CEV

Par
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et
l'Ingénieur Principal des Etudes et Techniques de l'Armement TAPELLA
Centre d'Essais en Vol
13128 ISTRES AIR, France

RESUME

Pour les moteurs d'avions de combat, la validation du standard de série ne peut être définitivement acquise qu'après des essais en vol. Les essais de validation effectués par le CEV permettent ainsi de s'assurer du bon fonctionnement de l'ensemble des moteurs de série dans les conditions d'emploi les plus critiques et de vérifier en vol leur conformité à certaines exigences contractuelles et réglementaires. Après recensement des paramètres influents et inventaire des conditions d'utilisation qui doivent faire l'objet d'essais en vol, le principe des méthodes employées sera exposé. Ces méthodes consistent soit à essayer des moteurs"extrêmes" construits à partir d'éléments choisis ou réglés chacun en limite de tolérance, soit à simuler les configurations extrêmes par des réglages appropriés des régulateurs.

NOTATIONS EMPLOYEES

N	vitesse de rotation
T	température totale à l'entrée du compresseur
Π	taux de compression
P	pression de sortie du compresseur
C	débit instantané de carburant
d	densité de carburant

1. INTRODUCTION : BUTS DES ESSAIS DE VALIDATION

A la mise en service d'un nouvel avion de combat, il convient d'avoir la certitude que l'ensemble des moteurs de série fonctionneront correctement dès lors que chacun d'entre eux aura satisfait aux épreuves individuelles de réception.

Ce résultat ne peut être acquis qu'après la réalisation d'un programme de démonstration comportant notamment des essais

- au banc, en conditions de vol simulées

- en vol sur l'avion de combat auquel le moteur est destiné.

La démonstration en vol se fait dès que le standard de série est figé. En France, pour les programmes nationaux, cette tâche incombe au CENTRE D'ESSAIS EN VOL, établissement de la Direction Technique des Constructions Aéronautiques.

Au cours de cette démonstration, le CEV doit :

- vérifier l'adaptation du moteur de série à l'avion de série dans les conditions d'emploi prévues par les clauses techniques et s'assurer que les exigences réglementaires et contractuelles vérifiables en vol sont satisfaites

- valider des consignes et un domaine d'utilisation pour l'ensemble des moteurs de série en tenant compte de la dispersion liée aux tolérances de réglage et de fabrication.

De tels essais doivent donc être menés de façon à rendre compte du comportement des moteurs les "moins bons" susceptibles d'être fabriqués en nombre significatif. C'est pourquoi le CEV a été amené en accord avec les constructeurs, à adopter certaines méthodes de travail.

Nous vous proposons, après une brève énumération des essais en vol à entreprendre dans le cadre de la démonstration, d'exposer les principes de ces méthodes.

2. INVENTAIRE DES ESSAIS A EFFECTUER

La nature des essais à entreprendre et leur volume dépendent de plusieurs facteurs tels que la complexité du moteur et la nature des missions attribuées à l'avion.

Il est clair par exemple qu'un appareil devant évoluer dans un domaine de vol très étendu doit faire l'objet d'essais plus nombreux qu'un avion destiné uniquement à la basse altitude. De même l'existence de dispositifs particuliers (réchauffe, appauvrisseur de tir, régulation de secours) contribue à accroître le nombre de vérifications.

L'énumération qui va suivre constitue une enveloppe des essais en vol habituellement effectués dans le cadre d'une validation de moteur d'avion de combat classique.

Ces vérifications peuvent se ranger en trois catégories :

2.1. Essais relatifs à l'utilisation courante.

- vérification de l'absence de phénomènes anormaux tels que décrochages compresseur, instabilités, "blocages" de régime et dévissages, ou détermination des zones d'apparition de ces phénomènes
- contrôle du fonctionnement avec réchauffe
- évaluation de la pilotabilité et de la rapidité des transitoires, notamment en patrouille serrée
- vérification du respect des limites de fonctionnement validées (vitesses de rotation, températures de turbine, environnement thermique et vibrations, etc...)
- mesures de performances, notamment pour ce qui concerne les pertes liées aux conditions d'avionnage.

2.2. Pannes et dispositifs de secours.

- étude des conséquences des défaillances de certains équipements, notamment de calculateur
- vérification de l'efficacité et du fonctionnement correct des dispositifs de secours tels que surpuissance ou régulation de secours
- rallumages en vol

2.3. Utilisations particulières.

- étude du comportement moteur aux hautes incidences et en vrille, surtout lorsque celui-ci doit impérativement rester sain comme cela est le cas pour les avions "école"
- évaluation de la pilotabilité et du comportement moteur lors de ravitaillements en vol lorsque l'alimentation en air risque d'être perturbée par la présence de la perche et du panier de ravitaillement en amont de l'entrée d'air
- vérification de la compatibilité avec les armements dont l'emploi est susceptible de perturber le fonctionnement des moteurs tels que canon, roquettes ou missiles et contrôle de l'efficacité des dispositifs de protection associés (décharges, appauvrisseur de tir...)
- étude du comportement en givrage naturel
- essais en conditions climatiques extrêmes (temps chaud et temps froid)
- contrôle du fonctionnement en environnement radio-électrique perturbé, notamment pour ce qui concerne la partie électronique de la régulation.

Cette énumération non exhaustive donne un aperçu de la diversité et du volume des essais à prévoir. Dans la réalité, le CEV bénéficie au moment de son intervention de l'expérience du constructeur et les problèmes de fonctionnement qui peuvent subsister à ce stade du développement sont parfaitement connus. Cela permet bien souvent de restreindre le champ des investigations et de porter l'accent sur les points jugés les plus critiques.

3. IDENTIFICATION DES PROBLEMES POTENTIELS DE FONCTIONNEMENT

Aucune famille de moteurs n'est totalement exempte de problèmes de fonctionnement, fussent-ils mineurs.

Dans le cas le plus favorable, si la mise au point a abouti, ils ne se manifestent que dans les conditions extrêmes d'utilisation et ne concernent qu'une faible proportion de moteurs.

La nature de ces problèmes de fonctionnement est sauf exception connue bien avant que ne débutent les essais officiels de validation. Deux types de phénomènes sont le plus souvent rencontrés :

3.1. Le décrochage compresseur :

Ce phénomène s'observe lorsque la marge de décrochage devient trop faible.

Pour un turboréacteur simple corps simple flux, cette marge peut s'exprimer dans le champ Π, $N/\sqrt{T}$ par l'écart $\Delta\Pi$ entre point de fonctionnement et ligne de décrochage. On distingue deux cas représentés sur la figure n° 1 :

- le fonctionnement stabilisé pour lequel la marge est la distance entre lignes d'adaptation et de décrochage
- les transitoires d'accélération pour lesquels la marge est la distance entre butée d'accélération et ligne de décrochage.

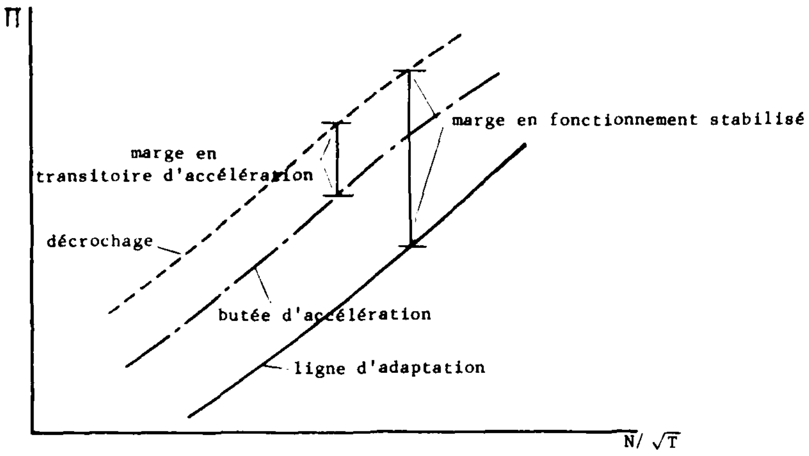

figure n° 1 - Marges de décrochage

3.2. Le "blocage" ou le dévissage :

Ces phénomènes correspondent à une annulation de la marge d'accélération qui peut se représenter dans le champ Π, N/ √T par la distance entre ligne d'adaptation et butée d'accélération (figure n° 2). Il y a blocage ou dévissage lorsque ces deux lignes se croisent.

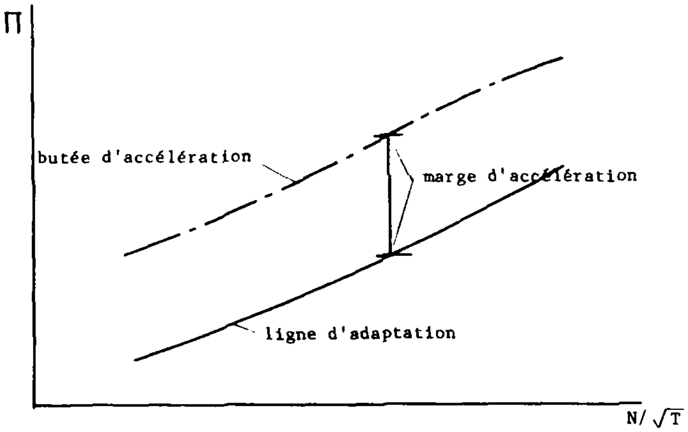

figure n° 2 - Marge d'accélération

Dans un souci de simplification, nous nous limiterons dans la suite de l'exposé aux seuls phénomènes de décrochage et de blocage.

Par suite des dispersions de fabrication, les limites d'apparition de ces phénomènes varient d'un moteur à l'autre. De plus, pour un même moteur, elles dépendent d'un certain nombre de paramètres.

Il convient donc de connaître tous les paramètres susceptibles de faire varier ces limites.

4. PARAMETRES INFLUENTS

Les paramètres qui favorisent le décrochage ou le blocage peuvent se classer en plusieurs catégories :

4.1. Paramètres extérieurs.

Certains paramètres de vol ont une influence plus ou moins marquée sur les marges de décrochage ou d'accélération. Parmi ceux-ci il convient de citer l'altitude, le nombre de Mach, la température extérieure, l'incidence ou le dérapage dont dépendent les caractéristiques d'entrée d'air, notamment l'hétérogénéité. Certains d'entre eux peuvent avoir une influence sur la position de la butée dans le champ compresseur. Par ailleurs, les variations rapides de température d'impact peuvent changer l'état thermique du moteur et donc son adaptation (déformations de la veine susceptibles de modifier les caractéristiques de turbine ou de compresseur).

L'affichage de ces paramètres n'est pas toujours maîtrisable, d'où la nécessité de transcriptions lorsque les essais ne sont pas réalisés dans les conditions prévues. Cela est particulièrement vrai pour la température de l'atmosphère qui peut être très différente de celle désirée si les essais n'ont pas lieu pendant la saison favorable.

4.2. Réglages du moteur.

Les paramètres susceptibles de faire varier les marges peuvent se classer en trois catégories :

- les réglages de régulation tels que celui de la butée d'accélération
- la géométrie de la veine (calage des aubages, sections de distributeurs de turbine ou de tuyères, jeux...)
- des paramètres de "qualité" tels que les caractéristiques des compresseurs ou de la turbine (position de la ligne de décrochage, pente des lignes iso-vitesse, rendements...).

Par exemple, sur un turboréacteur simple corps simple flux, le décrochage compresseur est favorisé par :

- le relèvement du débit de butée d'accélération au maximum de la tolérance (figure n° 3)
- le montage d'un distributeur de turbine de section minimale
- une ligne de décrochage "basse".

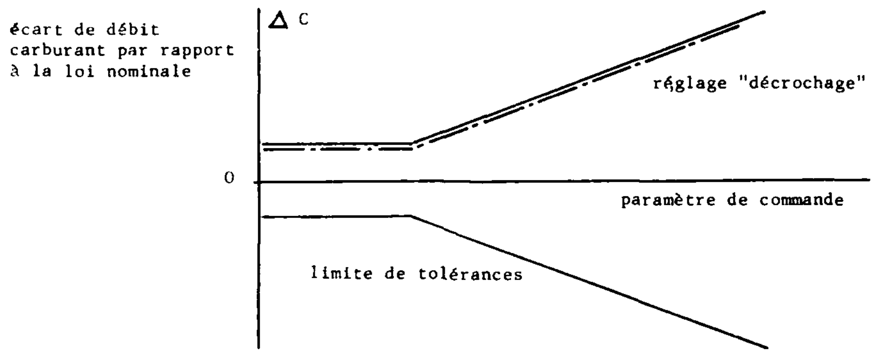

Figure n° 3 - tolérances de la loi de butée d'accélération

On peut ainsi définir des configurations "décrochage" et "blocage", obtenues par empilage des tolérances dans un sens défavorable, dont les marges respectives vis à vis de ces deux phénomènes sont les plus faibles.

A titre d'exemple, le tableau ci-dessous donne la définition de ces configurations pour un turboréacteur simple corps simple flux.

	Configuration "décrochage"	Configuration "blocage"
Butée d'accélération	Maximale	Minimale
Distributeur de turbine	Fermé	Ouvert
Tuyère	Fermée	Fermée
Ligne de décrochage	Basse	-

4.3. Ingrédients.

Les caractéristiques du carburant et du fluide employé pour la régulation peuvent avoir une grande influence. Par exemple, si le dosage du carburant est volumétrique, tout changement de densité se traduit par un changement de débit massique. Or pour un même type de carburant, les variations de densité peuvent être très importantes si l'on tient compte des tolérances et des variations de température comme le montre la figure n° 4.

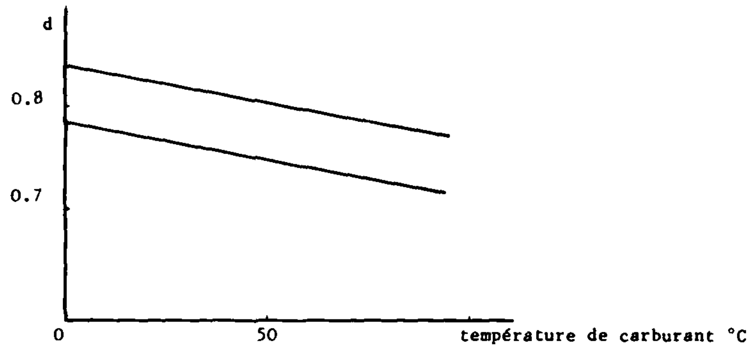

figure n° 4 - Plage de densité du TRO

En admettant que le pouvoir calorifique ne varie pas (ce qui est vrai en première approximation) et s'il n'y a pas de dispositif de compensation automatique, l'emploi de carburant dense ou froid favorise le décrochage et l'emploi de carburant léger ou chaud le blocage.

La présence d'un correcteur de densité sur le moteur permet le cas échéant de simuler des écarts de densité.

5. REPRESENTATIVITE DES ESSAIS

Il ressort de ce qui précède que pour valider l'adaptation du standard de série, il suffit de contrôler le fonctionnement des configurations extrêmes dans l'ensemble du domaine de vol revendiqué. En fait, l'empilage systématique de toutes les tolérances dans un sens défavorable n'est pas toujours réaliste dans la mesure où certains réglages ne sont pas rigoureusement indépendants et risque d'être inutilement contraignant.

Il y a lieu de tenir compte d'une part de la probabilité de réalisation de ces configurations extrêmes et d'autre part de la gravité des risques encourus. On peut par exemple admettre sur un bimoteur certains défauts de fonctionnement qui ne seraient pas acceptables sur un monomoteur.

Ces considérations conduisent parfois à choisir pour les essais des configurations légèrement moins pénalisantes que les configurations extrêmes et d'accepter un taux de couverture de la série inférieur à 100 %.

6. METHODES EMPLOYEES

Deux méthodes d'essais ont été expérimentées.

La première consiste à essayer des moteurs dont la qualité des composants et les réglages sont rigoureusement ceux des configurations extrêmes retenues. Cette méthode est appliquée depuis longtemps par le CEV. Elle a été employée avec succès pour la validation des versions récentes de la famille ATAR.

La deuxième consiste à utiliser un moteur dont les réglages sont quelconques mais préalablement identifiés et à simuler le comportement des configurations extrêmes par un déréglage approprié de la butée d'accélération. Cette méthode de simulation trouve sa justification dans l'équivalence vis à vis des phénomènes de blocage et de décrochage entre changement d'adaptation ou de ligne de décrochage et déplacement de même amplitude de la butée d'accélération dans le champ compresseur. Telle est la méthode qui a été adoptée pour le contrôle de l'adaptation du turboréacteur LARZAC à l'ALPHA-JET.

7. DISCUSSION : AVANTAGES ET INCONVENIENTS DE CHAQUE METHODE

1ère méthode :

Tout l'intérêt de cette méthode réside dans le fait qu'elle rend possible le contrôle du comportement réel des moteurs extrêmes dans tous ses aspects sans risque d'erreur d'interprétation.

En fait, l'impossibilité de maîtriser certains paramètres extérieurs tels que la température de l'atmosphère et la densité du carburant rend parfois nécessaire le recours à certaines simulations. Celles-ci sont réalisées le plus souvent par l'intermédiaire du correcteur de densité en affichant une valeur différente de celle préconisée.

Par ailleurs, la réalisation des configurations extrêmes suppose une sélection préalable des différents composants. Or, à ce stade du développement, le matériel déjà construit n'existe qu'en nombre limité, d'où un risque de ne pas disposer à coup sûr de tous les équipements les plus "critiques".

Enfin, la méthode n'est facilement applicable qu'aux moteurs de conception simple pour lesquels le nombre de configurations est peu élevé. Pour un turboréacteur double corps double flux par exemple, la prise en compte des risques de décrochage de chaque compresseur peut conduire à un accroissement prohibitif du nombre de configurations à réaliser.

2ème méthode :

La méthode de simulation est parfaitement adaptée aux moteurs complexes pour lesquels le nombre de paramètres influents et de configurations à essayer est élevé.Son principal intérêt est qu'il suffit en principe d'un seul moteur pour réaliser l'ensemble des essais. N'importe quel moteur prélevé à la sortie de la chaîne de production peut convenir pourvu qu'il soit parfaitement identifié. Chaque configuration s' obtient en remplaçant le régulateur d'origine par un régulateur dont les réglages ont été spécialement modifiés.

Il faut signaler en contrepartie un certain nombre d'inconvénients qui rendent nécessaires quelques aménagements :

- il ne peut pas être rendu compte simultanément de tous les aspects du fonctionnement d'une configuration. Par exemple les temps de reprise qui pourraient être mesurés lors de la simulation de la configuration "décrochage" n'ont a priori aucune signification.
- il n'est pas possible de simuler une configuration donnée par un réglage unique valable pour toutes les conditions de vol. Cela tient au fait que les lois de butée d'accélération sont en général des lois simplifiées qui ne tiennent pas forcément compte de tous les paramètres. Par exemple une loi de la forme

$$C = P \times f(N)$$

qui ne tient pas compte de la température T ne se transcrit pas de façon unique dans le champ (Π,N/ $\sqrt{T}$).

Il apparait donc nécessaire, soit de prendre des marges supplémentaires, soit de disposer d'un dispositif correcteur.

Le correcteur électronique de butée du LARZAC a rendu possible pour ce moteur la deuxième solution.

- Pour ce qui concerne la détermination des limites de décrochage, la simulation par l'intermédiaire de la loi de butée n'est valable qu'en transitoire d'accélération. Un autre moyen doit être employé pour rendre compte du fonctionnement en régime stabilisé. Ainsi, l'étude du comportement du LARZAC avec hautes incidences et en vrille a été effectué avec un moteur dont les marges de décrochage avaient été réduites par diminution des sections des distributeurs de turbine.

8. CONCLUSIONS

Pour garantir le fonctionnement de l'ensemble des moteurs de série sur avion, les essais en vol doivent rendre compte du comportement des moteurs extrêmes réalisables compte tenu des tolérances de réglage et de fabrication.

Deux méthodes sont employées par le CEV pour y parvenir. L'une consiste à réaliser ces moteurs extrêmes et à les essayer effectivement, l'autre à en simuler le comportement par des réglages appropriés de la régulation ou d'autres équipements.

Chacune d'entre elles a ses avantages et ses inconvénients. La première est bien adaptée aux moteurs de conception simple mais s'accorde mal à la complexité des moteurs modernes. Pour ces derniers, il vaut mieux avoir recours à la simulation malgré ses imperfections.

ESSAIS DE GRANDS COMPRESSEURS AU C.E.Pr

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RESUME -

L'exposé comprendra une description du banc d'essai C3 du CEPr. Cette installation permet d'essayer de grands compresseurs double flux, dont la puissance peut atteindre 40 000 kW. La présentation portera sur les méthodes de mesure (peignages par sondes mobiles, mesures de débit, mesures sur les parties tournantes), et sur les méthodes d'essais et l'utilisation de l'ordinateur pour obtenir les champs compresseur des deux flux, en particulier à proximité de la ligne de pompage.

1 - INTRODUCTION.

L'examen des différents programmes de développement et de recherche en cours ou prévus, et en particulier le programme CFM 56, on conduit en 1973 à la décision de réaliser un banc compresseur de grande puissance. Compte tenu de l'existence au C.E.Pr d'un ensemble de turbines de forte puissance, le banc C3 a été implanté au bout de la ligne d'arbres correspondante, à partir de Mai 1975. La première rotation avec un compresseur Basse Pression de CFM 56 a eu lieu le 29 octobre 1979. Le banc C3 permet l'établissement des champs de caractéristiques complets du compresseur en essai, avec exploration des zones de flottement et détermination des lignes de pompage.

2 - DESCRIPTION DU BANC D'ESSAI C3 DU C.E.Pr.

Les principales caractéristiques du banc C3 sont présentées dans la figure 1. Le banc C3 permet d'essayer des compresseurs monocorps à double flux jusqu'à une puissance de 40 000 kW et une vitesse de rotation de 12 500 tr/mn.

Le circuit d'air à l'admission est dimensionné pour un débit de 500 kg/s, une pression comprise entre 0,1 et 2 bars et une température comprise entre - 35 et + 100°C.

La volute Basse Pression qui recueille le flux secondaire sortant du compresseur est dimensionnée pour un débit de 500 kg/s, une pression comprise entre 0,1 et 4 bars et une température comprise entre - 30 et + 300°C.

La volute Haute Pression qui recueille le flux primaire sortant du compresseur est dimensionnée pour un débit de 200 kg/s, une pression comprise entre 0,1 et 12 bars et une température comprise entre - 30 et + 500°C.

La figure 2 montre une vue en élévation du coeur de banc qui est ancré sur une table en béton reposant sur des supports élastiques.

2.1.- Entrainement mécanique.

La puissance nécessaire à l'entrainement du compresseur en essai est fournie par une turbine à vapeur accouplée à une turbine à air. La turbine à vapeur peut fournir une puissance de 25 000 kW et la turbine à air peut fournir une puissance de 18 000 kW. Un accouplement débrayable permet de faire fonctionner les turbines sans entrainer le compresseur.

Un inverseur rend les deux sens de rotation possibles. Un multiplicateur permet de choisir la gamme de vitesse adaptée au compresseur en essai en changeant le rapport de multiplication.

Le compresseur est entrainé par l'arrière par une ligne d'arbres qui traverse les volutes. La ligne d'arbres est composée de trois arbres reposant chacun sur deux paliers et accouplés deux à deux par des accouplements flexibles.

2.2.- Circuits d'air.

La figure 3 présente le schéma des circuits d'air. L'admission et l'échappement du banc se font actuellement à l'atmosphère. Dans un stade ultérieur, l'installation qui est dimensionnée pour permettre de simuler les conditions réelles de vol pourra être raccordée aux circuits de conditionnement d'air du Centre.

Le circuit d'admission comprend de l'amont vers l'aval :

- un silencieux d'admission avec trois vannes permettant le réglage de la pression d'admission,
- une tuyauterie droite de 3,20 m de diamètre et 17 m de longueur,
- un divergent avec à la sortie un régulateur en nid d'abeille et un filtre,
- un caisson de tranquillisation de 6 m de diamètre et 7,5 m de longueur,
- un pavillon et un conduit d'alimentation devant le compresseur en essai.

Derrière le compresseur, deux volutes recueillent indépendamment l'air de chacun des deux flux.

Le système de vannage Basse Pression (figure 4) comprend douze soupapes commandées par des vérins hydrauliques. Deux soupapes sont spécialisées dans la fonction "antipompage", huit dans la fonction réglage "gros débit", une dans la fonction "petit débit" et une dans la fonction "très petit débit".

Le système de vannage Haute Pression (figure 5) comprend douze vannes disposées en couronne dans la volute HP. Chaque vanne comprend un volet entrainé en rotation par un vérin.

Les circuits d'échappement HP et BP comprennent chacun une conduite droite qui se sépare en deux canalisations équipées de :

- un ensemble régulateur composé d'un filtre amont, d'un régulateur en nid d'abeille et d'un filtre aval,
- un venturi de mesure de débit,
- une vanne de sectionnement.

Les quatre tuyauteries aboutissent dans un collecteur et la mise à l'air libre se fait par un silencieux d'échappement situé à 54,5 m de l'axe du banc.

2.3.- Servitudes.

2.3.1.- Graissage du compresseur.

La centrale de graissage du compresseur comprend :

- un circuit de lubrification des paliers, des joints carbone et de la butée du compresseur,
- un circuit d'épuisement des carters et des paliers,
- un circuit de dégazage,
- un circuit d'alimentation du vérin hydraulique qui exerce une contre poussée asservie à la mesure de la charge axiale sur le roulement de butée du compresseur,
- un circuit de remplissage du réservoir par un réservoir auxiliaire.

2.3.2.- Graissage de l'inverseur, du multiplicateur et de la ligne d'arbres.

La centrale de graissage correspondante assure :

- la lubrification des paliers et dentures avec un débit compris entre 5050 et 6550 l/mn,
- le réchauffage, la filtration et la vidange de l'huile,
- la purification par centrifugation.

La centrale est munie d'un échangeur huile / eau.

Les pompes sont doublées pour des raisons de sécurité et en cas de panne électrique le graissage est assuré par la turbopompe qui reste alimentée en vapeur.

2.3.3.- Centrale de refroidissement.

Cette centrale assure le refoidissement des éléments mécaniques en contact avec les volutes :

- le système de barres et rotules qui relient et soutiennent les volutes,
- la motorisation des vannes HP,
- le carter de la ligne d'arbres qui est protégé par un écran thermique.

Ces éléments sont refroidis par circulation d'eau déminéralisée en circuit fermé.

3 - MESURES AU BANC C3.

3.1.- Mesures de surveillance du banc.

Tous les éléments du banc C3 sont équipés d'une instrumentation de surveillance importante. La surveillance porte essentiellement sur l'entrainement mécanique du compresseur d'une part, et sur les circuits de servitude d'autre part. Les valeurs des différents paramètres mesurés (pressions, températures, vibrations) sont indiquées par des appareils de mesure analogiques dans la salle de contrôle.

Des seuils d'alarme sont définis pour tout les paramètres importants, suivant la hiérarchie suivante :

alarme blanche → défaut mineur, évolution à surveiller
alarme jaune → défaut nécessitant une décélération rapide par action manuelle du pilote
alarme rouge → déclanchement automatique - la turbine à vapeur n'est plus alimentée et le banc s'arrête.

3.2.- Mesures sur le compresseur.

Les mesures effectuées sur le compresseur sont acquises par un ordinateur MITRA 15 propre au banc. La capacité de mesure du banc est de 2 000 mesures.

L'acquisition des mesures de pression est réalisée soit par des scanivalves, soit par des capteurs individuels pour les mesures acquises lors du pompage.

Les mesures de température se font par des thermocouples Chromel - Alumel.

Des mesures de vibrations sont effectuées sur les différents paliers du compresseur.

Lors des essais du compresseur BP CFM 56 une télémesure est montée dans le nez du compresseur. Cette télémesure transmet à une antenne fixe placée à l'extérieur du compresseur

les mesures effectuées sur les parties tournantes, essentiellement des contraintes sur les aubages.

Des sondes mobiles de pression ou de température télécommandées à distance peuvent être utilisées pour explorer certains plans de mesure du compresseur. Ces sondes peuvent être déplacées à la fois en profondeur et en orientation angulaire.

Des mesures par fil chaud sont effectuées derrière la roue fan du compresseur et servent comme moyen de détection du pompage.

Des mesures de pression instationnaire avec des capteurs de type kulite peuvent également être faites.

Des moyens plus complexes de mesure tels que l'utilisation du laser pourront éventuellement être implantés dans le banc dans un stade ultérieur.

3.3.- Mesures de débit.

La mesure du débit d'air entrant dans le compresseur est effectuée en utilisant les résultats d'un étalonnage préalable de la manche d'entrée du compresseur.

La mesure du débit est également faite au niveau des quatre venturis à l'échappement du banc.

La figure 6 montre l'équipement de mesure d'un venturi.

Un peigne de 11 prises de pression totale permet de déterminer le profil de la couche limite et de calculer un coefficient de débit.

Pour mesurer la pression différentielle au col du venturi on utilise deux capteurs différentiels de gamme différente (7 kPa et 17,5 kPa). Jusqu'à 6 kPa on utilise le capteur de gamme 7 kPa et au dessus de 6 kPa on utilise le capteur de gamme 17,5 kPa. Ceci permet d'améliorer la précision de la mesure. Des pressostats permettent une fermeture automatique des circuits par des électro-robinets en cas de surpression.

La pression totale et la température totale sont mesurées en amont du col du venturi. La température de paroi est également mesurée au col pour tenir compte des variations de section dues à la dilatation.

Les sections des quatre venturis sont échelonnées du gros venturi du flux secondaire au petit venturi du flux primaire. Selon le débit d'air correspondant il est possible de sélectionner pour chacun des flux soit le petit venturi, soit le gros venturi, soit l'ensemble des deux venturis. Le nombre de Mach au col des venturis ne doit pas être trop faible car la mesure du débit devient alors trop imprécise; pour limiter les pertes de charge on ne dépasse pas un nombre de Mach au col de l'ordre de 0,5.

La comparaison entre le débit mesuré à l'entrée du compresseur et la somme des débits mesurés par les venturis est safisfaisante.

4 - METHODES D'ESSAI.

Tous les calculs sont effectués en temps réel par l'ordinateur MITRA 15 du banc.

Trois modes de traitement différents sont utilisés lors d'un essai : le mode pilotage pendant les phases d'accélération ou de décélération, le mode champ pour la détermination des isovitesses et le mode pompage à proximité du pompage. Le choix de ces différents modes ainsi que les différentes options d'édition ou de tracé se fait par l'intermédiaire de roues codeuses à proximité du bouton d'appel du point de mesure.

4.1.- Mode pilotage.

Ce mode a pour but d'indiquer rapidement au pilote les différents paramètres dont il a besoin pour suivre l'essai.

L'acquisition ne porte que sur un nombre réduit de paramètres (en particulier seules les pressions sur capteurs individuels sont acquises).

Les paramètres réduits du compresseur (débits réduits, rapports de pression, vitesse de rotation réduite, rendements) sont calculés et affichés sur des écrans de télévision avec quelques paramètres de surveillance propres au compresseur (pressions d'enceinte, températures de paliers, etc...).

4.2.- Mode champ.

Ce mode a pour but de caractériser les performances aérodynamiques du compresseur et permet en particulier la détermination des isovitesses dans les champs compresseur du flux primaire et du flux secondaire.

L'acquisition porte sur la totalité des mesures. Les moyennes sont calculées par peigne et par anneau dans les différents plans de mesure.

Le calcul des différents paramètres réduits est effectué en temps réel et les résultats sont affichés sur les écrans de télévision et imprimés sur une imprimante rapide.

L'ensemble des acquisitions, des moyennes et des résultats de calcul est enregistré sur un disque magnétique.

Un pointé sur table traçante de la position du point d'essai dans le champ compresseur du flux primaire ou du flux secondaire peut être effectué.

4.3.- Mode pompage.

Ce mode a pour but la détermination précise des lignes de pompage dans les champs compresseur des deux flux. Les phénomènes étant instationnaires à proximité du pompage il est nécessaire d'effectuer une acquisition rapide à 10 Hz d'un nombre réduit de paramètres (environ 50). Cette acquisition est stockée dans une mémoire tournante qui ne conserve que les 60 dernières secondes.

A partir du dernier point champ effectué, ou se déplace sur une isovitesse en fermant légèrement les vannes à l'échappement du flux concerné, jusqu'à l'apparition du pompage détecté entre autre par le signal du fil chaud (figure 7). On ouvre alors les vannes d'antipompage et on arrête la scrutation du mode pompage.

On effectue ensuite le tracé sur table traçante de l'un des paramètres acquis au pompage (en général le fil chaud).

La courbe obtenue (figure 8) permet de déterminer de façon précise l'instant où le pompage est apparu. Cet instant est ensuite introduit dans l'ordinateur par l'intermédiaire d'un clavier.

Les moyennes qui sont utilisées dans les calculs sont alors recalculées à partir des moyennes correspondantes du dernier point champ dans le rapport des paramètres acquis au pompage rapportés aux valeurs correspondantes du dernier point champ.

Le même programme de calcul que pour le mode champ est ensuite utilisé pour calculer les différents paramètres réduits.

4.4.- Mode répartition.

Ce mode consiste à explorer un rayon du compresseur et à déterminer en différents points de ce rayon l'angle de l'écoulement, la pression totale, la pression statique et la température totale.

L'opération se fait en deux temps :

- à l'aide d'une sonde à trois becs on repère l'angle de l'écoulement et on mesure la pression totale pour différentes valeurs du rayon;
- après changement de sonde, on mesure la pression statique ou la température totale en orientant la sonde face à l'écoulement conformément aux relevés précédents.

Il est prévu que le déplacement des sondes en profondeur et en angle soit effectué de façon automatique.

5 - CONCLUSION.

Grâce à sa puissance qui peut atteindre 40 000 kW, le banc d'essai C3 permet d'effectuer les essais des grands compresseurs des moteurs civils actuels tels que le CFM 56 ou le CF6-32. Les essais de tels compresseurs offrent la possibilité d'explorer des zones de fonctionnement dangereuses pour le moteur complet (pompage, flottement) et permettent une indépendance du développement du compresseur par rapport aux autres parties du moteur.

PUISSANCE	40 000 kW	
VITESSE DE ROTATION	12 500 tr/mn	
ADMISSION	débit	500 kg/s
	pression	0,1 à 2 bars
	température	- 35 à + 100°C
ECHAPPEMENT BP	débit	500 kg/s
	pression	0,1 à 4 bars
	température	- 30 à + 300°C
ECHAPPEMENT HP	débit	200 kg/s
	pression	0,1 à 12 bars
	température	- 30 à + 500°C

FIGURE 1 PRINCIPALES CARACTERISTIQUES DU BANC C3

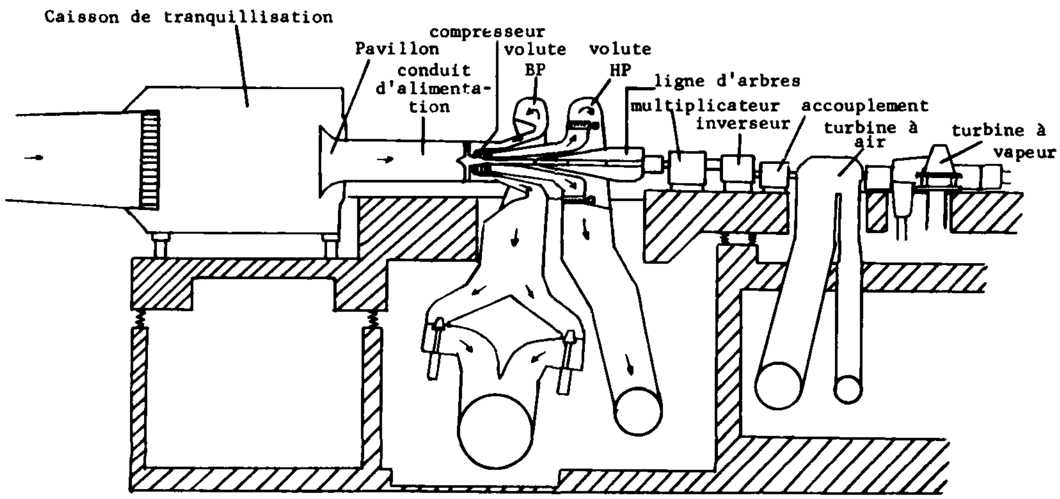

FIGURE 2 VUE EN ELEVATION DU COEUR DE BANC

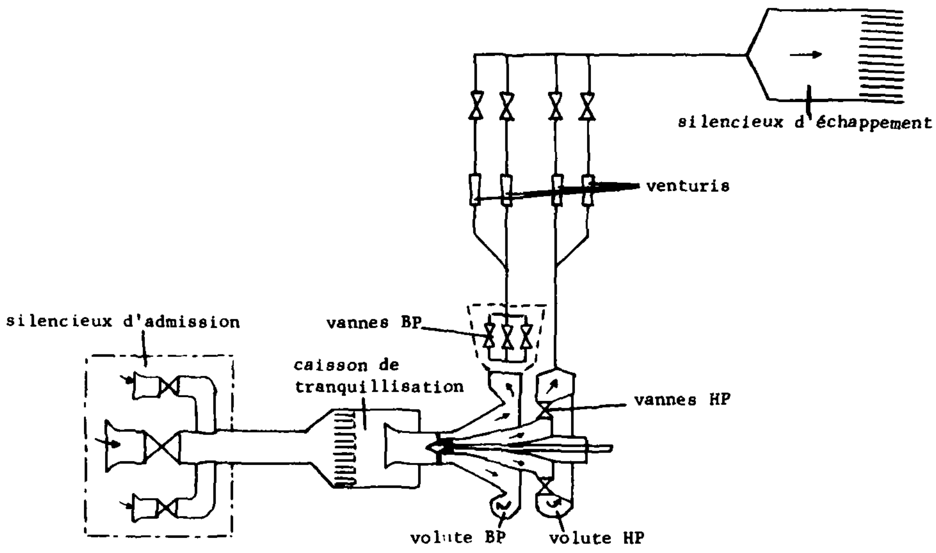

FIGURE 3 SCHEMA DES CIRCUITS D'AIR

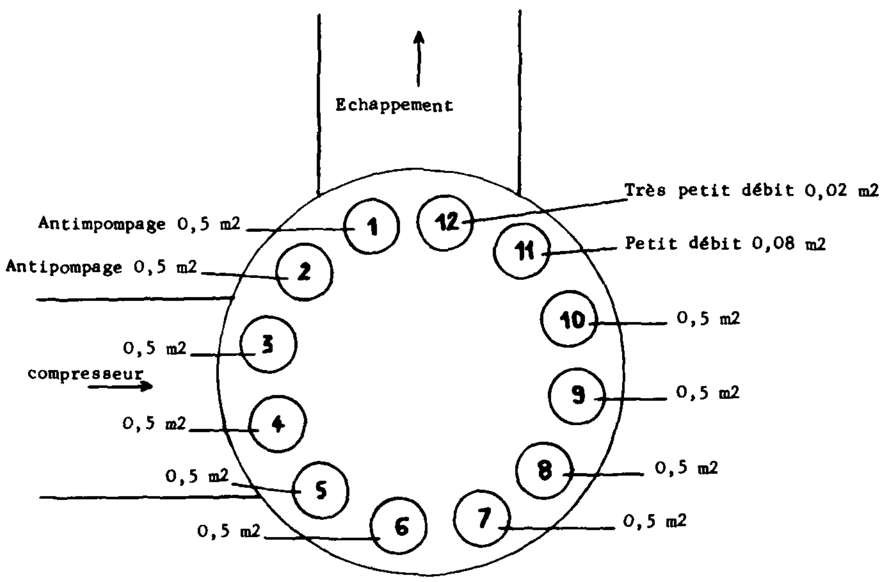

FIGURE 4 SYSTEME DE VANNAGE BP

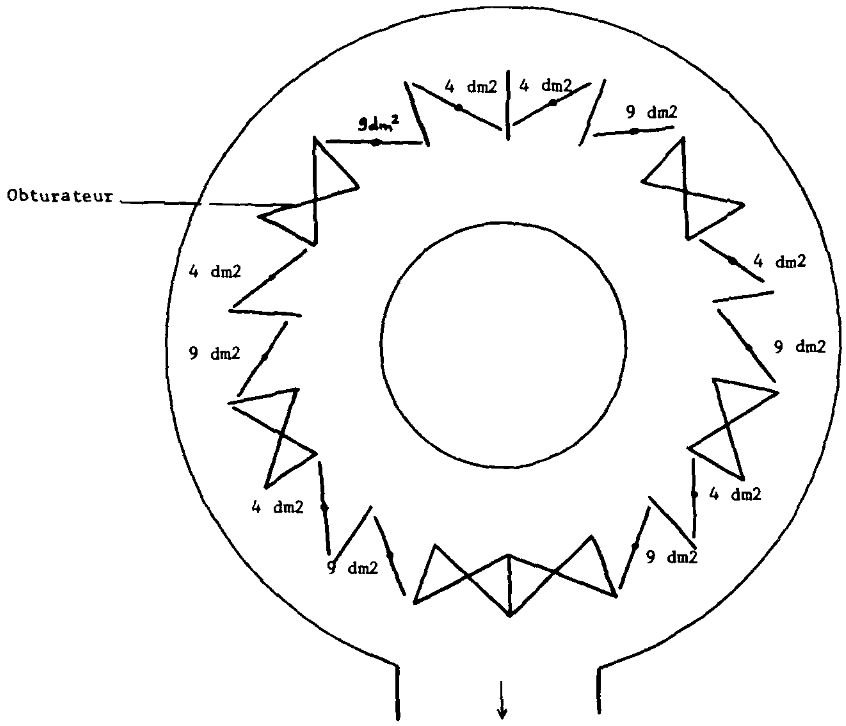

FIGURE 5 SYSTEME DE VANNAGE HP

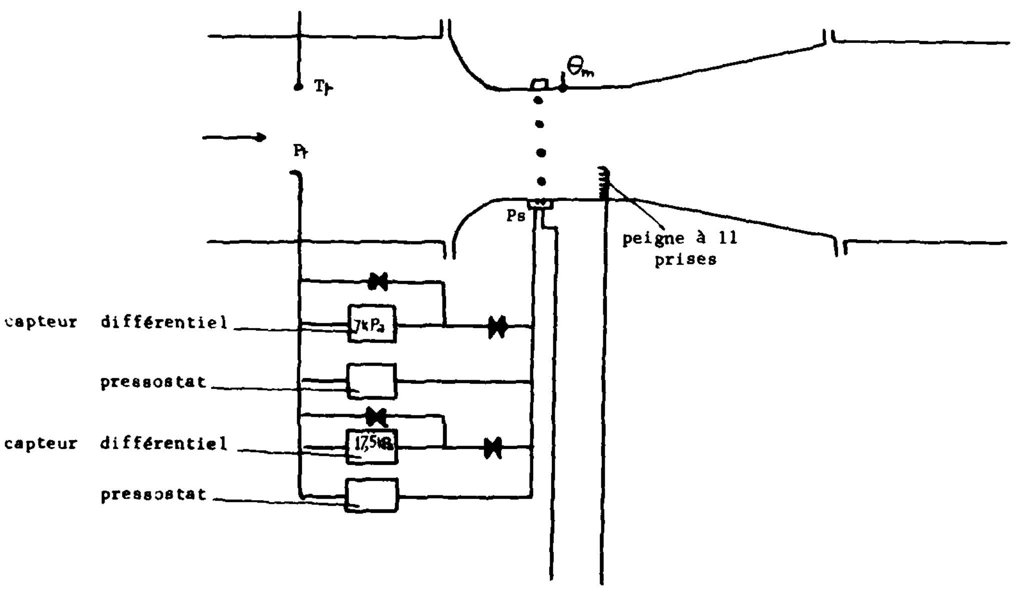

FIGURE 6 EQUIPEMENT DE MESURE D'UN VENTURI

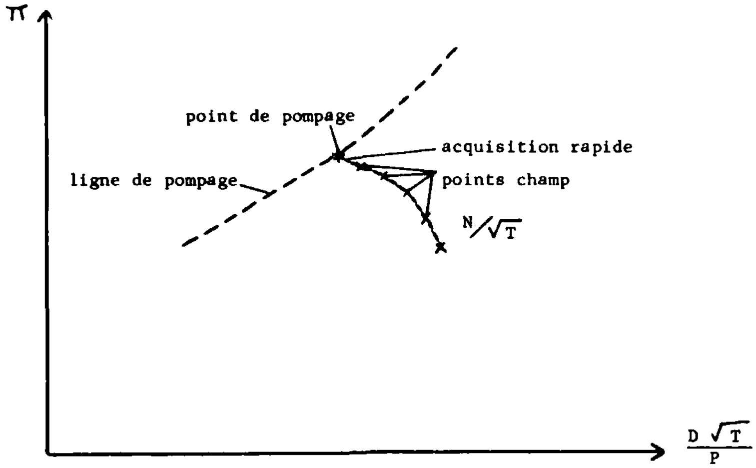

FIGURE 7 DETERMINATION D'UNE ISOVITESSE JUSQU'AU POMPAGE

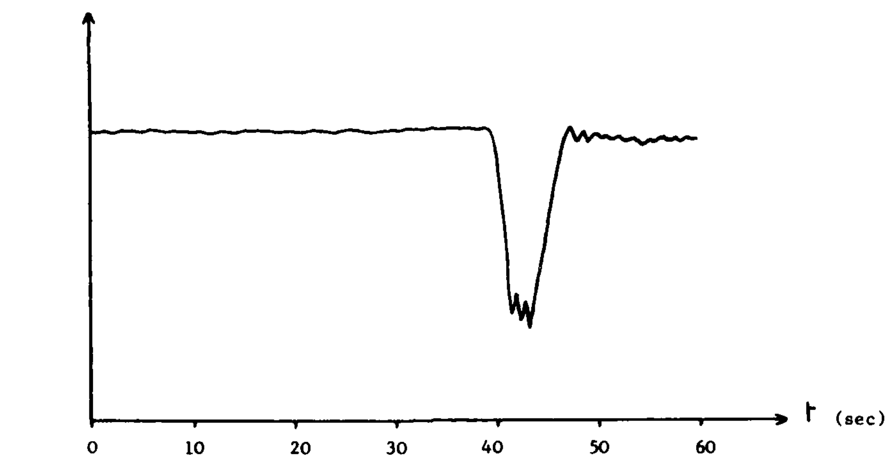

FIGURE 8 COURBE DE REPONSE DU FIL CHAUD AU POMPAGE

FULL ANNULAR COMBUSTOR TEST FACILITY FOR HIGH PRESSURE/ HIGH TEMPERATURE TESTING

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1. SUMMARY

A special combustor test facility has been built at DFVLR Research and Test Centre PORZ WAHN which is described in detail. On this facility, research on annular combustors has been carried out up to 16 bar and 820 K combustor inlet conditions. First of all the requirements for typical test procedures for high pressure/high temperature combustor testing have been established. After definition of these requirements the layouts for the test installations and the test bed were specified. As well as the design of special test equipment it was necessary to develop the testing and measuring techniques. Every effort has been made to achieve the most accurate high temperature measurement and gas analysis at the exit of the annular combustor. Results achieved by rig testing showed close similarity to those produced by engine testing.

2. INTRODUCTION

Compressor pressure ratios and turbine inlet temperatures of aircraft gasturbine engines have been increasing steadily. In both civil and military applications the trend is towards turbofan engines having compact high pressure, high temperature gas generators. Fuel efficient civil engines now in the development stage have overall compression ratios of up to 40. The latest military engines have turbine inlet mean temperatures exceeding 1700 K. In cycle and mission analysis of these modern compact engines it was assumed that materials, combustors and turbine cooling systems can be designed and developed to work satisfactorily. High overall compression ratios are followed consequently by high compressor discharge air temperature. As this air is the coolant for the flame tube and the HP-turbine, it offers only a low heat sink capacity. At the same time the cooling requirement will increase because of high gas temperature in the flame tube and high heat fluxes to the engine parts exposed to the gas. Thus, with increasing pressure the proportion of cooling air increases, i.e. for flame tube cooling it reaches more than 40 % of the combustion air. Therefore, as the amount of combustion air which is needed for a stoichiometric mixture in the primary zone takes nearly 30 %, the remainder for the dilution zone is very small. Intensive combustor development is therefore necessary in order to arrive at the temperature pattern factors required to guarantee adequate turbine life.

Progress has been made in designing combustors and calculating the aerodynamic/ chemical processes in a combustor (1)[1]. These calculations relate the chemical reactions according to the local conditions and mixture ratios to the aerodynamic flow field in the combustor. Fuel vaporization calculation and better understanding of the recirculation model of the primary zone permit determination of flame stability limits, combustion process and gas emission specimen.

In addition to the theoretical work a great amount of detailed model rig testing has been performed to generate well established data of the aerodynamic and thermodynamic behaviour of combustor components, such as diffusor and flame tube cooling devices. All these data are very useful for the proper layout of a combustor, however the complex effect of aerodynamic and thermal load on the combustor can only be shown in tests under actual engine conditions.

The simulation of these engine conditions requires a specially designed test site and rig. The planning objective for the combustor test site and combustor rig has to follow the development requirements in the engine programme. At first, a research combustor rig and subsequently an engine parts rig (using engine parts) was designed for high pressure and high temperature operation.

Exact simulation of all actual engine operating conditions was the ultimate goal. However, technical and economical considerations led to some compromises. As durability problems of high temperature loaded components dictate TBO and life time of an engine it was the main goal to simulate the influence of gas temperature on combustion process, pattern factors and flame tube wall temperatures.

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1) Numbers in parentheses designate References at end of paper.

3. MATCHING COMBUSTOR TEST SITE CAPABILITIES TO COMBUSTOR REQUIREMENTS

At DFVLR Research and Test Centre PORZ WAHN, the existing compressor capacity for the required air massflow is 20 bar. By installing a new air heater the combustor rig inlet temperature could be increased to 850 K. The measuring equipment, i.e. temperature/gas sampling probes, had to be designed to withstand peak temperatures up to 2000 K. A special traverse gear for probe rotation at the combustor exit and the exhaust ducting including the throttle valve had to be designed to stand mean gas temperatures up to 1750 K.

3.1 Air supplies

The existing test facilities are supplied by a common compressor station consisting of low pressure (π = 5) and medium pressure (π = 4) compressors. They can be coupled in parallel or in series. Optionally, five HP-vessels of 1000 m^3 total volume can be filled at night by a separate HP-compressor (60 bar) to give some 70 000 kg HP-air in addition to the compressor air for testing purposes. The combustor test procedures have been chosen in such a way that for maximum air demand by use of vessel air the electrical power for the compressors is limited to approx. 11 Megawatt for economical reasons.

A simplified scheme of the combustion test site is shown in fig. 1. The air from the central compressor station is fed to the combustor test site by a piping system which contains two metering orifices (0,2 - 5/1,5 - 30 kg/s) for accurate measurement of the combustion air flow rate. The metered airstream is fed to the air heater and then passes through a 30 m long, well insulated piping system to the test site. This arrangement had to be chosen because of local restrictions in heater location and it unfortunately results in a maximum thermal piping expansion of 360 mm and in a temperature loss of nearly 75 K. For special cooling requirements, there is a separate metered high pressure air line into the test cell.

3.2 Combustor Air Heater

Exact duplication of all engine operating conditions concerning combustor inlet temperature was the main goal. Near maximum temperature and pressure at engine HP-compressor discharge occur during high speed low level flights. As the aircraft then needs maximum thrust the combustor must also be capable of maximum heat release under these conditions. All test and measurement equipment was designed for these conditions. The design criterion for the air heater was to heat the maximum test facility airflow of 20 kg/s up to 820 K at combustor inlet. In accordance with this and the 75 K temperature loss the maximum temperature rise of the heater was set at 640 K. This temperature rise with a desired time stability of $\pm$ 2 K and the most economical design led to the chosen convection type heater.

The temperature rise of the heater at 1 bar is 400 K to 550 K. Above 2 bar any required temperature rise between 200 K and 550 K can be set up. A heating up time of less than 40 minutes and a TBO of 300 hours was guaranteed and could be proven. The heater, see fig. 2, consists of an oil-fired combustion chamber and a folded gas pass to the heat exchanger. To alter the maximum 1300 K high gas temperature, cold dilution air can be mixed gradually with the hot gas before entering the heat exchanger. The heater is controlled automatically by a three stage regulation system. For big changes of heat flux the amount of burnt fuel and hence the hot gas temperature is influenced, small changes are controlled by the dilution air / hot gas mixing ratio. For accurate temperature regulation a cold air bypass of the heater enables a temperature stability of $\pm$ 3 K.

3.3 Test Site Supplies

Fuel supply

The fuel supply system was designed for the development of combustors with different fuel spray systems. Therefore a three line system was chosen. Pressures up to 100 bar and flow rates up to 3000 l/h can be covered. Metered backing lines allow for presetting of required flow rates. For quick and accurate measurement three different flow rate measuring systems have been chosen. An accuracy of $\pm$ 0,2 % o.r. is guaranteed by an automatic volumetric system. The impulses from the calibrated flow turbines are fed to the automatic data acquisition system.

For special endurance tests with cycling combustor heat releases, the fuel can be automatically chopped from a chosen maximum to a minimum flow rate with individual time settings. For investigation of elevated fuel temperatures on fuel-boiling behaviour in manifold and spray systems, the fuel temperature can be increased to 430 K.

Cooling Water Systems

The existing main cooling water system is made up with a boost pump (10 bar) and then used to cool all combustor exhaust ducting. Each separate line is flow rate-controlled according to water exit temperature to restrict the overall amount of cooling water to an allowable rate of 160 m^3/h. From this medium pressure water system an injection water system is fed by another boost pump (35 bar) which injects water into the hot ga downstream of the combustor. A regulation system which meters the flow rate in relation to the amount of burnt fuel keeps the water loss to a minimum. A fuel/water ratio of 1:5 has proven to be satisfactory and keeps the steam plume at the exhaust stack nearly invisible.

For cooling of the thermally high loaded probes at combustor exit a separate closed-loop decarbonised water system was designed. The pressure of this system is adjusted by a bypass regulation system to keep the pressure difference between water and gas to a minimum. This method and a safety shut-off system prevent burn out of all probes in case of damage to one single probe. By additionally monitoring water exit temperature at each probe, damage has so far been completely prevented.

3.4 Combustor Test Rig Arrangement

The best method of handling thermal piping expansions between air heater and combustor rig and also to take up maximum axial loads of 250 kN from the throttle valve, was to install the whole rig and its complete measuring equipment on a moveable rig dolly, see fig. 3. Hot combustion air enters the rig at the front support which can be disconnected for combustor mounting purposes, hydraulically sliding on the basic frame. Front support, basic frame and rear support take the high axial loads from the rig. The rear support carries the exhaust throttle vale. Thermal expansions between front and rear support are taken by a compensator upstream of the combustor. To prevent any buckling force on a thin walled (1,6 mm) engine combustor casing, the compensator can be installed with pre-tension. This arrangement first seemed to be very complex but it has proved to be a safe and easy mounting device. Only 4 mechanics and 3 engineers are necessary to fulfill a comprehensive test programme with 3 different combustors within a week, having a 10 hour day shift. Up to now more than 650 HP running hours have been accumulated without great difficulties or damage to the equipment.

4. TEST LINE

4.1 Combustor Test Rig

The research Combustor Test Rig is shown in fig. 4. It consists of the inlet section with its measurement instrumentation, a two row flow simulator, outer/inner casing and the annular flame tube. By means of a grid, by variable guide vanes and by blockage rings the required compressor discharge velocity profile and turbulance level can be generated. The section downstream of the flow simulator contains an interchangeable prediffuser. The dump diffuser divides the annulus air into approximately equal proportions for outer and inner flame tube supply. At the end of the outer annulus, HP-turbine bleed is simulated by a metered blow off line. The combustion outer casing carries 11 fuel nozzles and two high energy torch igniters. Different up-to-date fuel spray systems have been investigated, such as atomizers and vaporizers. Best results have been achieved using a fuel prevaporization concept, (2), (3). The flame tube consists of sections welded together using highly efficient machined cooling rings. At the exit of the flame tube, in the plane of the nozzle guide vane (NGV), measurement of gas temperatures and gas sampling are performed. To ensure that during testing the probes at combustor exit (6 off) are well centered in the hot combustor exit plane, they have to be adjusted to give approximately 3,5 mm offset.

4.2 Traverse Gear

For reasons of simplicity most of HP/HT Combustor development work is usually done by using either a number of fixed probes or integral well mixed gas values downstream of the combustor, (4), (5). At NASA-Lewis Research Centre a newly designed test facility for HP-combustor and HP-turbine development uses an internal rotating assembly (6). A similar design was chosen by Rolls Royce Bristol. Problems with "internal" traverse gears are often caused by the hot gas environment of the measuring wiring and the mechanism for rotation and control.

After a design evaluation in 1972 it was decided to build a traverse gear with "external" connection of the probes, see fig. 5. External connections permit easy checking of all instrumentation wiring, easy tubing and individual adjustment for radial probe positioning. The chosen traverse gear design consists of a fixed front member, a T-Shaped rotating ring for support of 6 equispaced probes, a hydraulic drive system, a control unit and a fixed rear member. To take off any axial load from the T-ring, a superstructure of 8 bridge pieces is built over it. The bridge pieces take the axial loads generated by the high internal pressure. To allow for the thermal and elastic expansion of the superstructure only one single gear carries the rotating T-ring. All fixed and moving parts are water cooled. By means of a system of piston type rings and HP sealing air chambers any hot gas leakage to the atmosphere can be prevented. Careful attention is paid to balancing the pressure for sealing air in the upstream and downstream annular HP air sealing chamber. On the outer diameter of the T-ring all wires and tubes are run together to form a common flexible harness to wind up during rotation of a full sweep of 120^{o}, see fig. 6. The pictures give a view onto the front member of the traverse gear and the exhaust duct downstream of the combustor exit. The cruciform water cooled pipe structure centers the inner water cooled linings and contains a water injection nozzle on the centre line.

4.3 Exhaust Throttle Valve

The design principle of the internally watercooled exhaust throttle valve (butterfly type) is that a hollow and rigid internal structure takes the load from the HP-gas pressure, see fig. 7. A separate water jacket enables the construction of narrow passages for intensive water cooling. The water jacket with its elevated temperature and non uniform thermal expansion can move freely on the internal body. Therefore a multisealing design at both ends of the axis was necessary to prevent leakage of cooling water into the gas or rather HP hot gas penetration into the 10 bar cooling water system, depending on actual gas pressure. For ease of installation the single piece valve is mounted between two flanges of the exhaust ducting.

5. INSTRUMENTATION AND MEASUREMENT EQUIPMENT

Combustors of modern aeroengines are highly loaded, i.e. they are required to produce a large heat release in a small volume. It is therefore important to prove their performance which must approach 100 percent burn-out rate of the injected fuel together with a reasonable temperature distribution to ensure a definite HP turbine life. Therefore the technology of temperature measurement and gas analysis for high pressure and high temperature gas had to be developed at MTU in order to enable combustor development.

5.1 Temperature measurement

Probe Design

Thermocouples of Pt Rh/Pt are usually used for high temperature measurement. Using the combination PtRh10/Pt, the thermocouple beads stand temperatures up to 2000 K (melting point). The problem of temperature measurement is twofold: as a probe body is required which will stand these high temperatures, it has to be water-cooled intensively, and hence the heat flux from the thermocouple bead to this heat sink causes the bead temperature to be somewhat lower than the actual gas temperature.

For easy measurement of the whole combustor exit, three rakes, each incorporating five open beads, were found to be sufficient for accurate temperature field determination. The traverse gear has then only to rotate through a 120 degree sector. The following criteria have been established for the temperature probe design:

- durability of probe body and beads up to 2000 K at 20 bar
- easy maintenance or replacement of the thermocouples
- foolproof connection systems to prevent mixup of bead wiring and faulty contacts
- good reproducibility of geometrical bead configuration
- good response of thermocouple bead
- probe design to measure gas temperature as near as possible to the wall
- no catalytic effect on bead surface in case of unburnt products in the gas.

As a result of continuous development we came up with a hook-like probe body, see fig. 8. The head is of slender form held by a 22 mm diameter shaft. The cooling water impinges on the hot leading surface of the probe and is drained along the inner structure. Five separate welded in tubes enclose the teflon insulated 0,5 mm diameter thermocouple leads which end in ceramic tubes at the probe head. These ceramic tubes of 2.8 mm diameter contain four holes, two of them for PtRh10 support wires. A four-wire bead design was found necessary for reasons of durability and to keep the bead centered for all thermal expansion positions. To prevent catalytic postreaction the beads are Al_2O_3 coated with a layer of 6 - 8 μm thickness.

Temperature Correction

Open bead temperature rakes have been used first during atmospheric combustor test periods. Combustion efficiency according to thermocouple mV-reading was always too low compared to the gas analysis efficiency. Therefore, it was corrected in relation to gas efficiency. For the purpose of comparison, hot gas calibration tests of the probes have then been performed. A definite influence of hot gas Mach-number, gas temperature and bead configuration on the error between bead and actual gas temperature was found. Starting the HP combustor test series, the correlation of this atmospheric calibration leads to pyrometric efficiencies above 100 %. In the first instance this efficiency could only be corrected in relation to the corresponding gas analysis efficiency. This method was unsatisfactory. Therefore, some effort was put into probe improvement together with some analytical heat balance calculations of the standardized bead configuration.

In this heat balance model the bead has to perform a heat balance considering forced convection, conduction and radiation. Any catalytic effect could be neglected because of Al_2O_3 coating. A schematic of the thermal heat balance model is given in fig. 9. Parametric calculations showed a reasonable sensitivity of the computer model. By applying a temperature influenced emissivity of the Al_2O_3 coating a good coincidence of atmospherically calibrated values and theoretical data was obtained, as can be seen in fig. 10. Corrections by this calculated relationship are applied to each individual temperature reading, taking into account the typical test point data such as air pressure, air temperature, air/fuel ratio, combustor exit Mach-number, bead emissivity etc. In all atmospheric and HP/HT combustor test evaluations discrepancies of not more than 3 % (25 K) between pyrometric and gas analysis efficiency are obtained. According to this the peak gas temperature at combustor exit has an accuracy better than 1,5 percent, i.e. 1,3 percent accuracy of the temperature pattern factors OTDF and RTDF.

5.2 Gas Analysis

Gas analysis has been employed to determine the gaseous combustion products and smoke emission. The efficiency of the combustion process can be readily determined from the portion of total unburned hydrocarbons-UHC and Co. The concentration of smoke for example is mostly a measure for the mixing quality of the fuel-rich reaction in the primary zone. Combustor gas analysis at a higher pressure level than in normal atmospheric tests has the advantage of comparability with actual engine results. The problem however with rig results is to get a true and representative value for each specimen.

Gas Sampling System

High temperature and pressure might cause postreaction of the sample in the probe if fast quenching of the unburnt products and radicals cannot be achieved. In order to avoid too much investigation into developing sample techniques some theoretical calculations have been performed which showed that the lower the concentration of unburnt products, the lower the rate of postreaction, and hence the better the reliability of the determined combustion efficiency. For a typical 15 bar test point the accuracy of combustion efficiency is within a band of nearly 0,5 percent if the determined efficiency is better than 99 percent. As nitric oxides show practically no tendency to further reaction below 1500 K it could be proved by HP-rig test results, that the well known pressure law for NO formation $EI_{NO} \sim p^{0,5}$ could be adopted for our combustion systems. As it was also proved that the rig smoke results compared well with the engine exhaust values, we gained confidence in the used gas sampling technology.

For sampling gas emissions from HP combustor rig tests three five-point water cooled probes were used, see fig. 8. The width of the probe corresponds to the annular combustor exit plane height. The cooling water impinges on the hot leading surface of the probe and thus gives good cooling of the body and allows for fast quenching of the gas specimen in each of the five separate lines. After cooling down the gas, the five gas sampling lines are joined in a main duct and its mean gas temperature is monitored at probe exit. During the stepwise temperature traverse at the combustor exit gas samples are taken from three equispaced probes in order to obtain a full set of gas emission data. All lines to the instruments are heated, and the arrangement was built up corresponding to the EPA requirements as far as possible, see fig. 11. To minimize the response time of the system, the gas flow rate is kept as high as possible by venting the excess gas. Thus, a response time of 10 seconds plus 10 seconds for value stabilisation gave only a 20 second period for each test point sampling. All instruments operate continuously and the results are traced permanently by multi-channel chart recorders. Before and after each test point the instruments are calibrated with high accuracy reference gas. During start of the combustor and fuel-rich burning during test point set-up, contamination of the sampling system with liquid fuel or soot is prevented by blowing purge air through the lining system. Each probe can be connected separately to the instruments by means of automatic shut-off valves if non-mixed individual values from the circumference are desired.

Gas Analysis Equipment

A recommendation for the procedure for continuous sampling and measurement of gaseous emission from aircraft turbine engines is laid down in the "Aerospace Recommended Practice ARP 1256", (7). The equipment which was used corresponds to this standardising technology. The photograph of fig. 12 shows the set-up of the equipment.

- Total Unburnt Hydrocarbons - UHC

  The total unburnt hydrocarbons are measured with a flame ionization detector (Model FID 3, Testa, Germany). This special model is insensitive to oxygen and is completely heated to 150 °C. Because of a pressure stabilizer it has a stable read out even under inlet pressure variations. The flow rate is kept to 50 l/h and its measuring ranges are:

  0-10/100/1.000/10.000/100.000 ppm Vol

  For calibration a set of different gas concentrations (C_3H_8 in N_2) are used.

. Nitric Oxides - NO_x

The nitric oxides NO_x=No+No_2 are measured using a chemiluminescence analyser (Model 10 AR, Thermo Electron Corp., USA). This model has been chosen because of its good sensitivity resulting from the vacuum reaction chamber (25 Torr). The low pressure also prevents extensive cross-sensivity to water vapour. The lines and the capillary module are also heated to prevent NO_2 fall out in condensed water. The analyser including the capillary modūle is calibrated with NO and NO_2 mixtures in N_2 with the flow rate of 50 l/h, the same as for the measuring set-up. By passing the gas through a NO_2/NO converter the NO proportion of the NO_x value can be measured. Span setting ranges are:

0-2.5/10/25/100/250/1.000/2.500/10.000 ppm Vol NO_x

. Carbon Monoxide - CO, Carbon Dioxide - CO_2

These specimens are continuously measured by the usual nondispersive infra-red analyser (NDIR). The used model shows an excellent thermal stability and no long time drift as a result of its double stage cuvette system (Model UNDR 5N, Maihak, Germany). The line heating is set to approximately 150 ^{o}C and the gas is filtered, dried in a 2 ^{o}C water trap and fed to the instruments at a flow rate of 60 l/h. Span setting ranges are:

0-0.2/0.5/2.0	% Vol	CO
0-3/10/30	% Vol	CO_2

. Smoke (optical extinction method)

The smoke measuring procedure is laid down in ARP 1179 (8). This procedure involves the divertion of a small fraction of the engine exhaust stream which is passed through a standardized filter, the loss of its reflectivity being measured. Since this method was specially conceived for engine compliance tests it has some drawbacks regarding combustor gas measurement. The main disadvantage is its inability to carry out continuous smoke emission measurements. Therefore an improved optical smoke meter was designed at MTU which permits continuous measurement with sufficient accuracy at low smoke concentrations. Further details are given in (9). The optical instrument employs a chopped dual beam optical system. The light from the source is divided into a measuring beam through a steadily blown smoke chamber and a reference beam. Comparison of the two light intensities leads to the extinction value of the actual gas sample passing the heated smoke chamber.

5.3 Data Acquisition

A semi-automatic data collecting system was designed. Depending on the type of combustor test, different data sequences can be chosen. A basic programme which collects a fixed set of data for test point evaluation, is activated before and after each test point traverse. A schematic of the data acquisition system is given in fig. 13. On command from the control unit, the multiplexer digitizer scans all determined signal input channels. For pressure measurement transducers and scanivalves are used. For higher precision, pressures are taken as differences to a finally calibrated reference pressure, normally a static pressure at combustor inlet. Scanning velocity can be adapted to full pressure settlement in the lining system. Temperature accuracy in the case of combustor exit gas temperature measurement is mostly governed by the quality of the mV-reading/gas temperature correlation, not by the accuracy of the digitizer instruments.

The input signals are immediately emitted on a punched tape (BCD-Code) and fed into a teletyper for a printing of all data outputs in mV-units for control purpose. Fifteen channels can be traced on chart recorders parallel to the automatic digital system for control. During a 120 degree sweep, gas samples and temperatures are normally taken simultaneously. The step increment of the traverse gear can be selected by digital switches, but a 5 degree step is preferred. This mode gives 360 individual temperatures and 24 gas sampling values for a full traverse at combustor exit. Considering a data collection sequence time of 20 seconds for one step, one comes up with nearly 7 to 8 minutes for a complete set of data for a single test point.

6. TEST AND ANALYSIS PROCEDURES

6.1 Typical Test Procedure

After a series of model testing, atmospheric combustion testing and altitude relight testing the performance of the combustor under actual engine conditions has to be proved. These HP/HT-combustor tests should precede the engine testing. To achieve good compatibility not all of the dimensions of the rig combustor in relation to the engine combustor should be scaled and great care should be taken to ensure the simulation of the dominant engine characteristics.

The HP/HT-combustor test procedures have been established in such a manner, that at first the aerodynamic design values are proved and found applicable to the combustor even under high thermal loading. Special attention is then drawn to the primary zone performance (recirculation, mixing quality, flame tube wall temperature). By altering the air temperature and pressure and the air/fuel ratio as well, the optimum of combustor performance can be determined in a parametric manner. The second stage of development deals with the optimisation of the combustion process to meet the engine design requirements. The most important features are combustor durability and exit temperature pattern.

High pressure combustion tends towards smoke production, so a lot of development has to be put into smoke reducing flame tube modifications or burner design improvements. For a smoke reduction from emission values of 9 HSU to 4 HSU, for example, a development period of nearly two years was necessary and another period of two years brought the smoke value within 1,5 HSU well below the visibility limit of 2,5 HSU.

For this typical work accompanying an engine test programme a certain combustor test procedure was found to be the most economical, see fig. 14. In this HP/HT test procedure, measurements are first taken at part load conditions to prevent any damage to the combustor and the probes. In the second step full load is applied if the foregoing data allow for higher thermal load.

6.2 Data Processing and Test Evaluation

Comprehensive computer calculations are needed for data processing as each individual temperature reading has to be corrected to come up with the true gas temperature. As the combustion engineer not only desires to know mean values or global pattern factors but also temperature field plots, complex computer software was established. As a result one came up with temperature graphs, see fig. 15. On this graph lines of constant gas temperatures are plotted which enable an easy judgement of traverse quality.

The gas analysis results are gathered on multi-channel chart recorders. Scripts are evaluated manually. A typical chart is shown in fig. 16, it also contains the traverse gear stepping trace for better data correlation and evaluation. Automatic gas analysis evaluation during gas sampling is in preparation.

6.3 Comparison of Rig and Engine Test Results

As far as gas sampling results are concerned comparison of HP/HT combustor test and engine test data could be easily applied, as in the engine test programme mainly exhaust measurement equipment was installed. For analysis of engine test data it has to be born in mind that a certain deterioration of combustor exit values to turbine exit values occurs due to possible postreaction in the turbine section and due to mixing of combustor exhaust gas with turbine cooling air and internal sealing air. Temperature measurement by traverse of low pressure turbine exhaust gas showed a change in temperature profile shape and peak temperature location compared to combustor exit results. No congruence to combustor exit temperature profiles could be found. Gas analysis data however at engine nozzle discharge compare very well if a sample of the core jet stream is taken and corrected for dilution by internal bleed and turbine cooling air, see fig. 17. The measured emission values of unburned products (UHC, Co) decrease in the well known manner with increasing pressure and temperature in the combustor, i.e. shaft rotation or thrust of the engine. The formation of NO_x and the production of smoke both show a tendency towards increase, the NO_x formation corresponding to the rising gas temperature, and the smoke production corresponding to the increasing fuel richness and gas pressure. For reasons of comparison rig results are plotted into the engine emission curves. From this good agreement of gas analysis data it may be assumed that a similar good agreement exists for combustor exit temperature pattern factors at combustor exit. This could be indirectly proved after thermal paint tests with turbine nozzle guide vanes (NGV) fitted to the combustor rig and NGV tests in the engine.

7. CONCLUSIONS

Great effort has been put into the design and development of a high pressure/high temperature combustor test site and combustor rig. As well as the design of special test equipment, an intensive development programme for instrumentation equipment was necessary which was accompanied by theoretical layout calculations. For measurement of combustor rig and engine emissions recommended practices and gas analysis equipment led to good and comparable results. For fast response and reproducible smoke emission measurements with high sensitivity, two different instruments have been designed and proved to work satisfactorily. In accordance with these proven engine-to-rig correlations it is possible to develop an engine combustor to its full performance in a rig programme. Consequently not only time and cost of engine development work can be saved but also a better understanding of the combustion process is accomplished. In long term development risks can be minimised through proper combustor lay out.

ACKNOWLEDGEMENT

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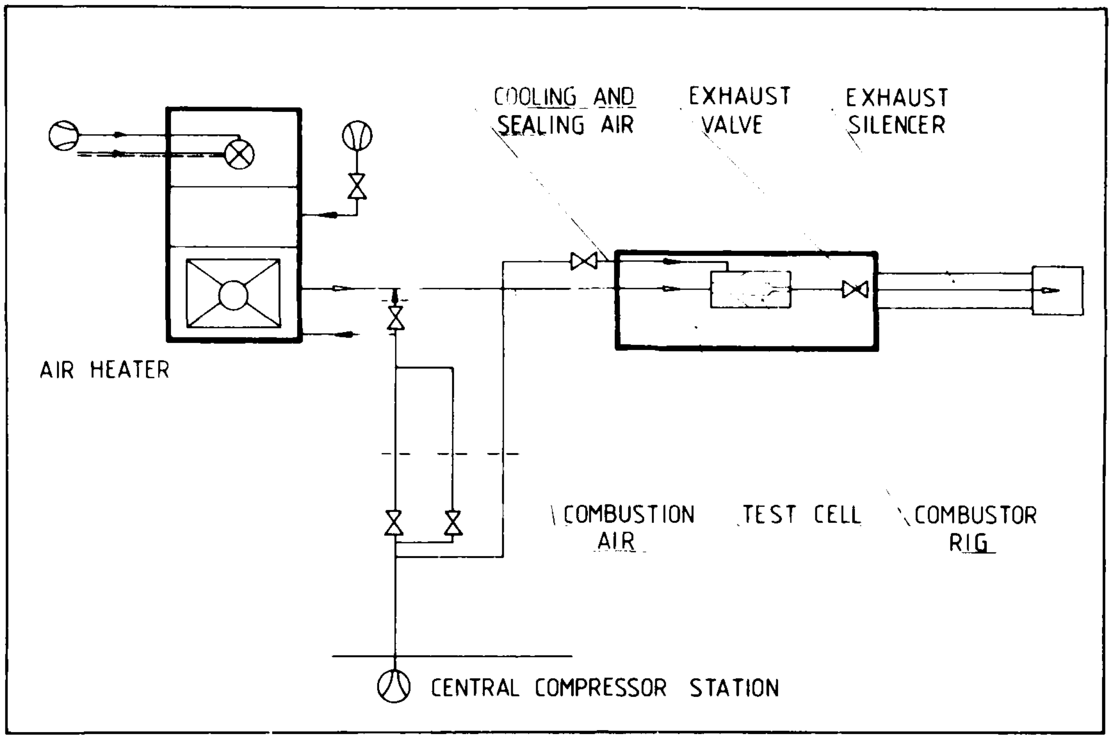

Fig. 1 Schematic of HP/HT Combustor Test Site

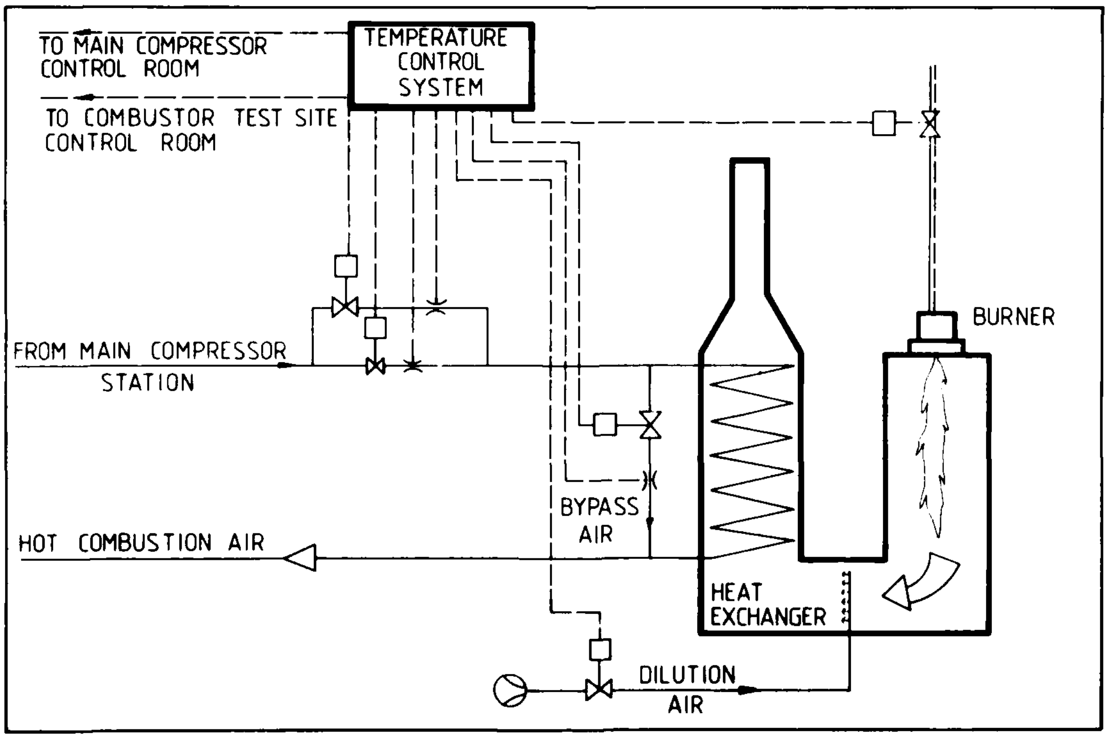

Fig. 2 Combustor Air Heater

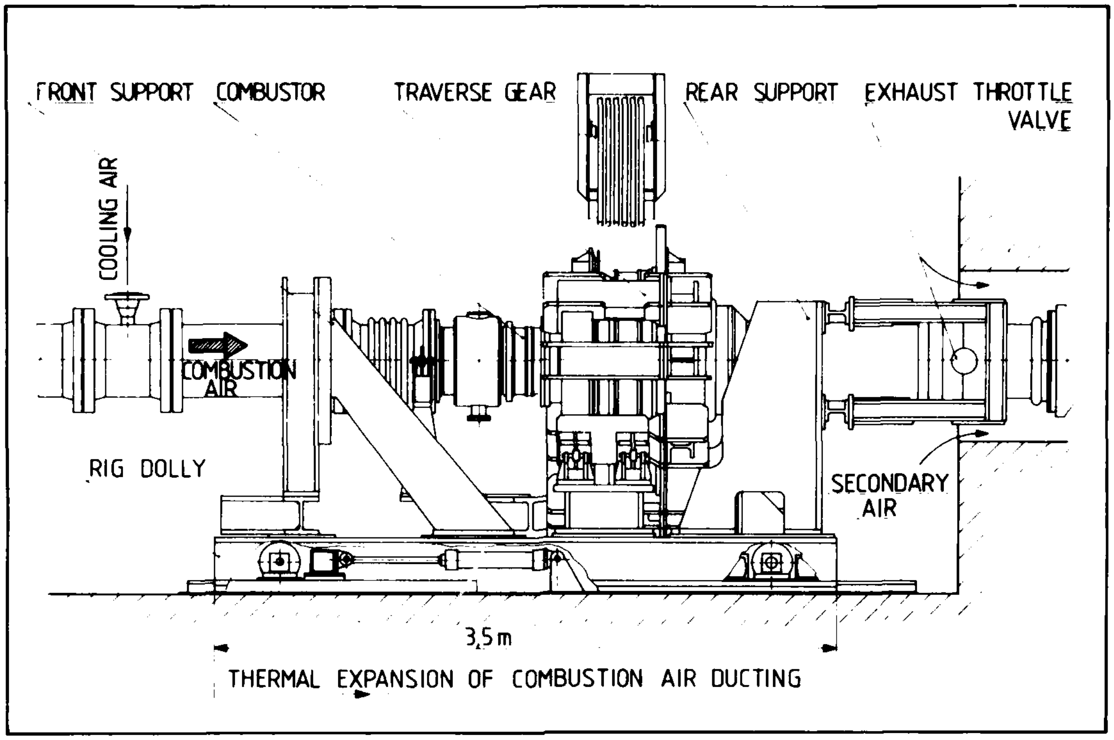

Fig. 3 HP/HT Combustor Test Rig Arrangement

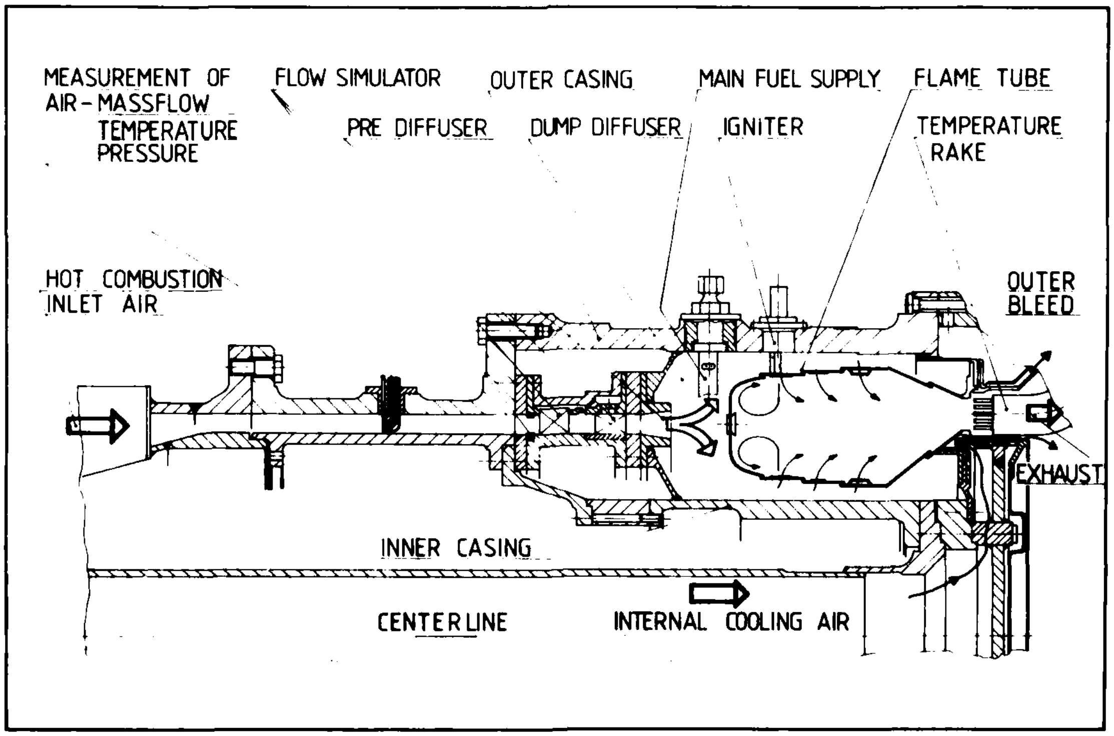

Fig. 4 HP/HT Research Combustor Test Rig

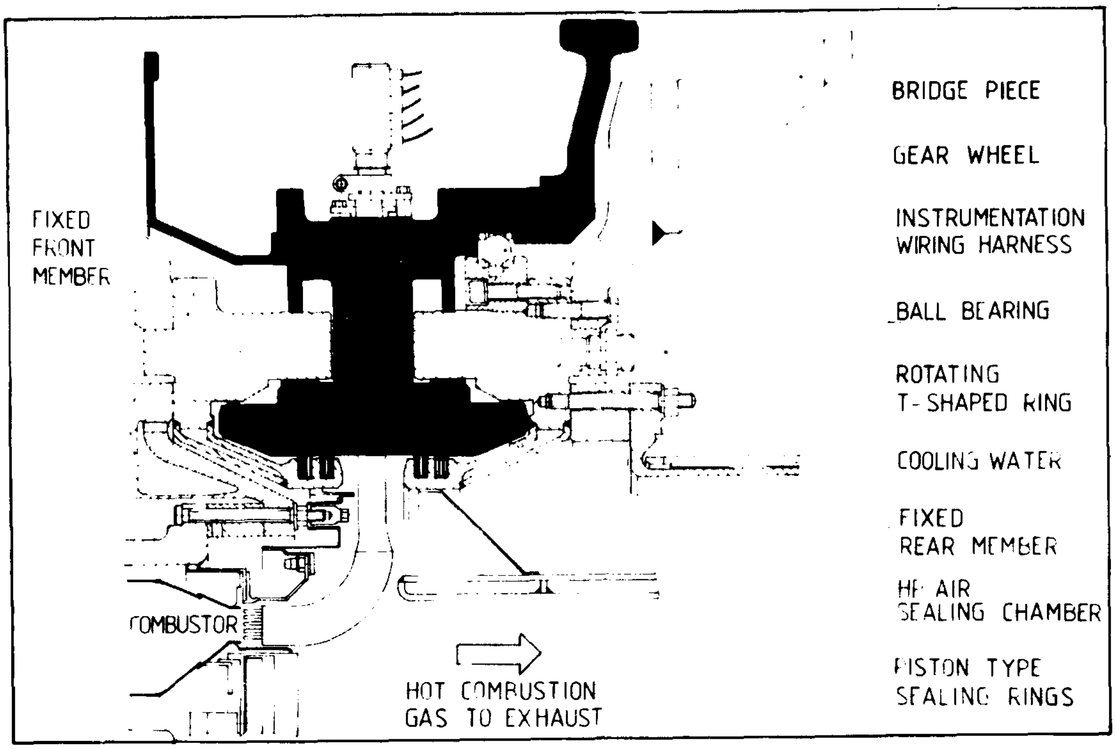

Fig. 5 Traverse Gear at Combustor Exit

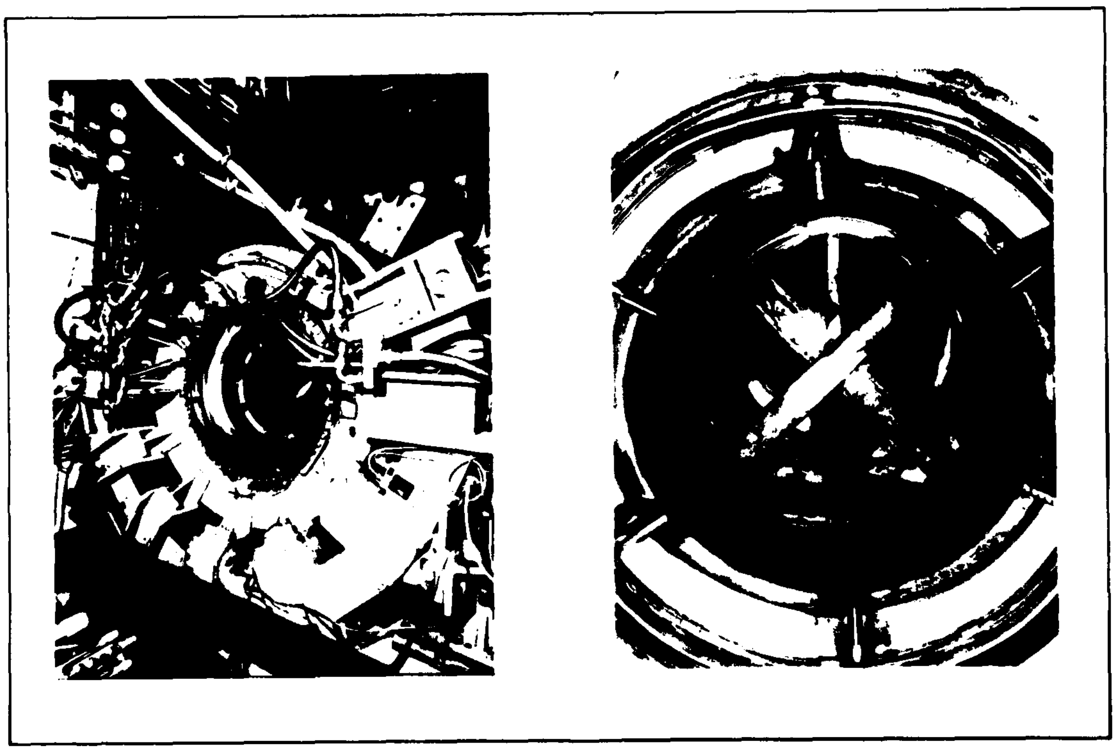

Fig. 6 Traverse Gear and Exhaust Ducting

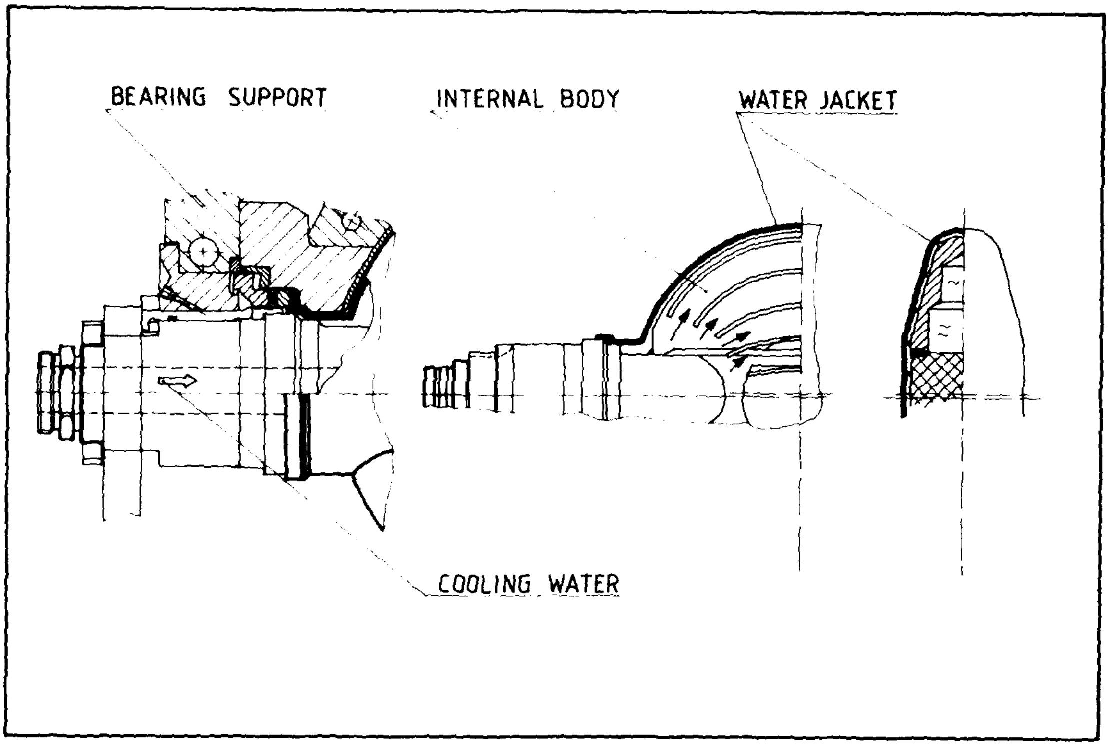

Fig. 7 Exhaust Throttle Valve

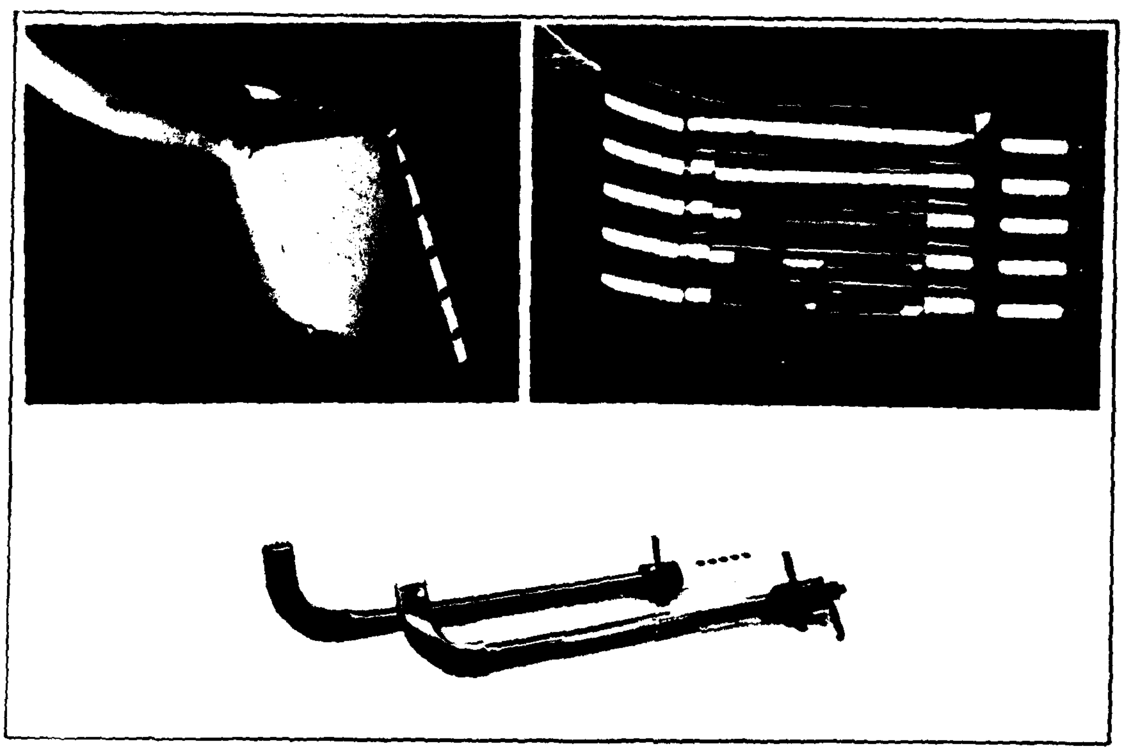

Fig. 8 Temperature and Gassampling Rakes

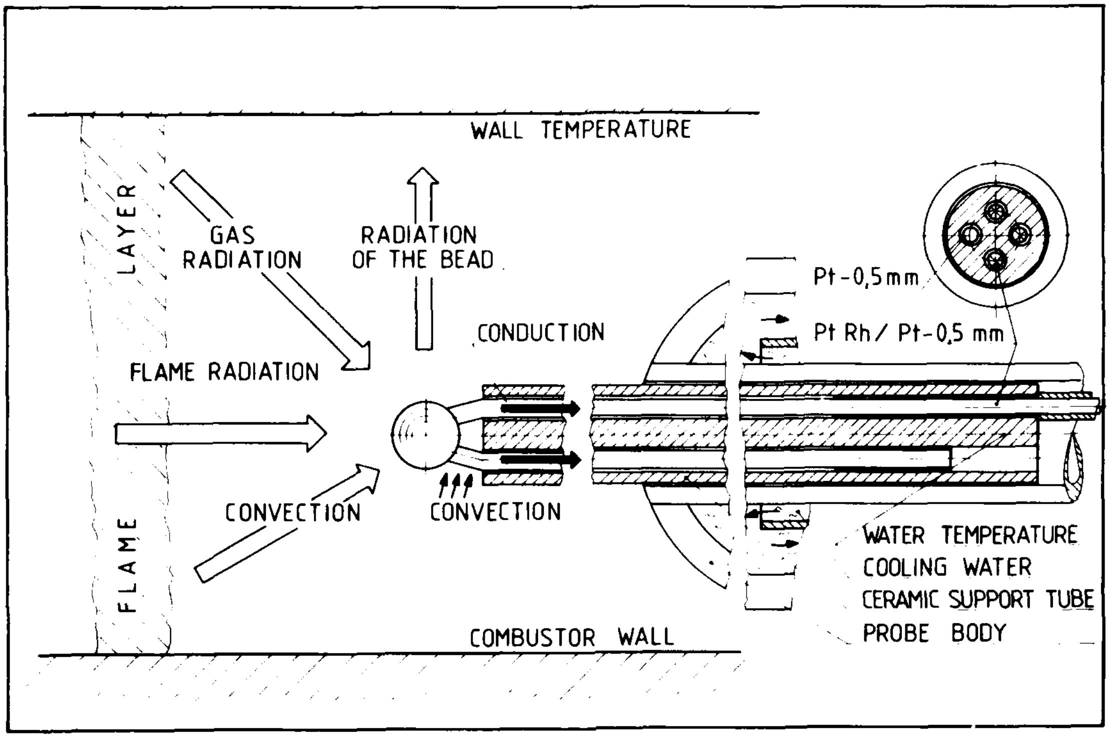

Fig. 9 Thermocouple Heat Balance Model

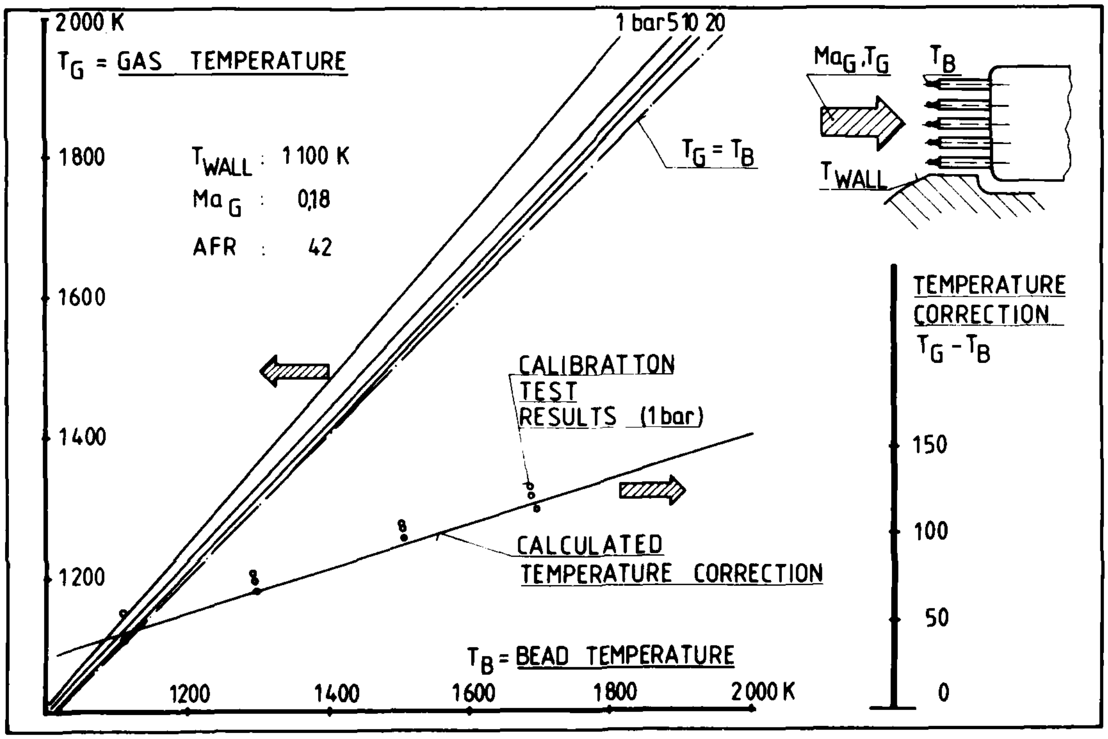

Fig. 10 Temperature Rake Bead Correction

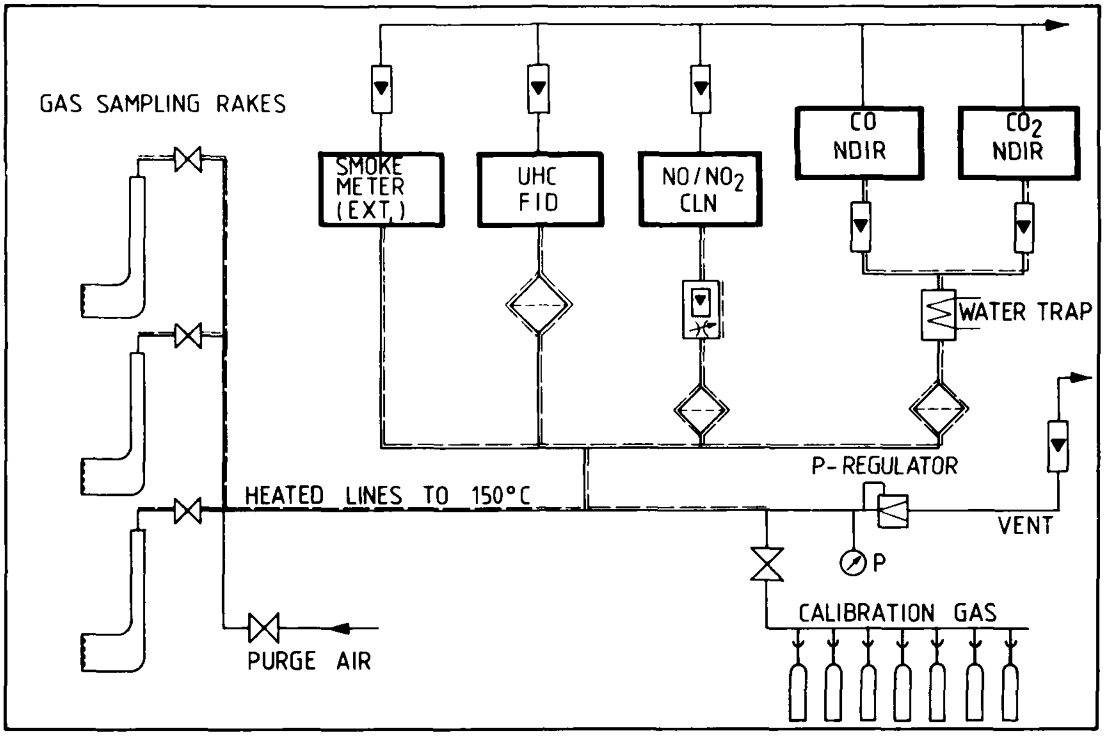

Fig. 11 Gassampling and Instrument Arrangement

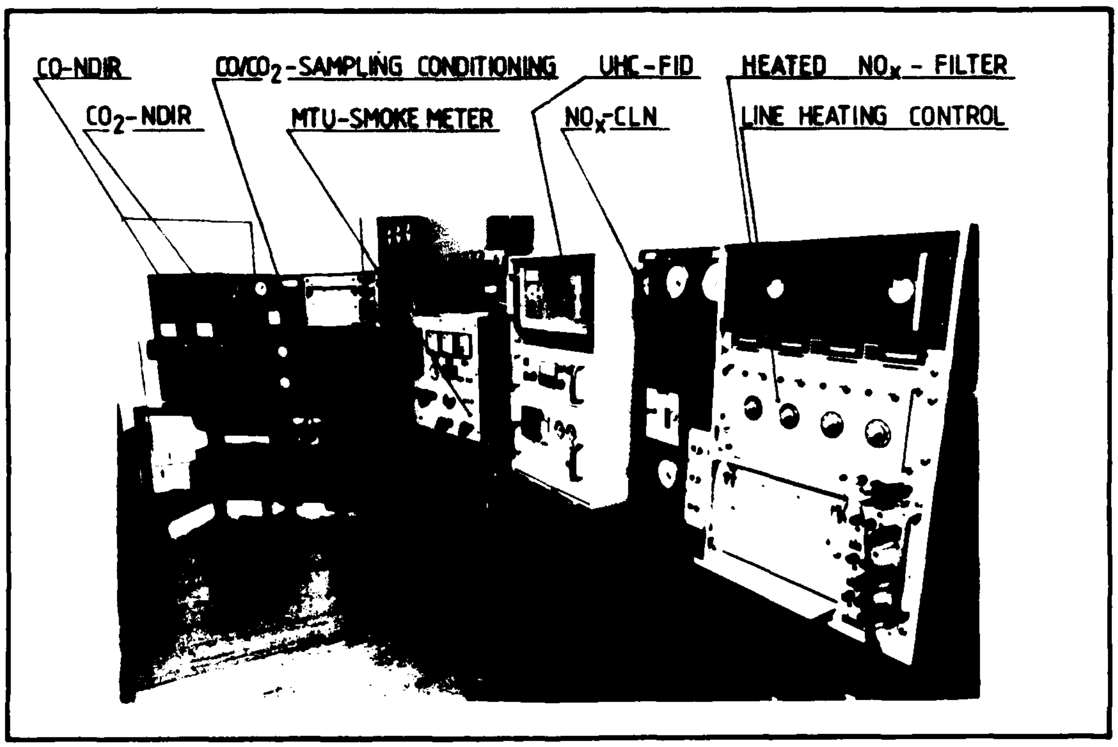

Fig. 12 Gas Analysis Equipment

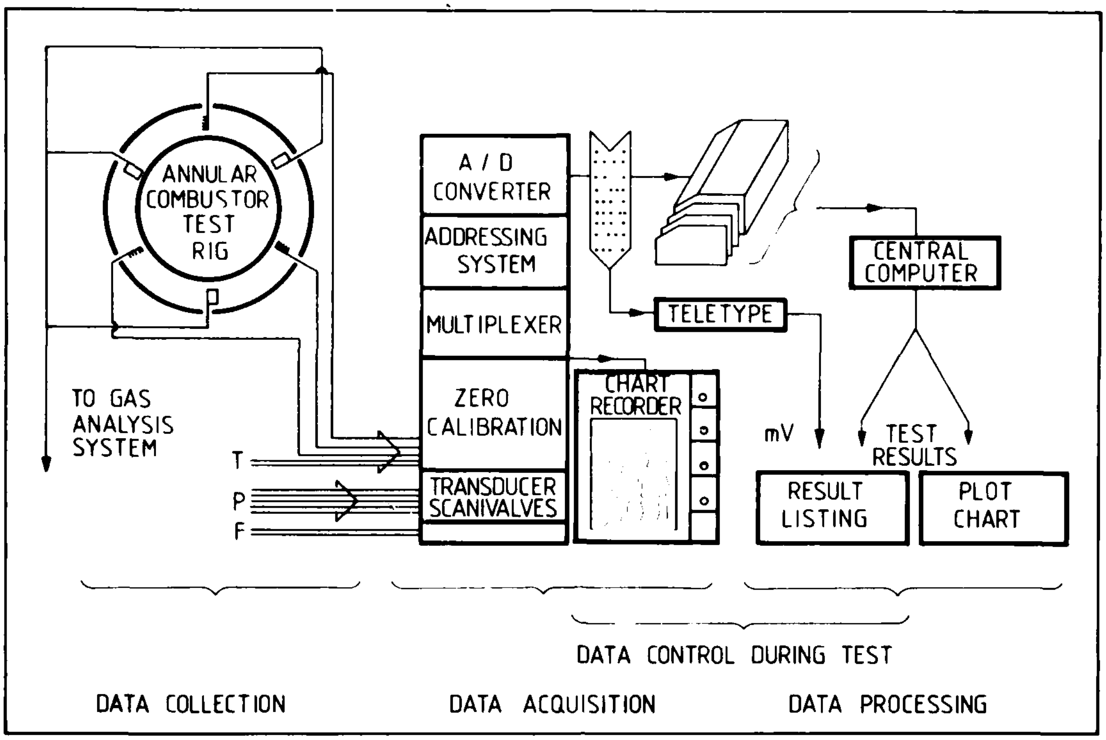

Fig. 13 Schematic of Data Acquisition System

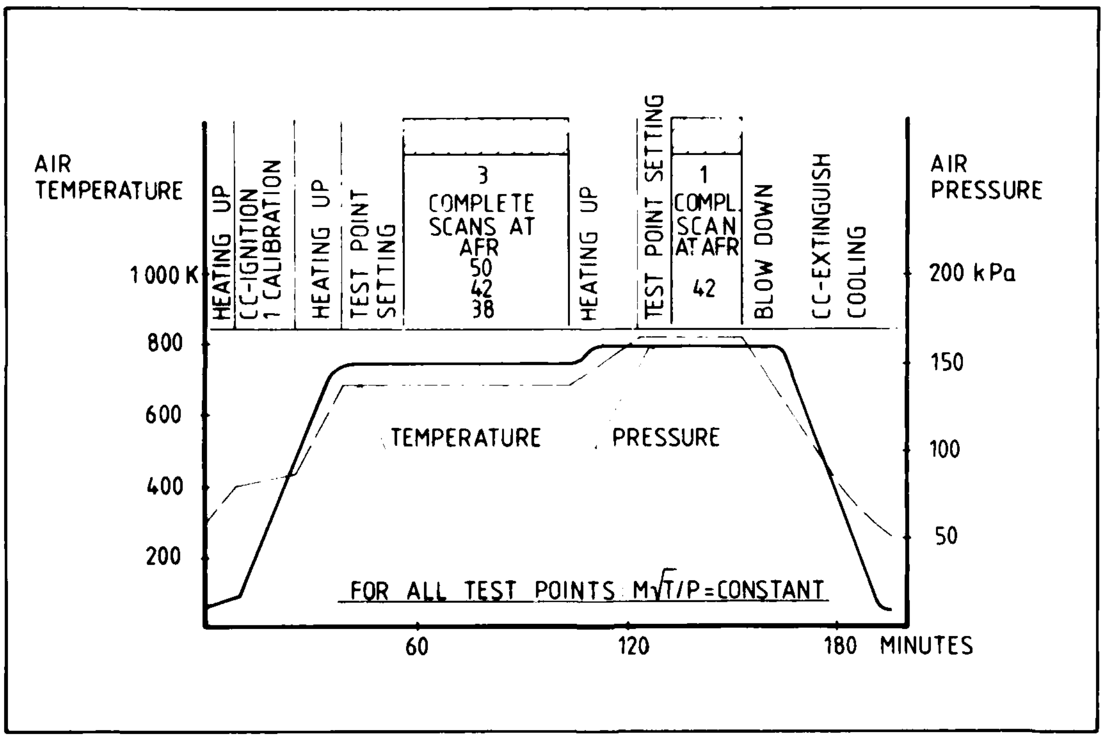

Fig. 14 Typical Combustor Test Cycle

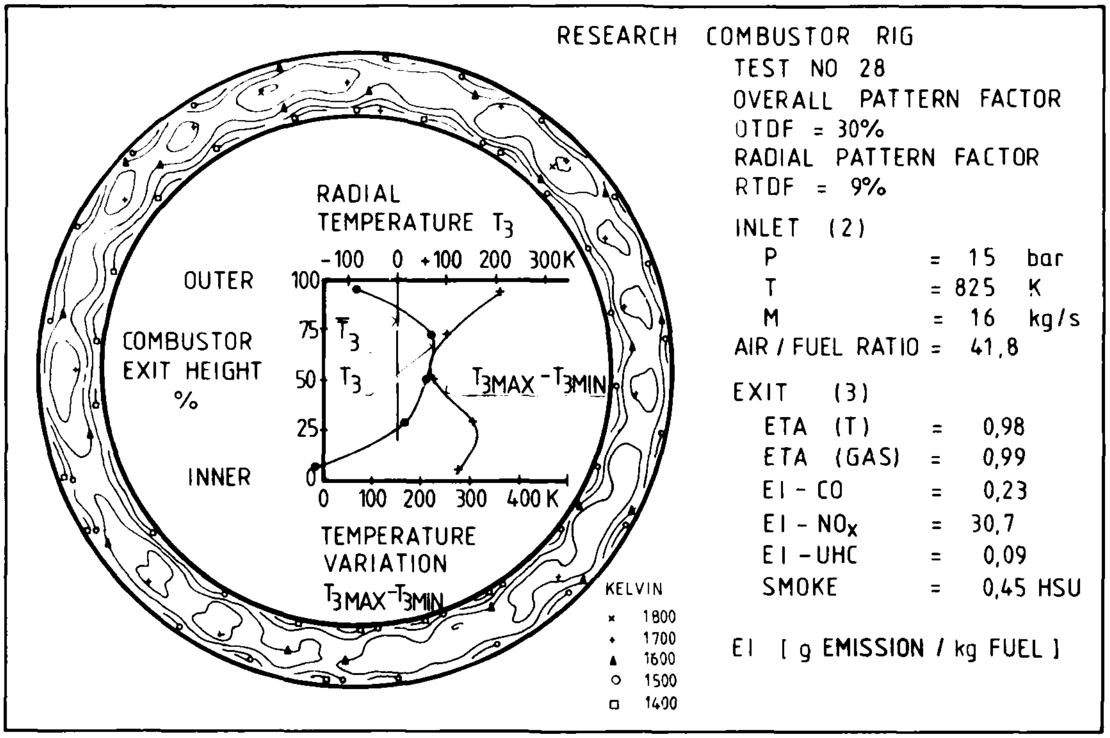

Fig. 15 Plot of Combustor Exit Gas Temperatures

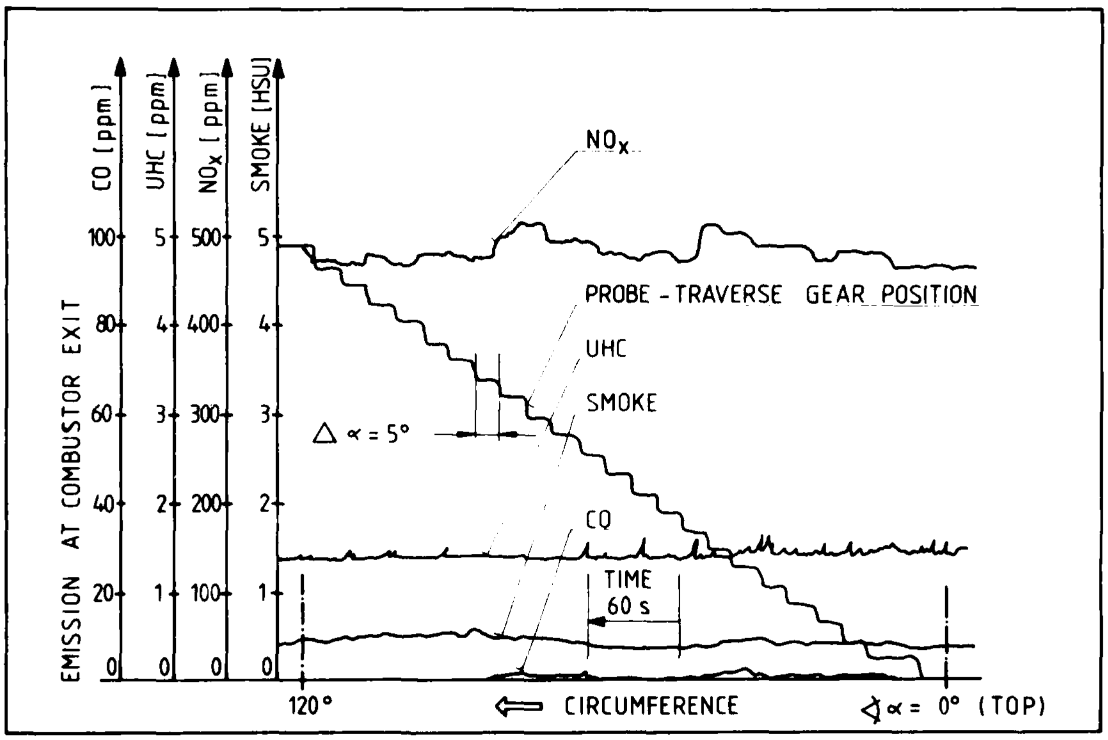

Fig. 16 Gas Analysis Chart

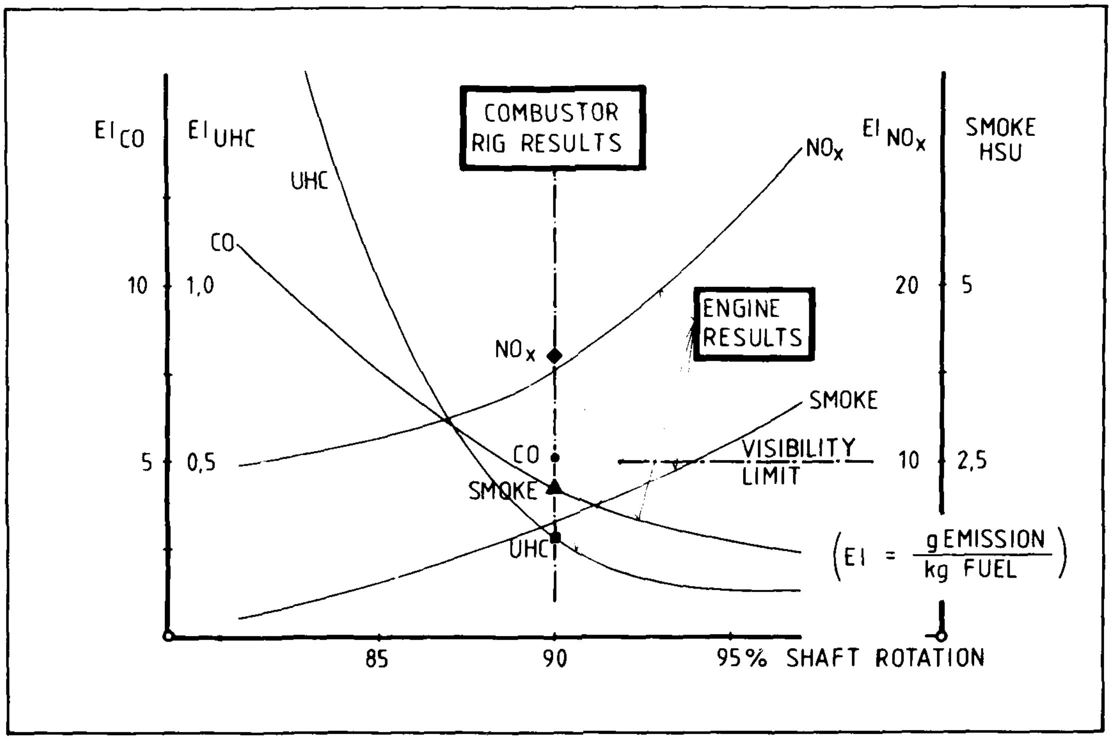

Fig. 17 Comparison of Combustor Rig/Engine Results

LOW PRESSURE TURBINE TESTING

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SUMMARY

The engine performance simulation model is the basis of all engine work and is also used for a reliable assessment of marketable performance, for the prediction of the flight performance of the aircraft, for the interpretation of the engine malfunctioning and then for a correct evaluation of the engine growth potential. The accuracy of the engine model is a function of the quality of the performance characteristics used for each component.

In accordance with this concept, within a turbofan development program, FIAT AVIAZIONE carried out a comprehensive investigation on the Low Pressure Turbine in order to define the component performance with the best possible accuracy. Different kind of tests have been performed, from bidimensional, rotating cascades and a cold flow rig test to "in engine" component testing. The advantages and the intrinsic limits of each kind of test are discussed herebelow.

The first part of this paper deals with the Low Pressure Turbine theoretical prediction methods used in FIAT AVIAZIONE and a correlation between predictions and rig results is also shown. The second part of the paper compares the rig results with the "in engine" ones measured with an appropriate instrumentation fitted on the engine.

LIST OF SYMBOLS

η	isentropic efficency
T_t	absolute total temperature
P_t	absolute total pressure
P_s	absolute static pressure
γ	specific heat ratio
C_p	specific heat at constant pressure
Π	power
M	mass flow
A	area
α	absolute flow angle
Ma	Mach number
Re	Reynolds number
R	gas constant
Y_{tot}	pressure loss coefficent = $(P_{t_1} - P_{t_2})/(P_{t_2} - P_{s_2})$
i	incidence angle
$\Delta H/U^2$	stage loading factor
$\Delta H/T$	specific entalpy drop
$M\sqrt{T}/P$	flow function
N_L	low pressure spool speed
$N_L/\sqrt{T}$	non-dimensional speed

Subscripts

1	turbine inlet section
2	turbine outlet section
REF	reference value
T	direct method
π	power method
WSC	wall static continuity method

1.0. INTRODUCTION

The optimization of a new engine project concerns all areas supporting the development. The engine simulation by mathematical models is quite important in order to explore the engine operating characteristic through the complete operating envelope. Such models are helpful both for assessing the marketable performance and as a diagnostic means for any modification needed by the general engine improvement; therefore they must simulate engine steady state operating conditions, starting characteristics, windmilling conditions and transient performance.

A reliable mathematical model cannot be worked out by simply extrapolating data from similar engines, but also an accurate analysis of the individual component behaviour is required. Such an analysis is generally based on theoretical predic-

tions during the preliminary design phase; afterwards rig investigations allow, within certain accuracy limits, both the aerodynamicists to verify their predictions and the performance specialists to improve the quality of their mathematical model.

The final step should be the assessment of each component performance when installed in the engine; however time and cost saving suggest to perform this step as soon as possible during the development program.

Within the development program of a turbofan engine a step by step investigation has been carried out by FIAT AVIAZIONE on a Low Pressure Turbine. Such a turbine has two stages with a mean radius of about 200 mm and an average blade height of 95 mm. The loading factor is $\Delta H/U^2 = 1.9$ on the first stage and $\Delta H/U^2 = 1.6$ on the second stage. The expansion ratio is over 2.4 : 1.

The above mentioned investigations have been carried out as follows:

a) Theoretical prediction of the turbine performance by using the method by Ainley & Mathienson as modified by Dunham & Came.

b) Profile pressure loss checks on annular cascades.

c) Check of the overall row characteristics on rotating and static cascades.

d) Testing of the whole turbine on cold flow test rig.

e) "In engine" turbine testing.

The results obtained together with the advantages and disadvantages of each single step of the investigation program are described in this paper. Sometimes only a few steps out of the five mentioned might be needed, depending on the type of the turbine to be analysed and on the kind of engine on which it is fitted.

2.0. ENGINE MATHEMATICAL MODEL

The need of mathematical models to describe the engine performance with a good accuracy during the several steps of an engine development program, implies the availability of the components characteristics in terms of flow capacity, pressure ratio, efficiency, specific enthalpy drop, surge margin and so on. Furthermore these characteristics should be defined at sea level static as well as within the flight envelope.

A comprehensive investigation through the widest range of operating conditions is therefore required with the aim of defining the influence on the theoretical components maps of the following factors:

- Reynolds number effect.

- Aerodynamic pattern upstream and downstream the components installed in the engine.

- Actual tip clearances on the engine.

- Heat transfer between core engine and by pass duct.

- Influence of secondary and cooling flows.

- Surge margins for compressors.

- Mechanical and aerodynamic limits.

The required amount of data and the difficulty of extending such an analysis to all the engine components make difficult the optimization of the mathematical model. For istance, due to the working temperature levels and the contained dimensions, the High Pressure Turbine is a critical component to be investigated in the engine; in this case an accurate definition of compressors and LP Turbine performance (compared with the overall turbine ones), allows the HP Turbine characteristics to be estimated.

Extensive testing is needed for the model optimization to identify the critical components in respect to possible engine performance shortfalls. This analysis, firstly carried out theoretically, is then supported by testing the engine at sea level as well as in the Altitude Test Facility, and afterwards in flight. The engine computer model can make profit of these data for further optimizations and at the same time is a very useful means for the preparation of the test program too.

The influence of a particular component on the engine performance is a function of: (i) engine configuration, (ii) component own characteristics, (iii) characteristics of the component coupled on the same shaft with the one under analysis. Appendix 1 shows for three different flight conditions the influence of the Low Pressure Turbine on the performance of a mixed flow, low bypass turbofan. The results of the analysis show in terms of thrust, specific fuel consumption, shaft speed, by-pass ratio the LPT influence on the engine matching and emphasize the engine growth potential by only modifying the LPT characteristics, provided that the resulting matching does not badly affect handling and shaft speeds.

3.0. TURBINE PERFORMANCE PREDICTION METHODS

The use of the analytical methods to predict turbine performance leads to simplified models due to the extreme complexity of the flow through the blades caused by three dimensional temperature and pressure distributions. The introduction of computerization allowed great improvements of these methods by making possible to be defined, with good accuracy, the required optimum blade shapes. This is applicable as well to off design performance evaluations.

The analytical method used in FIAT AVIAZIONE to predict turbine performance is the well known one by Ainley & Mathienson as modified by Dunham & Came (ref 1 and 2). This method uses semi-empirical relationships to calculate the turbine blade pressure losses as a sum of profile (Y_p), secondary (Y_s) and tip clearance losses (Y_k).

$$Y_{tot} = Y_p + Y_s + Y_k$$

These pressure losses are defined as the total pressure drop through the row divided by the dynamic head at the row exit ($Y_{tot} = (P_{t_1} - P_{t_2})/q_2$).

Sometimes this procedure leads to a turbine map quite different from the one obtained by subsequent rig tests as shown in Fig. 1. These differences are mostly due to inaccuracies in defining the total pressure losses with the above mentioned relationships; in particular the major uncertainity is related to secondary and tip clearance losses because of the complexity of this phenomenon and the interaction among the different sources of loss.

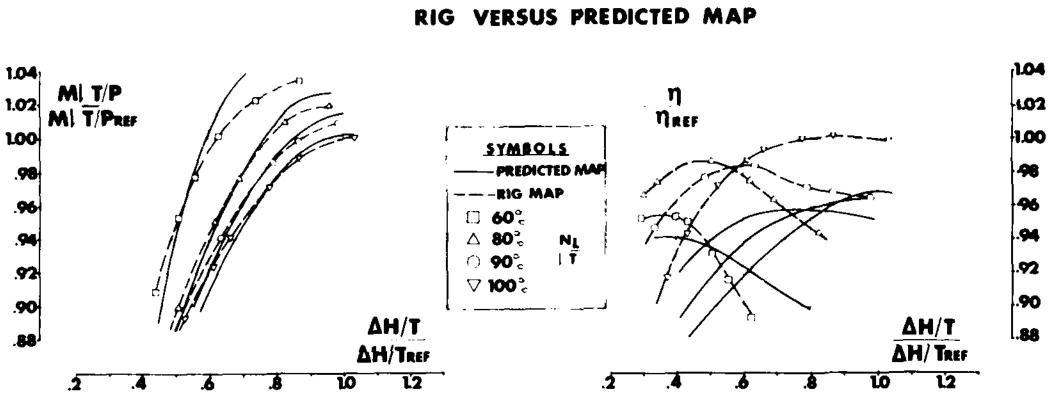


Fig. 1

In order to achieve a turbine map somewhat more realistic than that obtained in the above way, the component performance calculation program has been fed with input values relative to blade row pressure losses measured on cascade test rigs.

4.0 RIG TEST INVESTIGATION

Three rig test procedures can be used for pressure loss measurements: (i) rectilinear cascades, (ii) annular cascades, (iii) rotating cascades.

- Rig tests on rectilinear cascades are helpful for profile losses investigation. In this case secondary loss measurements are not meaningful: in fact it is quite impossible to simulate the blade span-wise static pressure gradient and blade surface/end wall boundary layers mixing causing such losses. Therefore, for conventional blade profiles, these tests become useless and time consuming; it is possible finding in the literature how to evaluate the profile losses for conventional geometry blades.
- Annular cascades rig tests are very useful to define the cumulative (profiles +secondary) pressure loss characteristics for stators. These cascades extend over a certain sector of the annulus; if the sector is not large enough, (i.e. narrow than 120 degrees), the measurements might be affected by the boundary conditions, causing in this way appreciable discrepancies between annular cascade and complete blade row. Unluckily such a rig test is not reliable for rotors because the blade spanwise static pressure distribution is different in the reference system, preventing the secondary losses from being investigated.
- Rig tests on rotating cascades allow accurate analysis on rotor pressure losses. In this case the relative total pressures must be worked out from the absolute total pressure and the absolute flow angles; following this tecnique only the mean circumferential value of pressure losses can be measured. Investigations of tip clearance influence on pressure losses can also be carried out, on this test rig, supplying useful data to transfer test rig results to engine and mathematical models.

An accurate evaluation of the pressure loss characteristics for each single row of a multistage turbine implies testing on particularly designed annular and rotating cascade test rigs, but such a solution was found too expensive and time consuming; besides the matching of the single results could produce some errors in the overall pressure loss picture. To overcome these problems FIAT AVIAZIONE have built a multi-use test rig on which the testing of single rows and complete turbine can be done. In this way pressure loss characteristics of the annular cascades (stators) and of rotating cascades (rotors) can be obtained alternatively taking into account the mutual influence of the blades rows previolusly tested.

4.1. Rig description

In Fig. 2 the possible configurations of the test rig, when arranged for cascade investigations, are shown. The four blade rows are interchangeable with other rows having different geometry (stagger angle and contained annulus modifications). These features allow to check many configurations on the same test rig with the same instrumentation.

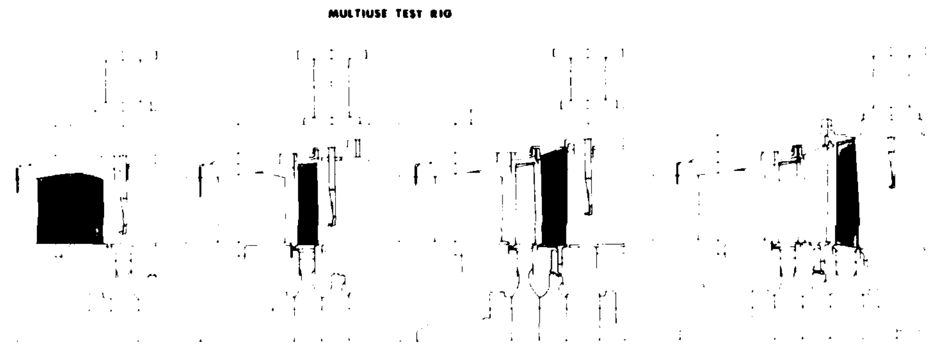


Fig. 2

The air mass flow supplied by a centrifugal compressor electrically driven is measured with a calibrated nozzle. The braking system is made by two hydraulic brakes of 12000 HP total power at 12000 rpm max speed. Bearing lube flow and temperature raise rate are measured during the test; this allow the absorbed power to be taken into account for turbine efficiency calculations.

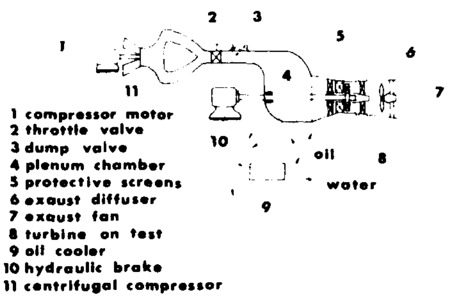


Fig. 3

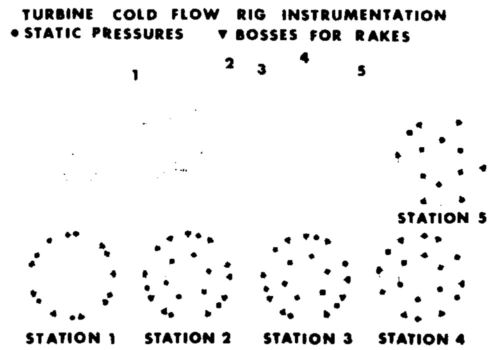


Fig. 4

The test rig is equipped (Fig. 4), with rakes of total pressure and temperature probes with five reading points each. The provided instrumentation allows three different methods to calculate the efficiency.

– Method 1 : based on temperature and pressure measurements upstream and downstream the turbine.

– Method 2 : based on brake power and total to total expansion ratio.

– Method 3 : based on brake power and total to total expansion ratio evaluated by calculation of downstream total pressures from static pressures, swirl angles, airflow and areas.

The relationships by which efficiencies have been calculated are listed in the following table.

The 2nd method (efficiency based on power measured) is the most accurate as it can be seen from the analysis reported in Appendix 2. The rig test efficiencies quoted in this paper refer to this method.

METHOD 1

$$\eta_T = \frac{T_{t1} - T_{t2}}{T_{t1}\left[1 - \dfrac{1}{\left(\dfrac{P_{t1}}{P_{t2}}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}}\right]}$$

$T_{t1}, T_{t2}, P_{t1}, P_{t2}$ MEASURED

METHOD 2

$$\eta_{\pi} = \frac{\dfrac{\pi}{M \cdot C_p}}{T_{t1}\left[1 - \dfrac{1}{\left(\dfrac{P_{t1}}{P_{t2}}\right)^{\frac{\gamma-1}{\gamma}}}\right]}$$

$\pi, M, P_{t1}, P_{t2}, T_{t1}$ MEASURED

METHOD 3

$$\eta_{W.S.C} = \frac{\dfrac{\pi}{M \cdot C_p}}{T_{t1}\left[1 - \dfrac{1}{\left(\dfrac{P_{t1}}{P_{t2}}\right)^{\frac{\gamma-1}{\gamma}}}\right]}$$

π, M, T_{t1} MEASURED

P_{t1}, P_{t2} CALCULATED ON THE BASIS OF

$P_{S1}, P_{S2}, A_1, A_2, \alpha_1, \alpha_2$

$$P_t = P_S\left(1 + \frac{\gamma - 1}{2} Ma^2\right)^{\frac{\gamma}{\gamma-1}}$$

$$Ma = \sqrt{\frac{-1 + \sqrt{1 + 2(\gamma - 1)\left(\dfrac{M\sqrt{T_t}}{P_S \cdot A \cdot \cos\alpha}\right)^2 \dfrac{R}{\gamma}}}{\gamma - 1}}$$

Table 1

4.2 Rig test procedure and results

To investigate stator characteristics at a given inlet Mach number and aerodynamic incidence, many measurements over a grid similar to that superimposed on Fig. 5 are required. The grid density must be higher in the wake region and at each point both total and static pressures have to be measured. In order to minimize the interference between instrumentation and flow, a remote-controlled yawmeter, previously calibrated on a wind tunnel, is to be adopted. An example of the results obtained from these measurements is shown in Fig. 5. From such a map mean pressure losses area weighted for several radial stations can be obtained.

In the case of a rotor the test is quicker not being necessary to move the yawmeter circumferential wise; in fact the measuring system, due to its inertia, behaves like a filter feeling the mean circumferential value only. The measurement, performed for several radial stations, should be then repeated for the required values of aerodynamic incidence.

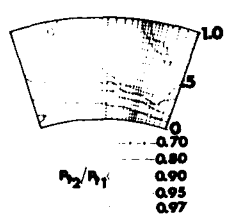


Fig. 5

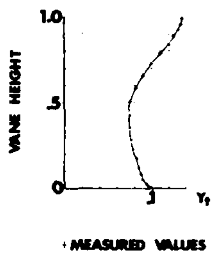


Fig. 6

From a set of pressure loss maps, shown in Figs. 5 and 6 for stators and rotors respectively, total pressure loss versus aerodynamic incidence characteristics can be worked out. The incidence variation for the 1st stator can be done by changing the stagger angle of the blades, while for the following blade rows it can be done by varying the rotor angular speed. The characteristics obtained on the multiuse rig for the LP Turbine, we are dealing with, have been compared in Fig. 7, with the Ainley & Mathienson curves.

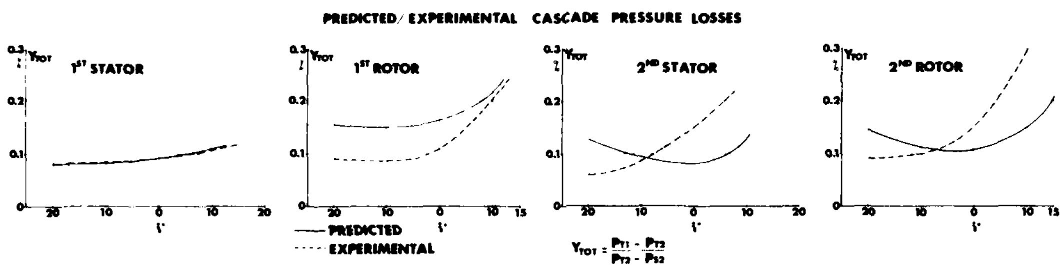


Fig. 7

To justify the appreciable differences recorded on the rows downstream the 1st stator, it must be pointed out that the pressure loss characteristics prediction as suggested by A. & M. method mainly refers to conventional blade profiles, while the ones used for the turbine under discussion have shaped by interactive method computer aided allowing for thinner blade shapes and better performance.

When all the cascade investigations are performed the rig is able to test the overall turbine in order to achieve a map. The turbine map calculated by using the measured row pressure losses vs incidence shows a good agreement with the overall turbine characteristics measured on the test rig (Fig. 8) for all the relative non dimensional speeds.

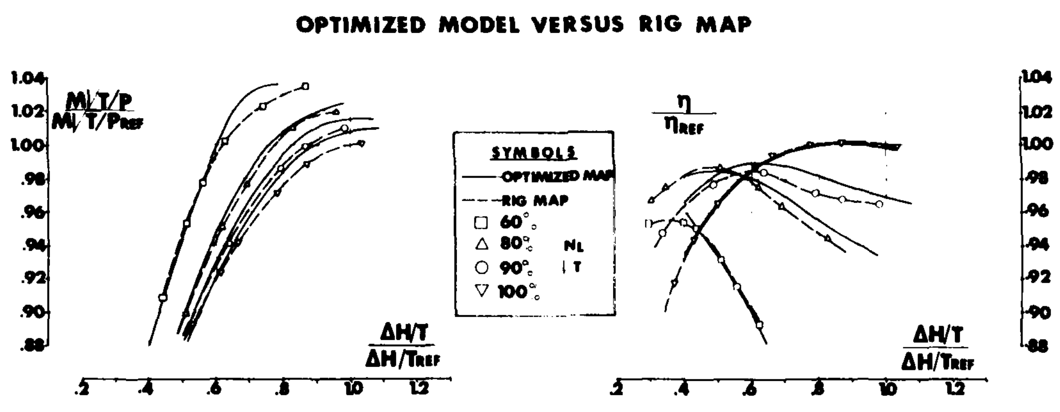


Fig. 8

5.0. L.P. TURBINE MATHEMATICAL MODEL CALIBRATION

An accurate calibration of the mathematical model allows reliable performance predictions for any geometry modification of the basic turbine (blade restagger and contained annulus modifications) intended to produce the required engine rematching.

When dealing with several possible configurations the mathematical model is an helpful and reliable means for a comparative analysis; the most suitable solution, and that only, can be then tested on the test rig saving in this way money and time. Provided the mathematical model is accurately optimized no appreciable deviations are to be expected from the last verification on the test rig.

The model optimization implies test conditions on the test rig as consistent as possible with the actual turbine behaviour in the engine; besides the mathematical model must include all the refinements necessary to cope with the "in engine" performance. The method used to minimize the discrepancies of test rig results and then the mathematical model (optimized through rig testing) against the "in engine" component behaviour is illustrated in the following table.

RIG TEST VS ENGINE	MATHEMATICAL MODEL VS ENGINE
Reynolds Number For the turbine under discussion the ratio Re on rig / Re on engine) is contained between 1.7 and 2.0 for the different ratings. The measured efficiency should then be corrected in accordance with the following relationship. $(1-\eta_{eng}) = (1-\eta_{rig})\ (Re_{rig}/Re_{eng})^{0.2}$	Reynolds Number When pressure loss characteristics are available for each row the Re correction can be done with the formula reported in ref. 2. $(Y_p + Y_s)_{corrected} = (Y_p + Y_s)\ (Re / 2 \times 10^5)^{-0.2}$
Air → cold gas → hot gas The change of Cp and R (function respectively of temperature and fuel air ratio) and the throat areas variation due to thermal and centrifugal effects partially counterbalance and often annul each other. In this way the rig tests and engine flow functions equality also guarantees the aerodynamic similarity in the axial direction. In order to guarantee the aerodynamic similarity in the tangential direction, as the two mentioned effects do not compensate each other, the following ratio should be used: $(N/\sqrt{T})_{eng} / (N/\sqrt{T})_{rig} = \sqrt{\dfrac{\gamma_{eng}\ R_{eng}\ A_{rig}}{\gamma_{rig}\ R_{rig}\ A_{eng}}}$	Air → Gas The mathematical model accounts for any Cp and R values, furthermore it is also able to correct Cp by flow static temperature through the rows being this more accordance with the real flow. Cold gas → hot gas The model could allow for geometry variations at different ratings, provided that it is correctly instructed (correct thermal analysis available).
Tip clearance The rig tests flexibility allows the definition of an experimental relationship to correct the performance for different running clearance values.	Tip clearance The mathematical model could be adapted to allow for the experimental curve obtained on the rig to predict both rig and engine turbine behaviour.
Flow distortion It is very difficult to obtain on the rig a satisfactory simulation of the flow distortions. Furthermore that could imply very complex and expensive modifications of the rig.	Flow distortion The mathematical model, being one-dimensional, cannot consider particular profiles for the inlet parameters. Moreover it must be pointed out that "generally" flow distortions attenuate through the rows and then should affect the pressure losses of the 1st row only.

6.0. "IN ENGINE" TURBINE PERFORMANCE INVESTIGATION

In order to improve the knowledge of the turbine characteristics when installed in the engine it may be useful to investigate their performance using the engine as a test bench.

The original target for this test was the drawing of an installed turbine map. The test procedure was in fact studied in such a way to explore the whole map (different fans, variable nozzle area, different bleed configurations, particularly shaped nozzles to split hot and cold flows). Unluckily the intrinsic difficulties of this kind of analysis did not allow the achievement of such an ambitious target; however this kind of test turned out to be very useful to define: (i) pressure and temperature profiles, upstream and downstream the turbine, (ii) absolute efficiency level in operating conditions.

6.1. Instrumentation

The engine readout is not problem-free and to our experience it can be regarded as reliable only when a large number of measuring points and a particularly accurate procedure are used. The type and density of the instrumentation to be fitted on the engine depend on the kind of measurement errors affecting the results. In fact, pressure and temperature distorsions on the engine and furthermore the high temperature values achieved, sensibly affect the choice of thermocouples type and position.

Overall measurement error limit has been defined for the LP Turbine under discussion, by analysing the possible causes of error: systematic and random.

<u>Systematic errors</u> (SE)

– Temperature and pressure probes inaccuracies	:	Eliminated by installed probe calibration.
– Data acquisition system uncertainty	:	Negligible because of transducer calibration.
– Pressure profile inaccuracy due to limited number of probes	:	Negligible because of the flat shape of profiles.
– Temperature profile inaccuracy due to limited number of probes	:	Estimated from previous tests in about 1K at turbine inlet and oulet.
– Temperature inaccuracy due to rakes axial displacement	:	Core to by-pass duct heat exchange, taken into account.

<u>Random errors</u> (RE)

– Entry and exit temperature instrumentation precision	:	± 0.15 percent
– Entry and exit pressure instrumentation precision	:	± 0.30 percent
– Number of points taken at the same rating		

The overall measurement error limit is defined as follows:

$$\mathrm{OMEL} = \sqrt{(\mathrm{SE})^2 + (\mathrm{RE})^2}$$

Following the above analysis the instrumentation has been a ranged as shown in Fig. 9.

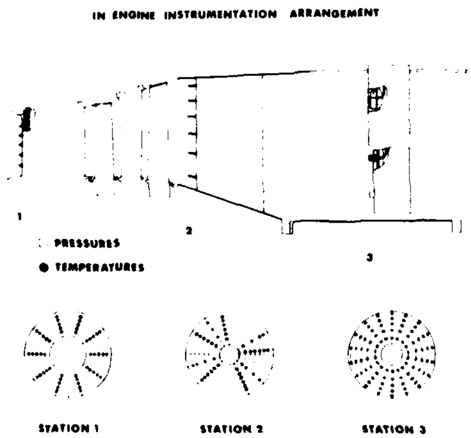


Fig. 9

Inlet and outlet thermocouples have been calibrated against master thermocouples in order to contain mean errors within ± 0.5 K around the max temperature values. Pressure probes of the Kiel type have been used.

The measurement error influence on efficiency evaluation is discribed in Appendix 2. It can be observed that this influence is rapidly increasing when decreasing turbine load. It appears then quite clearly how difficult is to obtain reliable efficiencies at low speed, even with good measurement accuracy.

The flow function at turbine inlet is influenced by possible errors in engine by-pass ratio estimates, other than by temperature and pressure errors at LP Turbine inlet.
In the present case the evaluation of the by-pass ratio is based on an energy balance method requiring measurements or estimates of the following parameters:

- Engine combustion chamber and by pass duct inlet temperature
- Total air flow at engine inlet
- Fuel flow
- Combustor efficiency and pressure losses
- Secondary flows
- Power offtakes

The obtainable accuracies for the above parameter measurements and the relative weight on air flow evaluation through the core engine are reported in table 2.

PARAMETER	ERROR ± %	RELATIVE WEIGHT ON MASS FLOW EVALUATION %
INLET MASS FLOW	1.0	0.2
LPT EXIT TEMPERATURE (CALIBRATED)	0.15	0.2
BY-PASS DUCT ENTRY TEMPERATURE	0.5	0.6
COMPRESSORS EXIT TEMPERATURE	0.5	0
FUEL MASS FLOW & HEATING VALUE	0.5	0.5
COMBUSTION EFFICIENCY	0.3	0.3
SECONDARY FLOWS	0.7	0.7
ENGINE ENTRY TEMPERATURE	0.3	0.5
LPT EXIT TEMP. PROFILE (CALIBRATED)	0.15	0.2

Table 2

The following turbine performance parameters precision (with 95 % confidence level) is expected around design point:

Efficiency	η	±1.5 %
Inlet flow function	$M\sqrt{T}/P$	± 0.7 %
Specific entahalpy drop	$\Delta H/T$	± 1.4 %

6.2. Test procedure and results

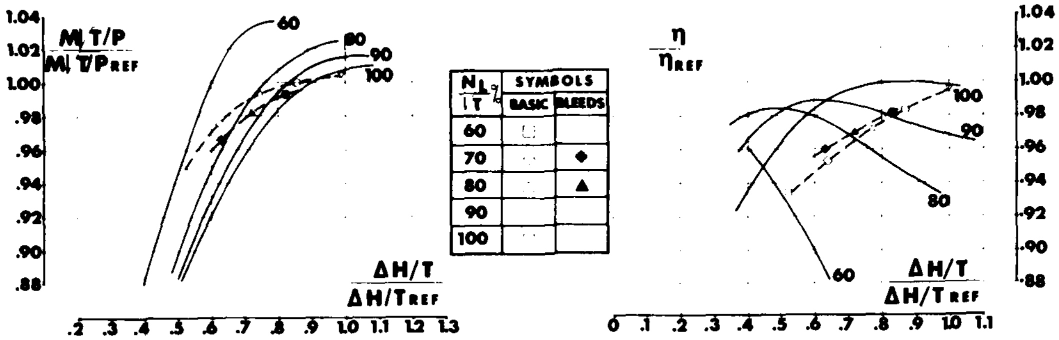
In the attempt of drawing the turbine map a test procedure based on engine geometry variation has been adopted. In fact the different components match themselves in a certain way when installed in the engine making so difficult to investigate the turbine map on points outside those lying along the steady state working line.

In order to obtain working line shifts, different fan, nozzle areas and bleed configurations have been tested. The engine mismatching has been also attempted by replacing the nozzle with a special one, splitting the hot and cold flows and varying the two flows geometric areas. Nevertheless all the mentioned configurations did not allow appreciable excursion of the working line unless in a very narrow band.

During the development of such a test program one realizes that pursuing a lot of different configurations to obtain the required running line excursion is complex, expensive and time consuming. At the end no better results than those obtainable with a good mathematical model can be expected.

Even if the original aim of the investigation has not been totally achieved, satisfactory results have been obtained both in terms of performance parameters level and accuracy; the results obtained are compared in Fig. 10 with the theoretical map (optimized through rig tests). Reynolds numbers, radial tip clearances, secondary and cooling flows have been taken into account for a consistent comparison. The majority of the experimental points lie on the working line. It can be observed that a good accordance in terms of efficiency, flow function and specific enthalpy drop has been achieved at a fixed relative non dimentional speed.

IN ENGINE LP TURBINE WORKING LINES



N_L % / √T	SYMBOLS BASIC	SYMBOLS BLEEDS
60		
70		◆
80		▲
90		
100		

Fig. 10

From the graph of Fig. 11 showing the measured data plotted versus N/ $\sqrt{T}$ it results that the predictions of the parameter measurement precision are confirmed (with a 95°/o confidence level).

TEST RESULTS AT VARIOUS BLEED CONFIGURATIONS WITH TWO DIFFERENT STANDARDS OF FAN

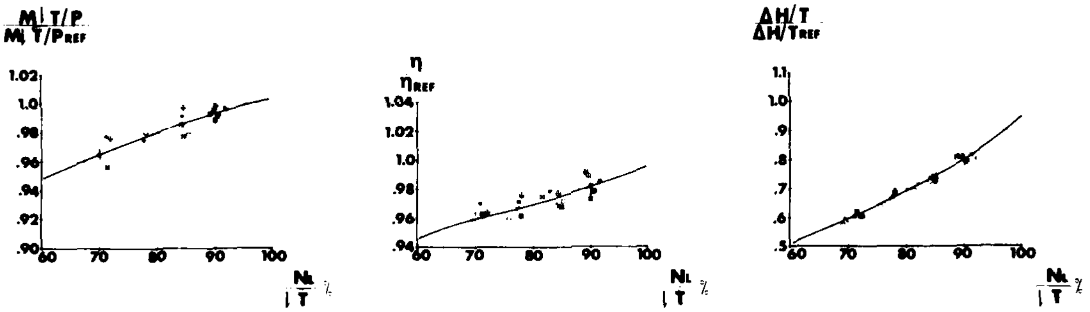


Fig. 11

When using the direct method, i.e. temperature and pressure measurement upstream and downstream the turbine, the efficiency calculated is much more accurate when the average (weighted) pressures and temperatures represent the real value (i.e. the systematic errors are removed).

One of the purposes of the investigation has been also to find (through subsequent tests) the optimum number of pressure and temperature taps for a correct average value definition, in the same time avoiding a too heavy instrumentation, which could alterate the turbine characteristics. When a too heavy instrumentation is fitted on one or more components the apparently satisfactory results obtained, often refer to a component or engine of different geometry.

The analysis carried out suggests that particular attention must be paid to the temperature reading distribution, while for pressures, due to the flat profiles, also a low instrumentation density gives a satisfactory answer.

In Fig. 12 the obtainable measurement accuracy is plotted against temperature reading density. It comes out that a satisfactory accuracy (i.e. ± 1K at max rating) can be achieved with an instrumentation density not higher than 400 reading points/square meter.

From what already told it emerges that further improvements on efficiency evaluation are hardly obtainable with the used measuring technique. Alternative techniques like hot film and laser anemometry could be used. However these techniques entail in addition to the background needed for the results interpretation, to overcome installation problems. The hot film anemometer, for instance, should be fitted in a part of the engine where high temperature gas is flowing and where an intensive analysis of the thermal and vibration pro-

MEASUREMENT ACCURACY VERSUS INSTRUMENTATION DENSITY

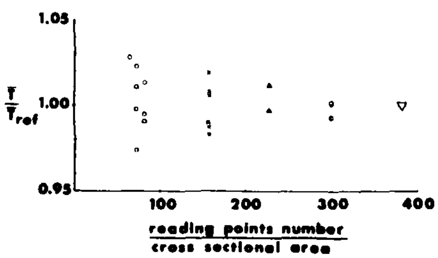


Fig. 12

blems is required. Furthermore this particular technique does not suit the hot region investigation very well. The laser anemometry does not suffer for the above problems, but on the other hand requires the installation of cumbersome devices on the outer casing of the engine. Both the mentioned techniques are very expensive and the data acquisition and interpretation can be done only after accurate calibrations.

7.0. CONCLUSIONS

The achievement of a reliable engine mathematical model, especially during the early development phase, is of primary importance for the success of an engine program.

Such a model requires an accurate definition of "in engine" component characteristics.

The extensive use of sophisticated computer methods allows the introduction of unique features to optimize the aerodynamic performance of a specific component.

Therefore, when dealing with high performance engine components, it is difficult to find a satisfactory generalized prediction method. It comes out that, mainly in the preliminary design phase, an intensive experimental investigation work is necessary to support and then verify theoretical predictions.

The rotating cascade tests, performed during the investigation program of the LP Turbine under discussion, proved to be the most meaningful rig tests from the point of view of the turbine simulation model. The best results can be obtained from a multiuse bench; in this way the interaction between rows can be taken into account without resort to particulary designed annular and rotating cascade test rigs, being this last solution too complex and expensive.

The LP Turbine tests carried out on the engine did not permit the drawing of a map. During the development of such a test program, one realizes that pursuing a lot of different configurations to obtain such a result in complex, expensive and time consuming, and no better results than those obtainable with a good mathematical model can be expected. Nevertheless this test allowed the determination of performance characteristics along the steady state working line and the definition of pressure and temperature profiles upstream and downstream the turbine.

Useful data can be worked out following the above investigation procedure for: (i) rig results to engine reduction and then improvement of the mathematical model, (ii) turbine design optimization, (iii) performance prediction of derivative configurations.

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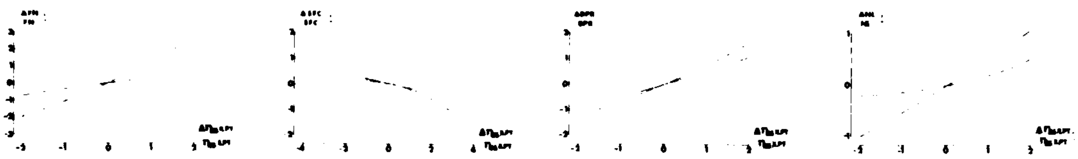
APPENDIX 1 – LP TURBINE INFLUENCE ON ENGINE PERFORMANCE

The influence of a particular component on the engine performance is a function of: (i) engine configuration, (ii) component own characteristics, (iii) characteristics of the component coupled on the same shaft with the one under analysis.

In the following figures the influence of the low pressure turbine on the performance of a mixed flow low by-pass turbofan is shown.

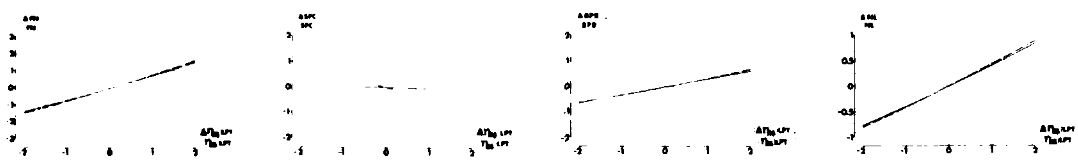
Fig. 1-1 – SEA LEVEL STATIC



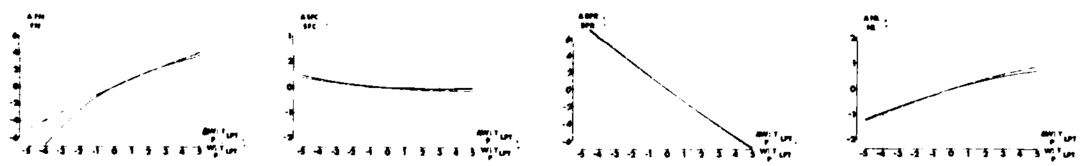
(a) (b) (c) (d)

(e) (f) (g) (h)

Fig. 1-2 – SEA LEVEL Ma = 0.9

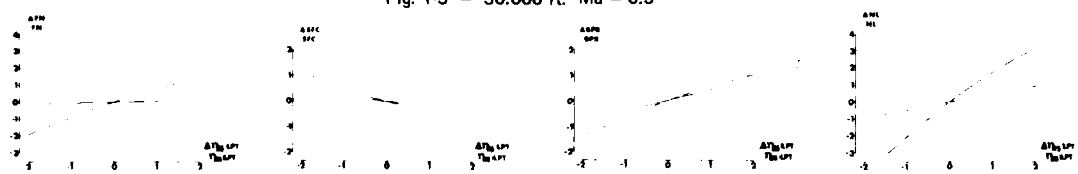


(a) (b) (c) (d)

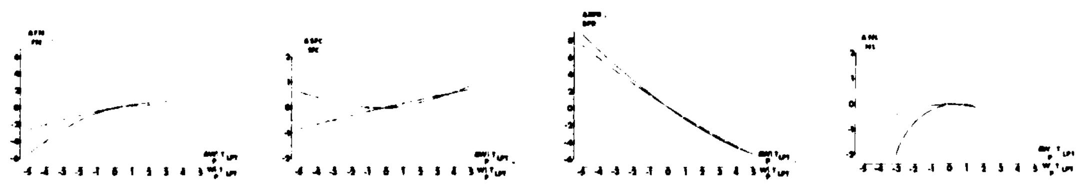


(e) (f) (g) (h)

Fig. 1-3 – 30.000 ft. Ma = 0.9



(a) (b) (c) (d)



(e) (f) (g) (h)

The flow capacity of the LP turbine has been modified by ± 5 % and the efficiency by ± 2 %. The analysis has been carried out with two different fan characteristics; the first with the efficiency versus speed relationship shown in figure 1-4 the second one with a constant efficiency versus speed.

The results of the above analysis show in terms of thrust, specific fuel consumption, shaft speed and by-pass ratio the LP Turbine influence on the engine growth potential by only modifying the LPT characteristics, provided that the resulting matching does not badly affect handling and shaft speeds.

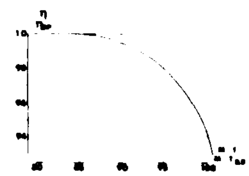


Fig. 1-4

APPENDIX 2 – EFFICIENCY EVALUATION METHODS

The most suitable method to be used for turbine efficiency evaluation on test rig is investigated here with reference to the three methods available listed in the table of paragraph 4.1.

The main differences among the three methods lie in the different way of calculating the actual power produced by the turbine (i.e. the numerator of the efficiency formula and also the denominator in the 3rd method). The (i) direct method requires the measurement of the turbine outlet temperature, the (ii) power method the measurement of power and air mass flow and the (iii) wall static continuity method the measurement of power, massflow, outlet static pressures and flow angles.

The assumed measurement precision for all the parameters required by the three above mentioned methods are:

– Total pressure measurement	$\Delta P_t/P_t$	=	±0.3 %
– Static pressure measurement	$\Delta P_s/P_s$	=	±0.3 %
– Total temperature measurement	$\Delta T/T$	=	±0.15 %
– Brake power measurement	$\Delta \Pi/\Pi$	=	±1.0 %
– Air mass flow measurement	$\Delta M/M$	=	±1.0 %

A – Comparison between method 1 and method 2

The mean square error has been calculated with the following relationships for direct and power methos respectively.

$$\Delta\eta_T = [(\delta\eta_T/\delta T_1)^2\,\Delta T_1^2 + (\delta\eta_T/\delta T_2)^2\,\Delta T_2^2 + (\delta\eta_T/\delta P_{t1})^2\,\Delta P_{t1}^2 + (\delta\eta_T/\delta P_{t2})^2\,\Delta P_{t2}^2]^{1/2}$$

$$\Delta\eta_\pi = [(\delta\eta_\pi/\delta T_1)^2\,\Delta T_1^2 + (\delta\eta_\pi/\delta P_{t1})^2\,\Delta P_{t1}^2 + (\delta\eta_\pi/\delta P_{t2})^2\,\Delta P_{t2}^2 + (\delta\eta_\pi/\delta M)^2\,\Delta M^2 + (\delta\eta_\pi/\delta\Pi)^2\,\Delta\Pi^2]^{1/2}$$

The graphs shown in Figs. 2-1 and 2-2 have been worked out by developing the above relationships. It results quite clearly that, with the assumed parameter precisions, the power method is largely better than the direct one. It can be also noted that in both cases the error increases rapidly when decreasing turbine load ($\Delta H/T$), making so difficult an accurate efficiency definition particularly at low ratings. Even if expensive torque measuring devices are required, the power method is the preferred one because of the better accuracy obtainable. On the other hand the direct method does not need power and mass flow measurements and when pressure and temperature profiles and radial work distribution are of interest, could be preferred or used in parallel to the power one, despite the worse accuracy.

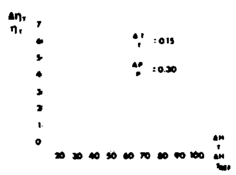


Fig. 2-1

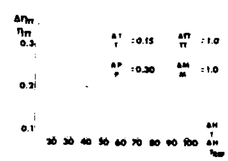


Fig. 2-2

B – Comparison between method 2 and method 3

The only difference between these two methods is the different way of calculating the turbine outlet total pressure. The theoretical comparison has been then based on ΔP errors only. The results are shown in Fig. 2-3. Also in this case the power method is allowing for better accuracy. However it must be pointed out that the wall static continuity method should be the only one able to define the real total to total efficiency, while the power method should allow a total to axial total efficiency evaluation only. On the other hand the use of Kiel probes, insensible to misaligne-ment between flow and probe axis up to ±45 degrees overcomes this problem.

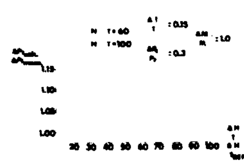


Fig. 2-3

DISCUSSION

Don Rudnitski, NRC, Ca

What method of determining flow direction was used? Was it a cobra probe with nulling techniques? If so, was it manual or computer controlled?

Author's Reply

It was a cobra type probe; the nulling method was used. It was not computerized, but manually done, using manometers.

Rig Investigation of a Two-Stage, Single Shaft Low Cost Turbine.

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Summary

In this paper is reported the development programme of a two-stage, single shaft turbine.

The programme is based on several tests performed in the single shaft two stage turbine configuration, the objective being the optimization of the off-design performance, correct distribution of the first and second stage workload and harmonization of the nozzle throat areas.

Tests have been carried out or are planned to investigate the effect on the efficiency of the rotor tip clearance, Reynolds number effect and cooling flows. The result analysis is compared with the predicted carpet of each stage individually and of the two stages together.

1. Introduction

The AR 318 Turboprop is the first turbine engine designed and developed by Alfa Romeo Aviazione. It is a low cost engine of 600 SHP for the application on a 10.000 lbs take off weight aircraft in the commercial commuter market.

As fig. 1 shows, it is a very simple engine and its particular components are designed to get easy maintenance. Its major modules are as follows:

- epicycle gear box including the lobe axial air intake;
- single stage high pressure ratio centrifugal compressor;
- high efficiency low pollution reverse flow combustor;
- integral casting two stage turbine.

From the first run, Sept. 1977, the engine has totalized 800 hours on the test bed and 50 flying hours on a Beechcraft King Air. Aerodynamic Rig (Turbine, Compressor, Flame Tube) and Mechanical Rigs (Gear Box, Spin Pit) have run for many and many hours to support the development of the engine.

In this paper is reported the work done on the turbine both on the cold rig and on the engine.

2. Cold Rig Test

2.1 Test Vehicle

The test vehicle was designed to simulate the turbine work[illegible] ditions of the engine. in order to optimize the relative throat are[illegible] zles and of rotors it was provided the feature to vary the stagger [illegible] the blades. The material selection was made to get the turbine [illegible] sions. The rig is sketched in fig. 2, it is important to empha[illegible] chanical solution adopted to facilitate the replacement of [illegible] (blades and discs). It is possible, by minor changes, to [illegible] the two stage configuration or in the single stage a[illegible] stage). The mechanical design allows zero leakage [illegible] required.

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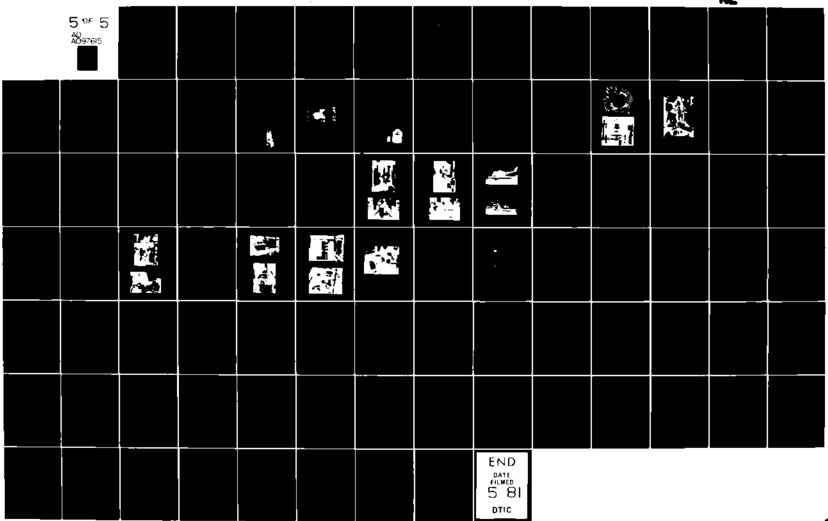

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The inlet duct simulates the engine combustion chamber exit. The rig is also suitable to accept engine parts from production standard definition.

The instrumentation adopted (as shown in fig. 3) was studied to get the overall turbine performance and the meridional flow field analysis for each row of blades. The probe locations in the indicated measurement plane are the following:

ITEM	PLANE
Total Temperature probes	0, 40, [illegible]
Anemometer probes	40, [illegible], 70
Static pressure probes	10, 11, [illegible]0, 30
	40, [illegible]0, 70
Clearance probes	[illegible], [illegible]

An automatic data acquisition system, inclusive of 100 pressure transducers, 100 termocouples, [illegible] capacitance probes, [illegible] vibration probes, connected by microprocessors to a Honeywell [illegible] computer allowed a fast recording. A specific software was produced to perform the data analysis.

[illegible]. Turbine Rig Facility

The facility was designed to cover (4 complete range of AR 318 turbine working conditions at different Reynolds numbers, allowing the facility an absolute total pressure at the turbine exit up to 0.1 ata. The mass flow is provided by a two stage screw compressor giving [illegible]0.000 Nm^3/h and a pressure ratio of 1.[illegible]. It is possible to set any combination of mass flow, turbine inlet pressure and temperature with a fixed exhaust pressure.

The facility has the capability to simulate the turbine cooling by additional flows regulated as a percentage of turbine main flow. A scheme of the facility is shown in fig. 4. The turbine is connected to a high speed dynamometer (30000 RPM, 900 CV) that allows a very accurate reading of the turbine shaft torque.

[illegible]. Cold Rig Test Programme

The programme was arranged to investigate different assemblies to optimize the turbine characteristic related to the engine performance.

So that the programme was developed as consequence of the engine test results to get the best turbine and compressor matching. In the table 1 are shown the turbine tests made during the AR 318 Advanced Demonstrator Programme.

[illegible] Calibration of initial engine standard turbine

The calibration was carried out for different speed lines at supercritical Reynolds number.

In figs. [illegible], [illegible] it was reported the corrected mass flow and the efficiency as a function of the specific work. The results showed a discrepancy between theoretical and experimental data. The total to total efficiency level was higher than the predicted by [illegible] percent, but the blockage value of corrected mass flow was [illegible] percent down.

As will be shown in the next paragraph a similar reduction of corrected mass flow was found analyzing the engine performance. So it was clear that the turbine was running with a capacity lower than the forecast.

[illegible] nozzle capacity

To solve the outlined problem it was decided to test the turbine assembled with a first stage nozzle 4 percent larger. The overall [illegible] results reported in fig. [illegible] and [illegible], showed an improvement of turbine capacity [illegible] compared to the previous test, but still down respect to the theoretical value (- [illegible]). The problem was to determine if there was an underestimation of the first stage nozzle blockage factor or if there was a mismatching between the two stages. Unfortunately total pressure and temperature measurements at the interstage station were not good enough to [illegible] the phenomena, because of the feature of transverse actuators. Two actions were undertaken to try to solve the problem. The first one was to make the air flow test to define the real capacity of first stage nozzle. The second one was to find the answer by aerodynamic study using all available experimental data.

In the diagram of fig. 9 is reported the ratio between the first and second stage nozzle capacity, evaluated by the equation:

$$Y = \frac{Q_1 A_1}{Q_2 A_2} = \sqrt{\frac{T_1}{T_2}} \left[1 - \left(1 - \frac{T_2}{T_1}\right) \frac{1}{\eta}\right]^{C_p/R}$$

versus the total specific work and the estimated first stage efficiency.

The diagram showed for all the assumed first stage efficiency and for the measured work split the choked nozzle was the second one; it was assumed for that calculation the theoretical blockage factor for both nozzles. So those results needed the air flow test of the nozzles.

c) Air Flow Nozzle Test

Such a test was conducted for a family of nozzles with different water flow time. In fig. 10 are reported the corrected mass flows versus the ratio of total inlet pressure and static exit pressure.

The correlation between the choking values and the water flow time (fig. 11) showed the capacity of nominal throat area first stage nozzle was as forecast and the used blockage factors were corrected. Those results confirmed the second stage nozzle capacity was not proportionate to the effective stage work split. As consequence the turbine rig was assembled with nominal throat area for the first nozzle and +4% throat area for the second nozzle.

d) Rematched nozzle capacity

The calibration was carried out for the some speed lines of previous tests. The results shown in fig. 12 and 13 confirmed the validity of previous hypotesis: the turbine got the predicted choking capacity with small change in efficiency level.

The Y parameter values were compared in fig. 14 for the three mentioned assemblies. It was confirmed the assembly with a large second nozzle was the correct one.

That configuration was taken on the AR 318 engine and the right matching with the compressor was obtained.

e) First stage calibration

The calibration of the nominal first stage was performed to evaluate if there were any interference effects between the first and the second stage.

The results confirmed the capacity previously found and an efficiency level 1.[illegible]% higher than forecast.

f) Reynolds number effect

The purpose of such a test was to verify the theoretical correl[illegible]

tion used for the engine performance prediction, to simulate the effect of altitude changes on the turbine characteristic. That because for a small engine like the AR 318, designed for the commercial market of the light commuter plane it's very important to have good altitude performance, (low S.F.C.) and then the capability to predict them.

The Reynolds number, defined by the relation:

$$Rey = K \frac{M}{T^n} \qquad K = f(c_a)$$

was simulated by varying the pressure at the turbine exit [illegible]. The minimum Reynolds number tested was 1.0000. The result was plotted in fig. 16 and indicate a decrease of efficiency and mass flow with Reynolds number, which critical value was found to be 00000. That was in agreement with the Anley and Mathenson theory adopted in the engine performance simulation fig. 17.

3. Engine tests

Due to the difficulties of build up an engine fully instrumented to read all the turbine parameters the analysis was made using engine performance diagnostic computer programme. The parameters were read just at interstage and turbine exit.

Herewith are mentioned the engine tests relative to the investigation on the turbine capacity, first stage rotor tip clearance, altitude performance.

a) Turbine Capacity tests

The tests were made building two engines with different turbine configurations. The first one (nominal nozzles capacity) gave the same indications obtained on the cold rig. An output of the engine performance diagnostic program is reported in table 2. It showed a turbine capacity defect (-3%).

The second test (second stage nozzle throat area +4%) gave the design capacity, confirming the rig test results, as shown in fig. 18.

b) Rotor tip clearance

The first stage rotor tip clearance was analyzed just on the engine because experimental data correlating tip clearance and efficiency on the rig were already available. The purpose of such a test was to verify also a method of the rotor shroud cooling to reduce its the thermal expansion.

It was adopted an impingement cooling method showed in fig. 19 that gave good results.

In fig. 20 are reported the efficiency level versus the tip clearance obtained during those tests. Must be noted the rotor tip clearance is estimated by the rotor and shroud temperatures recorded knowing the material thermal expansion.

c) Altitude test

The AR 318 engine was tested on a flying test bed. Some of the test points obtained during the flight programmes shown in fig. 21 are compared with the performance predictions obtained by the use of the component characteristic up to date as consequence of the components rig testing. The flight test programme is still going on but those first results seem to confirm what was understood by the Reynolds number investigation on the component tests.

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Turbine cold rig programme

Test	Objective	Configuration
1	Calibration of initial engine standard turbine	Nominal capacity nozzles
[illegible]	Rematch by changing nozzles	+4% first stage nozzle throat area
3	Air flow tests	First stage nozzles
4	Rematch by changing nozzles	+4% second stage nozzle throat area
5	Calibration of first stage turbine	Nominal throat area
6	Evaluation of Reynolds number effect	Two stage configuration

TAB. 1

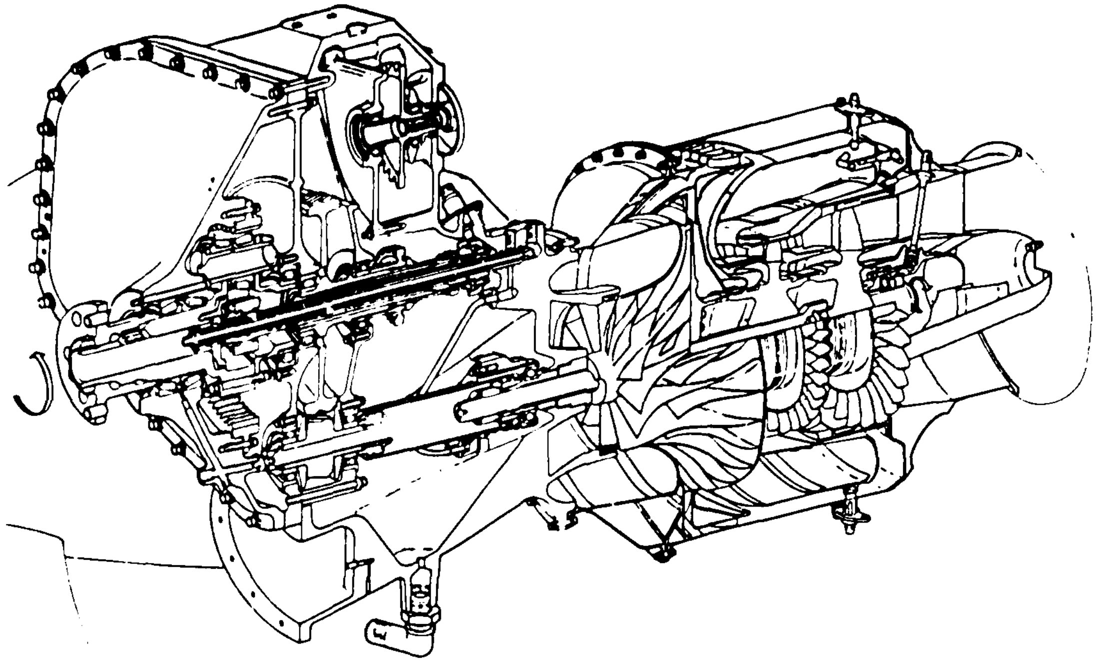


Fig. 1

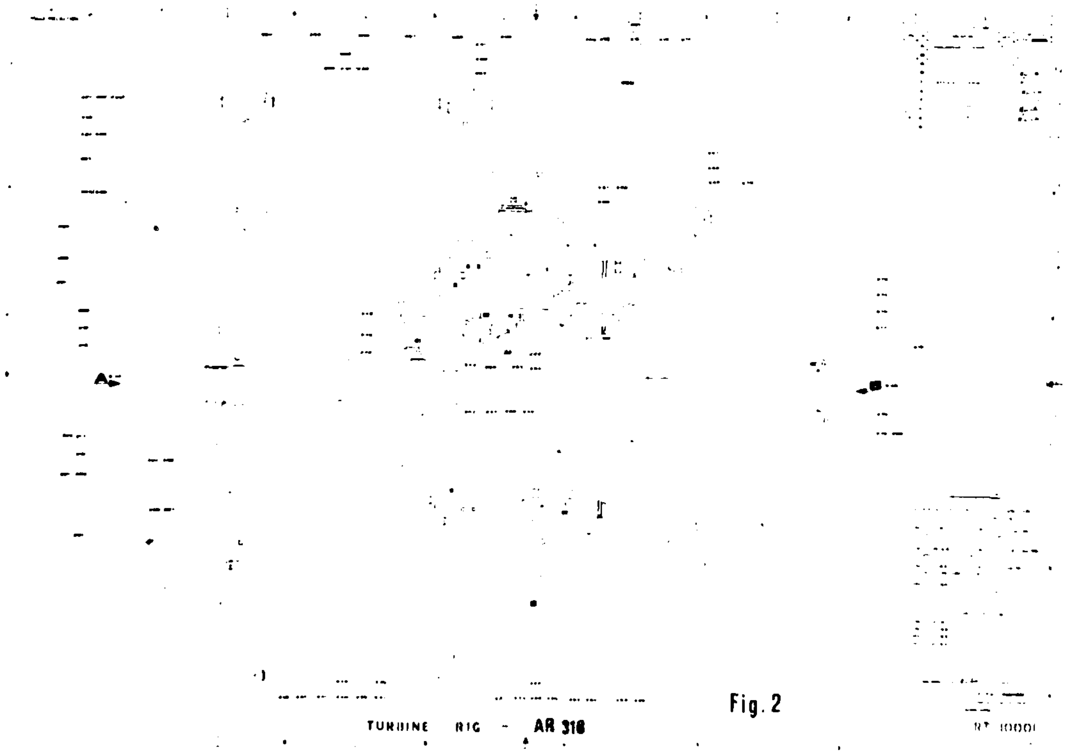


Fig. 2

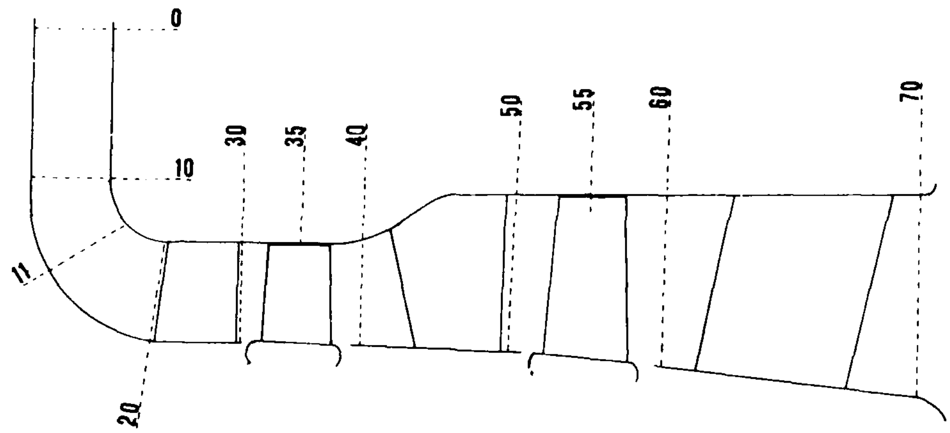


Fig. 3

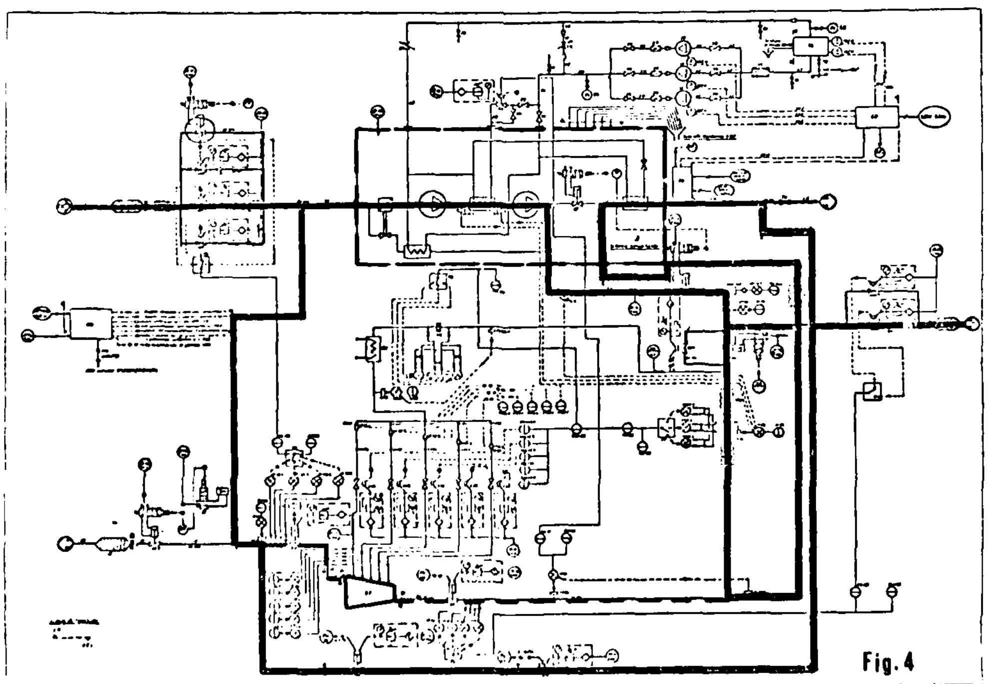


Fig. 4

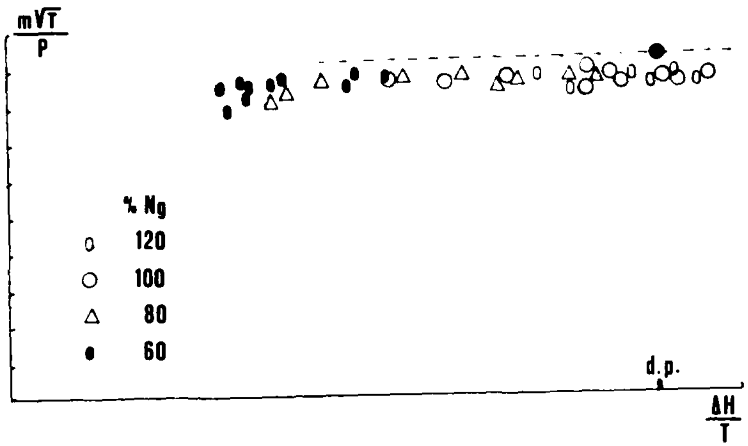


Fig. 5

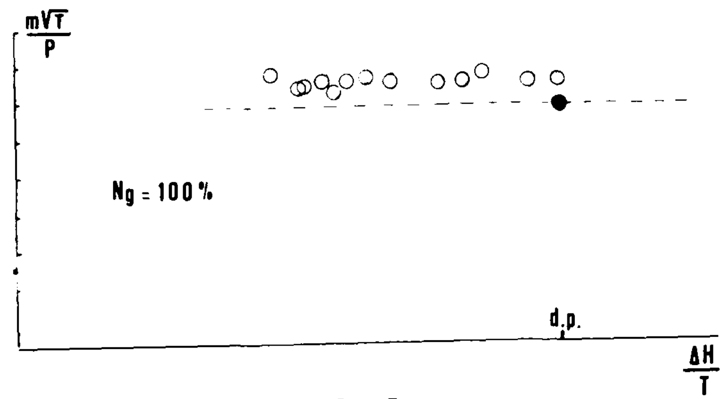


Fig. 7

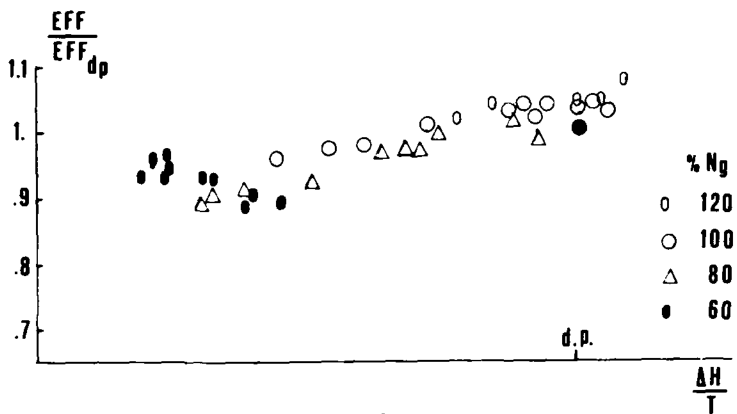


Fig. 6

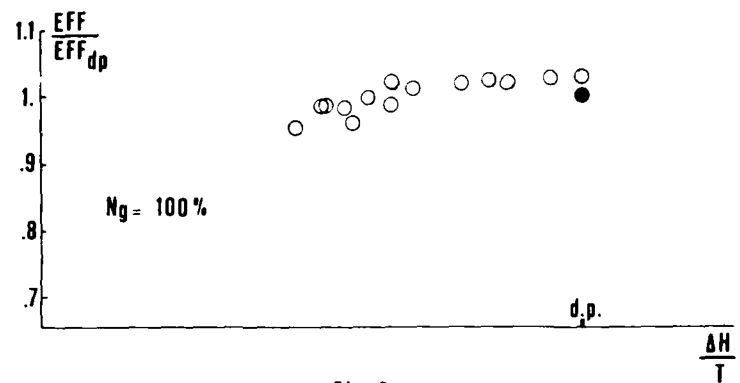


Fig. 8

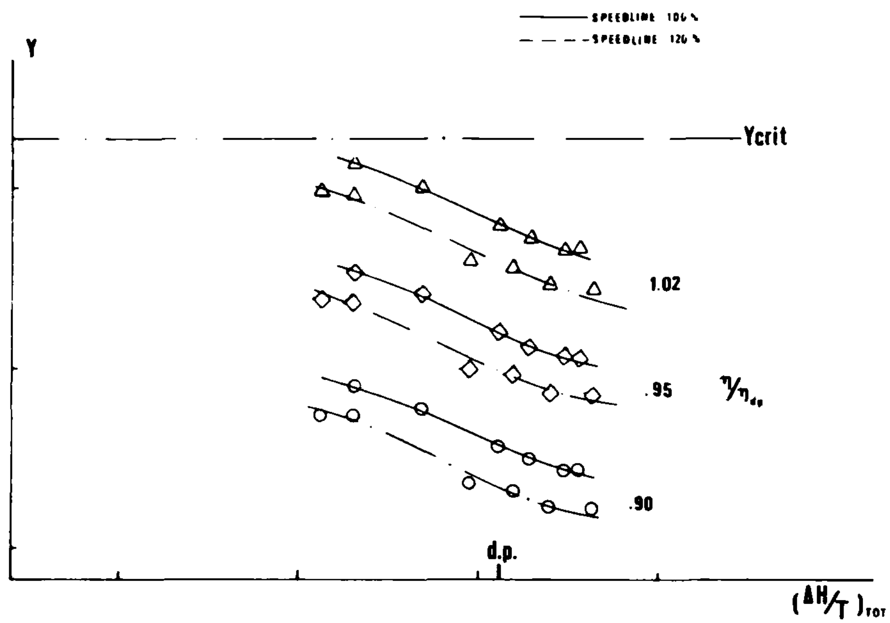


Fig. 9

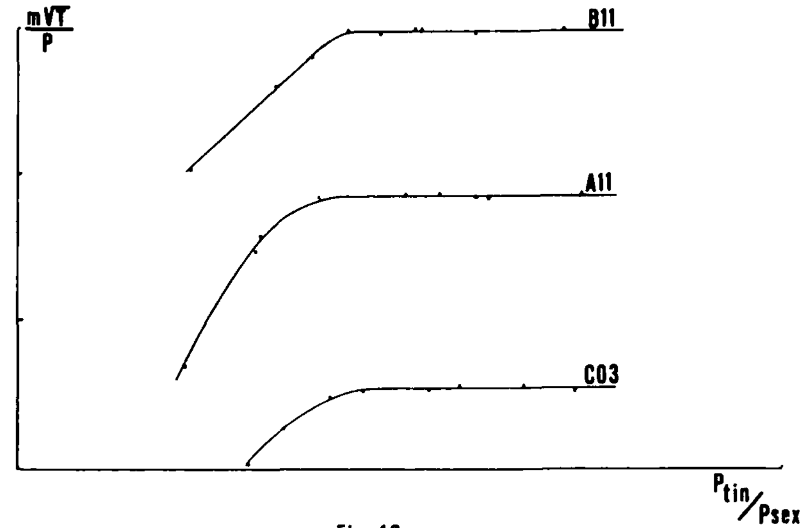


Fig. 10

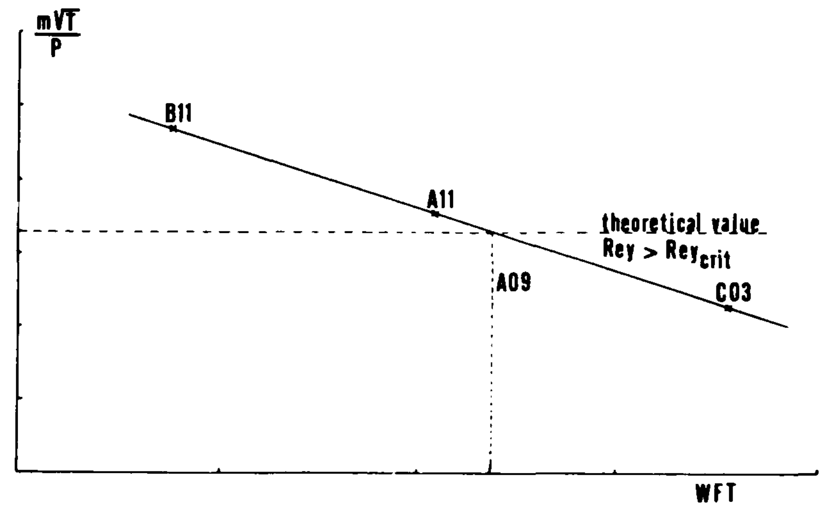


Fig. 11

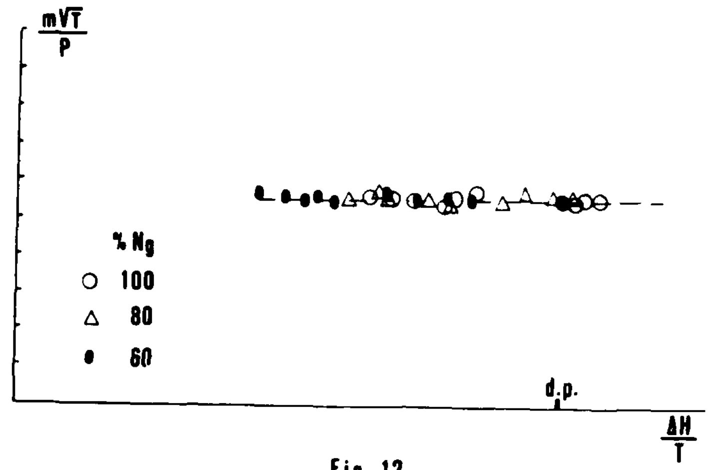


Fig. 12

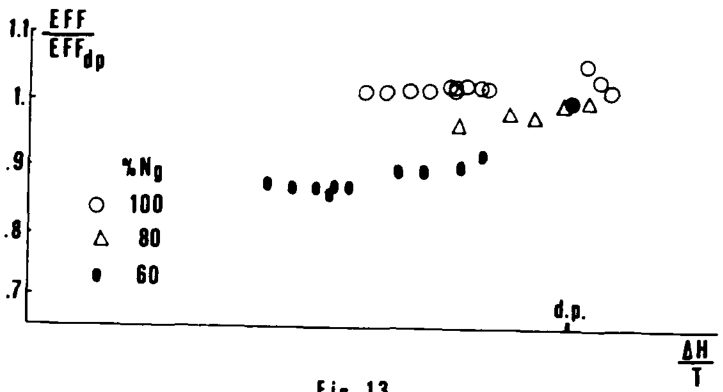


Fig. 13

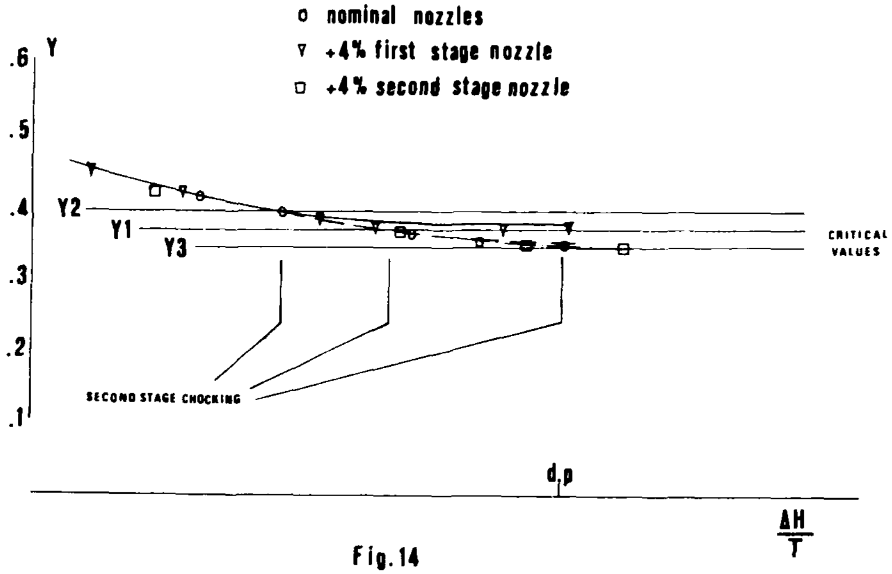


Fig. 14

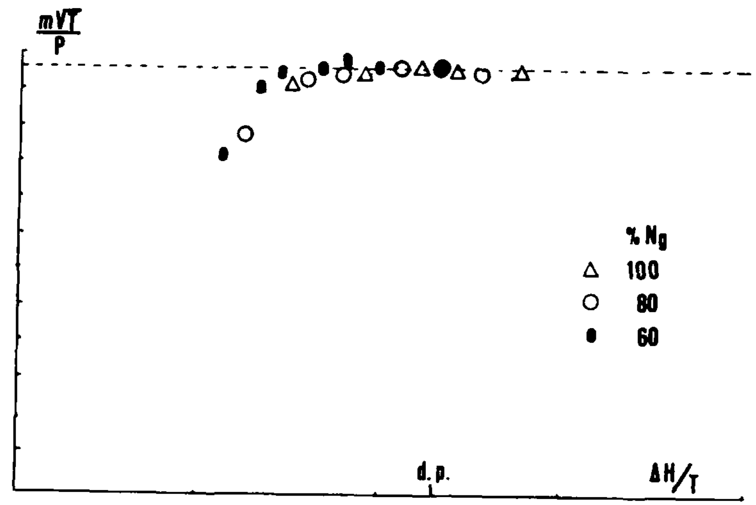


Fig. 15

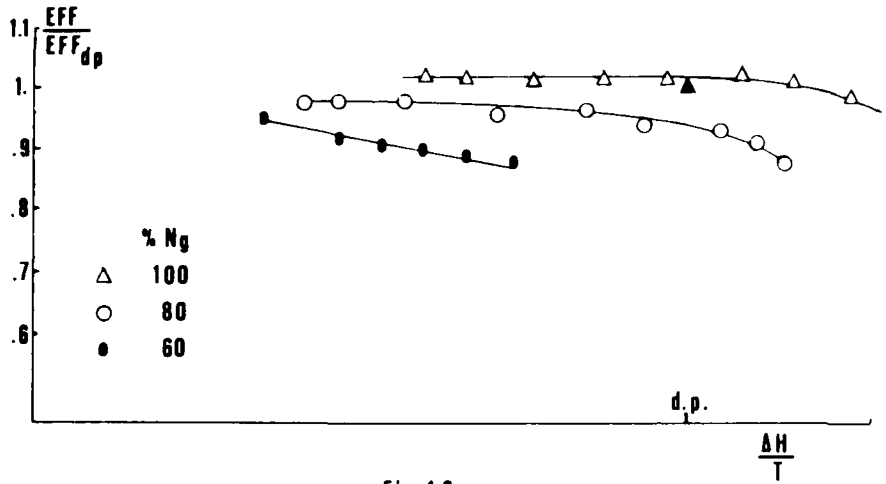


Fig. 16

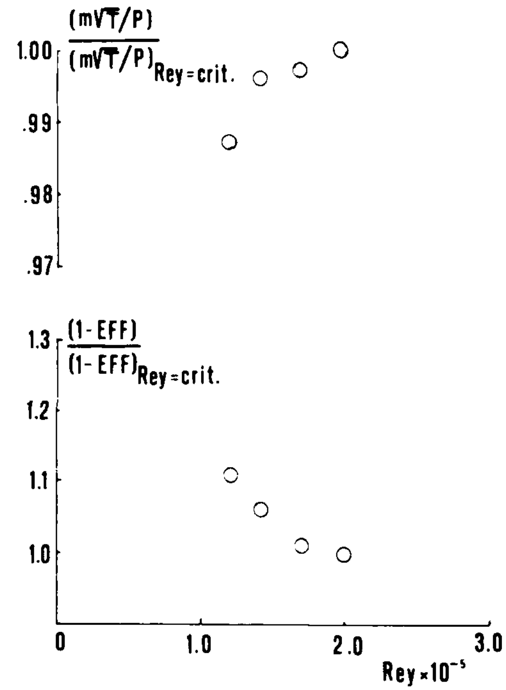


Fig. 17

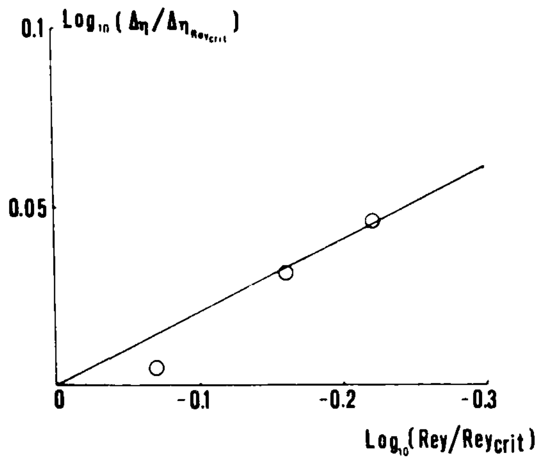


Fig. 18

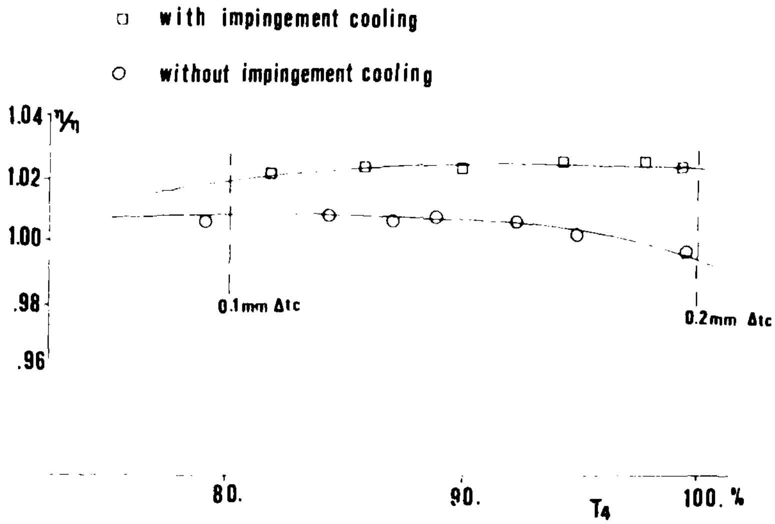


Fig. 19

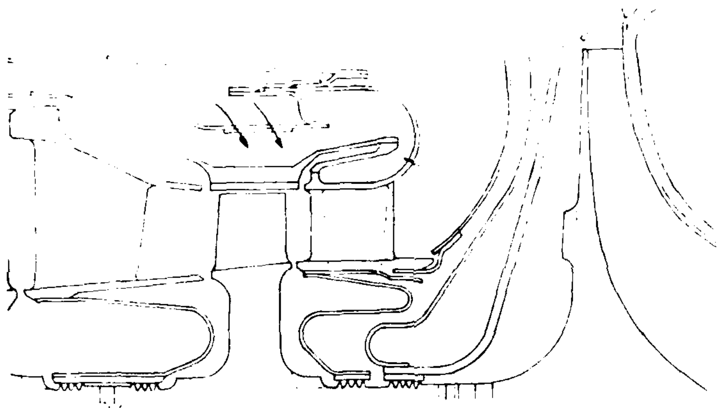


Fig. 20

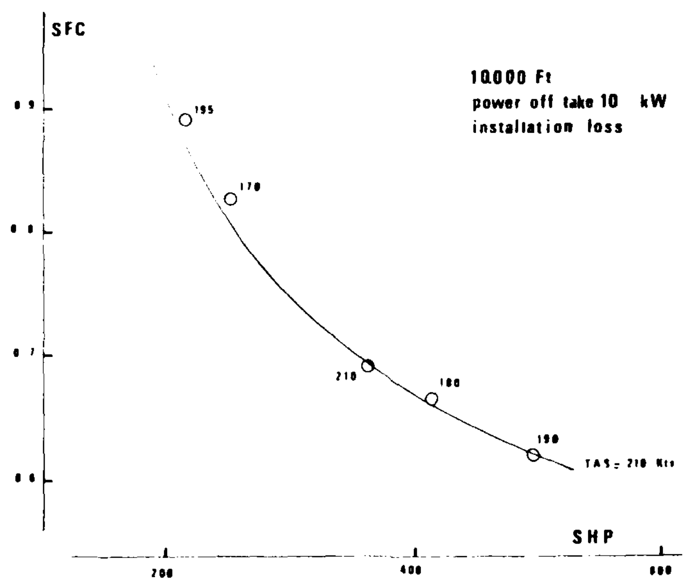


Fig. 21

PROGRAMMA ARP 007 DIAGNOSTICA MONOCOMPONENTE

AR 31H AL RT 1979/77 DIAGNOSTICA

PARAMETRO	DATI	DATO ATTUALE	DATO CORRETTO	DELTA PER CENTO	DELTA PESATI
1	45.230	45.200	45.218	-0.0265	-0.0265
2	1227.500	1231.500	1231.692	0.3415	0.3415
3	361.700	370.000	370.290	2.3750	2.3750
4	2.482	2.456	2.458	-0.9581	-0.9581
5	554.000	555.000	555.023	0.1867	0.1867
6	850.000	852.000	852.115	0.2649	0.2649
7	6.060	6.150	6.151	1.5013	1.5013

1	1227.500	1231.692	1231.676	0.3567	.3567
2	351.700	370.290	370.456	2.4208	2.4208
3	2.482	2.458	2.459	-0.9531	-0.9531
4	554.000	555.023	555.024	0.1844	.1844
5	850.000	852.115	852.261	0.2661	.2661
6	6.060	6.151	6.151	1.5035	1.5035
				SOMMA DI DELTA	1.2937

COMPONENTE	SCARTO QUADRATICO MEDIO	VARIAZIONE PERCENTUALE
3	0.602053E+00	-.144393E+01
5	0.620571E+00	0.412324E+00
6	0.711356E+00	0.144014E+01
4	0.783841E+00	0.610356E+00
2	0.914900E+00	0.114100E+01
1	0.116645E+01	0.150463E+01

IL COMPONENTE DETERIORATO RISULTA ESSERE :

LA TURBINA CON UNA VARIAZIONE PERCENTUALE DEL -1.4439 DELLA CAPACITA

CONTROLLARE STATORE PRIMO STADIO

Tab. 2

PROGRAMMA AHP 337 DIAGNOSTICA MONOCOMPONENTE [illegible] 1

AR 318 [illegible] 18/4/77 DIAGNOSTICA

PARAMETRO	DATO	DATO ATTUALE	DATO CORRETTO	DELTA PER CENTO	DELTA R[illegible]
1	49.000	48.500	48.534	-0.9417	-0.9417
2	1234.000	1233.500	1233.889	-0.0090	-0.0090
3	367.000	365.800	366.379	-0.1693	-0.1693
4	2.600	2.614	2.614	.5556	0.5556
5	553.000	554.000	554.047	0.1894	0.1894
6	860.000	867.000	867.235	0.8414	0.8414
7	6.300	6.220	6.222	-1.2381	-1.2381
1	1234.000	1233.889	1240.407	0.5192	.5192
2	367.000	366.379	372.137	1.3947	1.3947
3	2.600	2.614	2.619	0.3785	0.3785
4	553.000	554.047	554.078	0.1950	0.1950
5	860.000	867.236	872.454	1.4442	1.4442
6	6.300	6.222	6.227	-1.1610	-1.1610

SUM DEI DELTA [illegible]

COMPONENTE	SCARTO QUADRATICO MEDIO	VARIAZIONE PERCENTUALE
6	0.662848E+00	0.484750E+00
4	0.850254E+00	0.309471E+00
3	0.892323E+00	-.672370E+00
2	0.895271E+00	0.612023E+00
5	0.914991E+00	0.325438E+00
1	0.978613E+00	-.252441E+00

IL COMPONENTE DETERIORATO RISULTA ESSERE :

LA CAMERA DI COMBUSTIONE CON UNA VARIAZIONE PERCENTUALE [illegible] EFFICIENZA

CONTROLLARE [illegible] CAMERA

Tab. 3

THE MECHANICAL TESTING OF COMPRESSORS AND TURBINES FOR AIRCRAFT GAS TURBINE ENGINES

D. Norris
Manager, Mechanical Test
Engineering Department
Rolls-Royce Limited
Derby

SUMMARY

Over the past three decades, great advances have occurred in the theoretical appraisals which can be made of gas turbine engine components. Nevertheless the amount of mechanical testing done on such pieces has not declined but has become more extensive and detailed. Major rotating components, whose failure is potentially catastrophic to an aircraft, are the concern of much of this testing, which is aimed at proving and developing the mechanical strength or endurance of the piece.

This paper briefly reviews the principal tests, the techniques and some of the equipment which is employed at Rolls-Royce, Derby, for testing compressor and turbine components. The work relates to about ten different engine types, including the RB 211 family of engines, and to both civil and military applications. A considerable amount of research work also forms part of the activity.

INTRODUCTION

The advent of the computer has led to very significant improvements in the theoretical appraisal which can be made of mechanical designs. This has not however led to an overall reduction in the amount of mechanical testing employed on gas turbine engine components. Indeed the use of alternative materials and more complex designs, more stringent safety requirements and the need to extend fatigue lives has led to more testing being required. The introduction of the large fan engine has also created new testing requirements and therefore led to new equipment and techniques.

A variety of tests are necessary on selected components from one engine type. Therefore the manufacturer of a wide range of engines must be equipped to test very many different types of component in many ways. This is particularly the case regarding the major rotating components. The RB 211 engine has nineteen rotating discs and sets of blades which make up the compressor and turbine assemblies. Some hundreds of tests have been carried out on these components to investigate the design from many aspects. The entry of an engine type into service does not mark the end of this work since fatigue lives still need confirming and extending, tests may be required to support investigations into service incidents and the possible effects of changes in the manufacturing source need to be assessed.

The testing currently undertaken at Rolls-Royce, Derby relates to engine types ranging from those which have been in service for over twenty five years to new engines which have yet to enter service. Research work into new materials, designs and methods also constitute a very significant part of the work. About one hundred and fifty results are obtained annually from the principal tests on compressor and turbine components. Eight test facilities are available for this work and these are in use for twenty four hours a day.

This paper does not attempt to discuss all the types of test or all the techniques and equipment in use but gives a brief account of the principal activities and associated techniques together with a description of a major test facility.

THE NEED FOR TESTING

There are a number of reasons for the mechanical testing of compressors and turbines. Not only does the continued functioning of an aircraft engine - and possibly therefore the safety of an aircraft-depend upon the structural integrity of major components but the high energy of the fragments resulting from the failure of a component rotating at speed is potentially very dangerous. Demonstration of adequate mechanical integrity is therefore required by the Safety Authorities. To meet this requirement, Overspeed, Bird Ingestion, Blade Containment and Fatigue testing may be done on selected components or assemblies from the compressors and turbines of an engine.

After certification the use of alternative materials, new suppliers of material, different manufacturing techniques, design modifications etc. often makes further testing necessary during the development and service phase of engines. Another major requirement for mechanical testing may stem from the philosophy of the cyclic fatigue lifing policy which can depend on the testing of ex-service items for evidence to enable increases to be made in the declared service life. Research into design innovations for future engines such as the use of new materials or new methods of construction together with the need to obtain information to support future theoretical predictions of the behaviour of components under load also necessitates mechanical testing.

In order to be able to impose and control the conditions necessary for the proper testing of components it is necessary to use test rigs. Conditions which range from those usually encountered in an engine to those applying in extreme situations such as blade containment can thus be imposed on components. Such

tests are quite impractical using an engine. A test rig offers good control over the test conditions and gives rapidly obtained and relatively inexpensive results.

SOME PRINCIPAL TYPES OF TEST

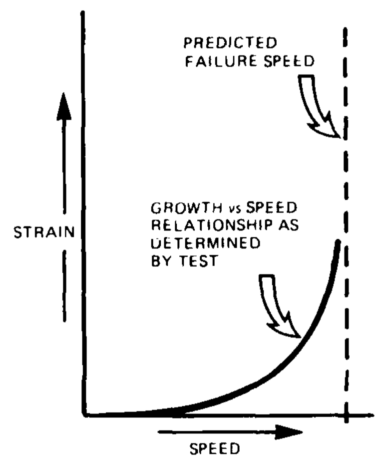
1. Overspeed Testing

This work is aimed at assessing the plastic behaviour and the ultimate strength of compressor or turbine discs.

It consists of running the test assembly at maximum engine speed and above and recording the permanent strain corresponding to those speeds. The purpose of the test is to demonstrate whether the growth of the component is acceptable up to the overspeed condition stipulated by the Safety Authorities. The test may be continued to the failure of the components for further information.

Prior to test the mechanical properties of the test piece are compared with the minimum specification properties for the material and the test speeds are factored accordingly. In this way the weakest disc which may be manufactured to that design is cleared by the test.

The component is tested for five minutes at each of the selected test speeds and is then dimensionally inspected in order to measure any permanent strain. The test may be terminated before failure of the disc following an examination of the dimensional changes. This type of test may be on a single disc or on an assembly of discs - such as a compressor drum. Such an assembly offers a very representative test since interstage spacer loads are fully represented and with care a number of the discs which make up the drum may be evaluated simply by removing the blades from each disc when it is evident that a reduction of load is necessary if a burst disc is to be avoided. The remaining, bladed, discs may then be tested further.



2. Fatigue Testing

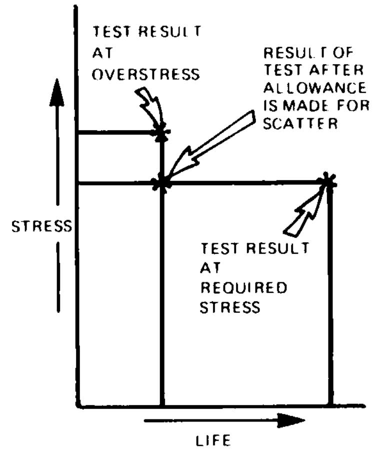
The determination of the safe cyclic fatigue life of components is a major testing activity. Safe operation, adequate spares provisioning and economic operation of gas turbine engines can only occur if the necessary replacement intervals of components are first established. Testing allows such lives to be predicted. Fatigue testing is also a valuable design aid in that Stress vs Endurance Curves for new materials may be determined from components subject to a representative stress field, and changes in disc or blade geometry or alternative manufacturing techniques can be evaluated both qualitatively and quantitatively. It is important that components which are used for test fully represent those components whose service fatigue life is to be established. This not only applies to the design of the components but also to the manufacturing processes, so that the geometry, metallurgy, surface finish etc. are all representative.

The tests basically consist of accelerating the component from a low speed to a speed chosen to generate a required stress level within the component and then reducing the speed to the low level. One fatigue cycle is thus imposed on the component. The test is conducted at a temperature which is consistent with the engine condition to be represented. Many thousands of fatigue cycles may be imposed on the component before the first evidence of cracking occurs. Further testing may then be done in order to investigate the design of the component with regard to crack propagation.

Consideration of the speed and temperature conditions to which compressors and turbines are exposed within an engine usually indicates the need for more than one test in order that the life of a disc be properly evaluated. For instance the peak cyclic stresses in either the rim or the bore of the disc may limit the fatigue life of the component and a test would then be required for each area. In order to generate the desired stress range in those areas of the disc two quite different tests may be necessary. As an example, the rim test would require blades to be fitted to the disc and may be run close to the relevant engine speed and temperature, but a test on the bore of the disc may not have blades fitted and would then be run at a much higher speed and probably at a different temperature.

The predominant stress cycle is usually at aircraft take off conditions, other points in a service cycle either giving no or negligible fatigue damage or else being accounted for in the interpretation of the results of the test.

In order to cater for the scatter which occurs in the fatigue properties of a material the stress range may be factored up from that predicted within the engine, or the resulting life found from the test is factored down, or a combination of these approaches may be used. The result of the test is interpreted using the assumption that the sample tested is the best specimen of those which it represents. More than one sample may be tested in which case the best test result can be used.



To achieve the correct stress in the part of the component under test it may be necessary to test an assembly. In that way the influence of adjacent discs on the stresses in spacers can be achieved and the spacer loads can be correctly imposed on a disc. It may also be expedient to test an assembly in order to rapidly identify highly stressed areas. Such areas can then be further assessed by tests designed specifically around that part of the assembly. Alternatively a single disc may be adequate for the purpose of the test. It is common practice to modify test components however, whether single discs or assemblies, such that the purpose of the test can be achieved. For example the rim features of a disc might be cut off so that they do not limit the testing of the disc bore. A test on the rim features of a disc however would require blades to be fitted to the disc. These usually are dummy blades, made specially for test purposes and featuring a simple block of metal in place of an aerofoil. Such blades can easily be designed to give the required rim loading, they reduce the power required to rotate the disc thus shortening the test, and also offer an advantage in the control of the test temperature.

Fatigue tests are done incrementally. After each increment the component is fully inspected, by various techniques, for cracking or for the extension of cracking. Other damage may occur such as the fretting of blade fixings and fatigue tests can provide a convenient vehicle for the assessment of treatments to prevent this.

3. Bird Ingestion Testing

The purpose of bird ingestion testing on a test rig is to investigate the effects on engine components of an impact by a bird. This usually is either for investigative development work or for engine certification by single bird impact. Multi bird impact certification testing is done using an engine.

Bird impact damage is usually confined to the front stage or stages of the compressor section of the engine. On large modern civil aircraft engines such as the RB 211 type, the LP compressor (or fan) is the component most exposed to bird strike. Such a component may be tested in two ways:-

(a) The complete fan assembly can be rotated in the test rig and the bird fired at aircraft forward speed into the blade.

or

(b) A single blade can be mounted and rotated in the test rig and the bird dropped under gravity into the path of the blade. Adjustment to the blade incidence and rotational speed is necessary to achieve the correct relative velocities.

The first method fully represents the service situation and is used for Certification Clearance testing but the second method is cheaper and offers advantages for analysis of the event. The test is observed using high speed cine cameras.

The radius at which the bird is ingested, the bird speed and weight, and the rotational speed of the rotor are all possible variables, but testing is usually at the most adverse conditions for the weight of bird under consideration.

The result of such a test may extend to permanent distortion of the blade leading edge or in an extreme case to failure of the aerofoil.

FAN BLADE AFTER BIRD IMPACT

4. Blade Containment Testing

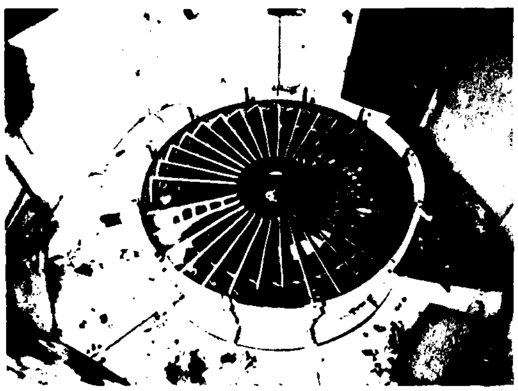
It is necessary, in the event of the release of a compressor or turbine blade, to demonstrate that the debris will be contained within the engine.

The strength of casings in this respect can often be satisfactorily demonstrated by calculation and from experience but if this is not possible, as for example in the case of the use of a new containment material - then testing is necessary.

Recent work at Rolls-Royce, Derby has been directed at investigations into the containment aspects of fan blades for the RB 211 series of engines. Such tests involve the mounting of the containment casing around the fan assembly within the test rig and the blade is released at the required test speed. Great care must be taken to ensure the best simulation of the engine. The position of the blades relative to the casing must be correct at the test speed under vacuum conditions; the rotor assembly must be built to a representative standard; temperature conditions must be correct if this is a significant aspect and the casing must be properly mounted.

Because of their nature, these tests are very destructive and expensive. On large components, blade off loads are very high (e.g. 70 tonf) and damage to the test facility can be significant.

Analysis of the cine films taken during the test, together with a detailed examination of the damaged components, yields valuable information. Those areas of the blade which cause most damage to the containment ring can be identified and assessment can be made of the energy of any pieces escaping from the containment area.



TEST CHAMBER AFTER
CONTAINMENT TEST

FAN BLADE AFTER
CONTAINMENT TEST

THE TEST FACILITIES

The test facilities at Derby are varied in their type and all the compressors and turbines from Derby based engines can be tested, either as full assemblies or as sub assemblies. The various requirements of the tests calls for different test rig designs but some features are common to the eight test rigs. They all are electrically powered and employ electrical braking and the test chamber can be evacuated of air. There are two principal types of rig.

TYPE 1

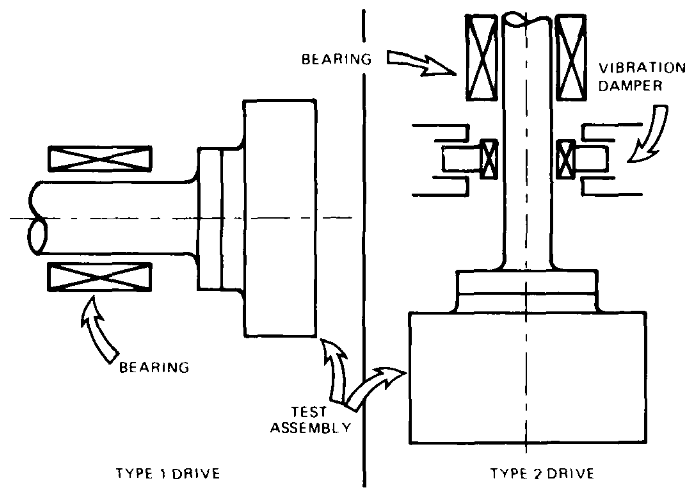
Those rigs having a short 'rigid' final drive shaft carrying the test assembly.

The first fundamental whirling frequency of these rigs is intended to exceed the test speed range. This type of rig is used for tests on short assemblies having an approximate length to diameter ratio of less than 0.5. They are also the rigs used for tests where the centre of rotation of the assembly must be constrained or where large unbalanced loads may occur - as for instance - on bird ingestion or blade containment tests.

TYPE 2

Those rigs having a vertical 'flexible' final drive shaft carrying the test assembly.

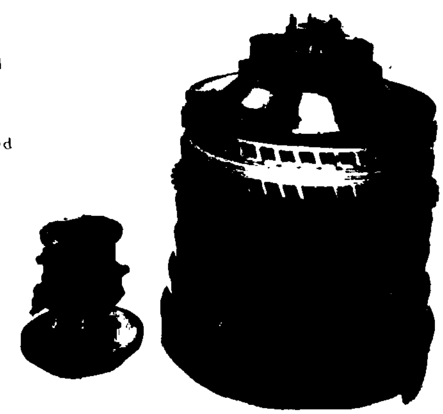
The first fundamental whirling speed of these rigs is very low and the test speed ranges are intended to lie between the first and second fundamental frequencies. Test assemblies having a high length to diameter ratio can be accommodated. This allows for the testing of complete compressor drums which may have a length to diameter ratio up to 3.
These rigs are primarily used for fatigue testing or overspeed testing where no large unbalanced loads are expected.



Many different types of engine generate a wide variety of test assemblies. These may be summarised as follows.

COMPONENT	POSSIBLE NUMBER OF STAGES IN TEST ASSEMBLY	APPROX. DIAMETER RANGE	APPROX. LENGTH RANGE	TEST SPEED RANGE
Fan	1	74" to 84"	8" to 10"	0-5000 r.p.m.
Compressor	1 to 12	8" to 43"	1" to 50"	0-25000 r.p.m.
Turbine	1 to 3	9" to 46"	1" to 14"	0-25000 r.p.m.

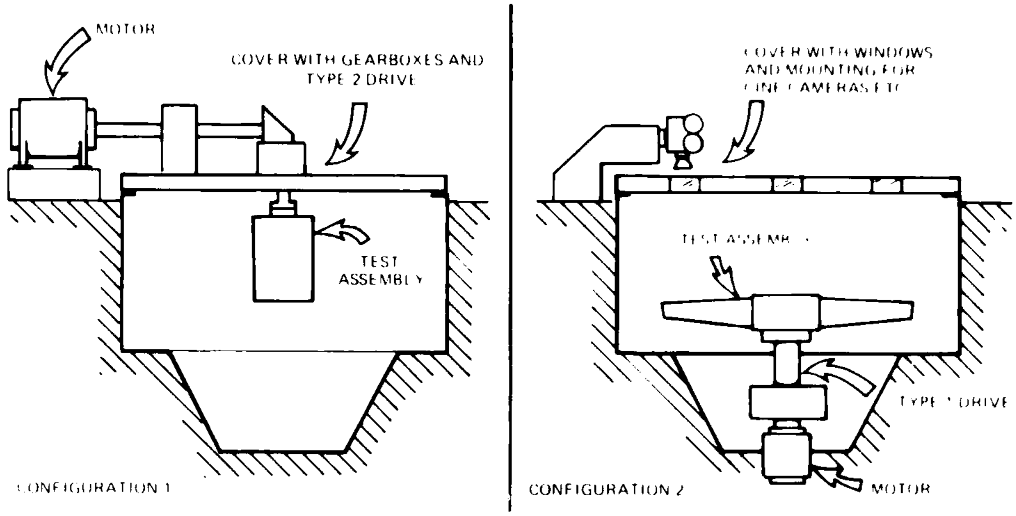
The range of rigs which are available for testing such components include four rigs of type 1 and four of type 2. The rigs are generally capable of speeds in excess of 25000 r.p.m. and motor powers of up to 1000 HP are used. Although the test chamber is usually evacuated of air - and most tests are carried out at a vacuum of 25 torr or less - in order to reduce the power required to revolve the disc and also to minimise dynamic heating effects, the large motor powers are still required in order to provide a rapid rate of speed cycling. Cycle times vary according to the required speed range, the gear ratio, the level of vacuum employed and of course the inertia of the test assembly but they can usually be completed within 10 seconds for small assemblies to 90 seconds for fan assemblies. Dwell times at top speed can be applied if necessary for the investigation of creep phenomena. Most of the rigs can be fitted with slip rings for use with straingauged or thermocoupled test assemblies.
The test chambers of the rigs are lined with containment material both to protect the vacuum chamber from damage in the event of a disc failure and also to minimise secondary damage to the disc fragments. The fracture faces of a piece of disc can yield valuable information on the reason for the failure. This containment material usually takes the form of aluminium blocks, but other materials such as sand has been used. The energy released following the failure of a disc is very great and large pieces of discs can penetrate steel plates to a great depth. Precautions were taken during the design of the rigs to reduce the risks of explosion, such as could occur following the loss of oil into the test chamber during a test at high temperature,



TYPICAL TEST ASSEMBLIES

and procedures are operated in order to cool the rigs before admitting fresh air (oxygen) to the test chamber at the end of a test increment.

All the rigs are fitted with duplicated speed monitoring equipment which incorporates an overspeed detector to minimise the consequences of electrical control failure. The speeds can be manually controlled for non cyclic testing or the rigs can be set to cycle automatically between predetermined speeds. One major test facility incorporates both the characteristics of Type 1 and Type 2 designs and can accommodate components of at least 84 inches diameter. It can be used in 2 configurations.



It can be seen that alternative covers are used for the test chamber, which is some 14 feet in diameter. One cover carries the gearboxes and drive system used with the 1000 HP (Type 2) drive and the other cover is a plate, which is fitted with the mounting features required for cine cameras and the bird ingestion equipment, and which incorporates glass windows. The bottom drive shaft is capable of speeds in excess of 7,000 r.p.m.,is approximately 8 in. diameter,and must withstand the loads imposed following the release of a fan blade. The out of balance loads following such an event can be as high as 100 tonf. The top drive shaft is approximately 1.75 in. diameter and passes through a damper fitted below the bearings. The damper serves to limit the excursion of the shaft when the speed passes through the first critical speed. No large imbalance loads are anticipated in the design of the top drive, which is capable of speeds in excess of 25,000 r.p.m. The test chamber may be evacuated down to about 5 torr, at which vacuum level less than 200 HP needs to be delivered in order to spin a RB 211 size fan at 3900 r.p.m. The control room, which is remotely situated for safety, contains all the necessary instrumentation and Thyristor equipment is used for motor control. Duplicated speed monitoring is provided on the final drive for both configurations and the time for one revolution of the bottom drive can be measured.

TEST TECHNIQUES

HEATING

Testing of components at elevated temperature usually requires the provision of heating equipment, although dynamic heating, particularly when engine blades are used, may be sufficient to achieve the required temperature. The distribution of temperature is also important and dynamic heating alone offers little or nor control over this aspect. Simple electrical resistance heaters are therefore frequently used either to improve the distribution of heat input or also to further elevate the temperature of the test assembly.

The heaters are mounted around the test assembly and separate control of individual heaters is provided. In this way the test assembly can easily be heated uniformly but if large temperature gradients are required then heat shields or cooling equipment is provided. Testing at temperatures up to 650°C is usual and thermal gradients of 20°C per inch can also be achieved in this way. Larger gradients require a more localised and potent technique and then eddy current equipment is installed. The use of electrical resistance heating which is cheap and reliable, limits the vacuum level in the test chamber to about 25 (torr). At higher vacuum,difficulties can occur with the electrical insulation.

TEMPERATURE MEASUREMENT

Accurate temperature measurement poses a real difficulty on this work. The ideal method would be to attach thermocouples to the test assembly and hence directly monitor the temperatures. However, this is impractical since welding to fatigue specimens is unacceptable, and whilst adhesive bonding can be used, it is frequently not a viable technique in view of the timescales in which the test must be done. Various alternative methods have been explored and the currently used technique employs the following temperature sampling approach. Thermocouples are mounted in fixed positions around the test assembly and other thermocouples are provided which can be brought into contact with the test assembly. The assembly is rotated at a steady speed and heat applied until approximately the required temperature is achieved. This is indicated by the fixed thermocouples. Those conditions are then maintained for 2 to 3 hours until it is considered that temperature changes within the assembly have ceased and that it is 'heat soaked'. The

assembly is then rapidly stopped and the heaters turned off. The movable thermocouples are brought into contact with the test assembly and their output recorded. Analysis of this output is then made and after correcting for certain known losses, the temperature of the test piece is derived. If necessary the process is repeated using different heat inputs until the required test temperature is achieved. The fixed thermocouples are then used for temperature monitoring during test. Whilst this procedure is lengthy - some eight or ten hours may be required to establish the required temperature when first starting a test, the technique offers a reasonable accuracy and is relatively simple.

Nevertheless a technique offering a constant and direct temperature measurement of the test assembly is still desirable in order to reduce costs. Development is in hand of pyrometry equipment, for this purpose.

CRACK INSPECTION

Fatigue testing is basically directed at establishing the life to the first cracking of a component. It is clearly implicit in such testing that frequent and detailed inspection for cracks must be done.

All components are inspected as part of the testing procedure not only after increments of testing but also before test. Whilst new components are crack inspected as part of the manufacturing process, it has proved valuable to repeat this process with the equipment and personnel who will subsequently inspect the test pieces after testing. In this way, qualitative inspection comment and continuity of knowledge on the component is achieved.

Test pieces have a binocular inspection, at up to X30 Magnification, over all their surface but with particular reference to the test area. Supporting technique include eddy current,ultrasonic, and penetrant dye inspections. All defects or indications of possible defects are recorded at each inspection. The etching of components in order to improve the quality of the inspection must be used with discretion. Components which have been subjected to fatigue can have their lives prolonged if fatigue damaged material is removed by etching. In a similar way etching should not be done on test surfaces which have a certain treatment such as vapour blasting. Inspections of the component are sometimes done whilst it is still fitted to the test rig. One technique that has been used in this way consists of introducing a penetrant dye to the surface of the rotating test piece. Thus a crack, which may be tightly closed, when the component is stationary, can be identified for subsequent investigation.

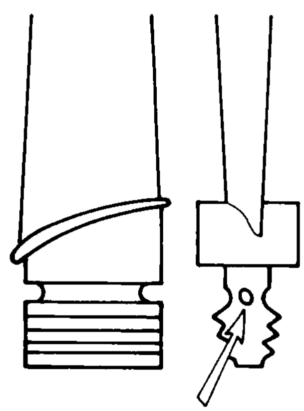
BIRD INGESTION

As referred to earlier, bird ingestion tests may be on a single blade or on full disc assemblies of blades. In both cases a real, or gelatine dummy, bird must be introduced into the path of rotation of the test piece. With the single blade, the bird is dropped under gravity from a suspension filament severed by a detonator. For the full assembly, the bird is fired by air pressure from a gun following the bursting of a diaphragm. Accurate coordination of the bird release with the lighting and camera operation and the rotational position of the rotor is essential on single blade tests and desirable on full assembly tests. This is achieved using a probe which signals to electrical timer boxes each revolution completed by the shaft. With this equipment an event can be initiated within $\pm 15^{\circ}$ of any point during rotation of the assembly at 4000 r.p.m. and the subsequent events filmed on cameras operating at up to 9,000 frames per second.

BLADE CONTAINMENT

As with bird ingestion tests, the coordination of the test is very important and similar equipment is used. In this way the position for blade release can be selected and the events filmed. The position for blade release is generally governed by the placing of the fixed windows through which the events are filmed. Optimum camera coverage is required and so the event is planned to occur in view of the principal cameras. In order to improve the quality of the photography, careful attention is paid to the painting of the test assembly and the test chamber itself, and the blade to be released is painted in sections such that its behaviour can be more easily analysed from the films.

Various techniques have been used in order to release blades from rotating assemblies. These include the original practice of simply weakening the blade such that it would be stressed to its ultimate load at the test speed and a technique whereby weakened blades of suitable material could be heated electrically at the test speed in order to depress the ultimate strength of the material. Neither of these approaches caused the blade to be released at a predictable time and were unsuitable for tests which required detailed analysis of the results such as could only be achieved using high speed cine filming. The technique currently used for blade release consists of weakening the relevant area of the blade such that at the test speed it is stressed to 95% of its actual ultimate stress. Explosive detonators are fitted within this highly stressed area and are fired electrically once the test speed has been achieved. This technique is used when the blade is to be released from the shank. It is this area where sufficient material is available for this approach. If the blade is to be released from the aerofoil then an alternative technique involving an explosive fuse is used. It is this latter method which is used with the hollow fan blade which is currently under development at Rolls-Royce.



DETONATORS FITTED
TO MODIFIED BLADE

FUTURE TRENDS

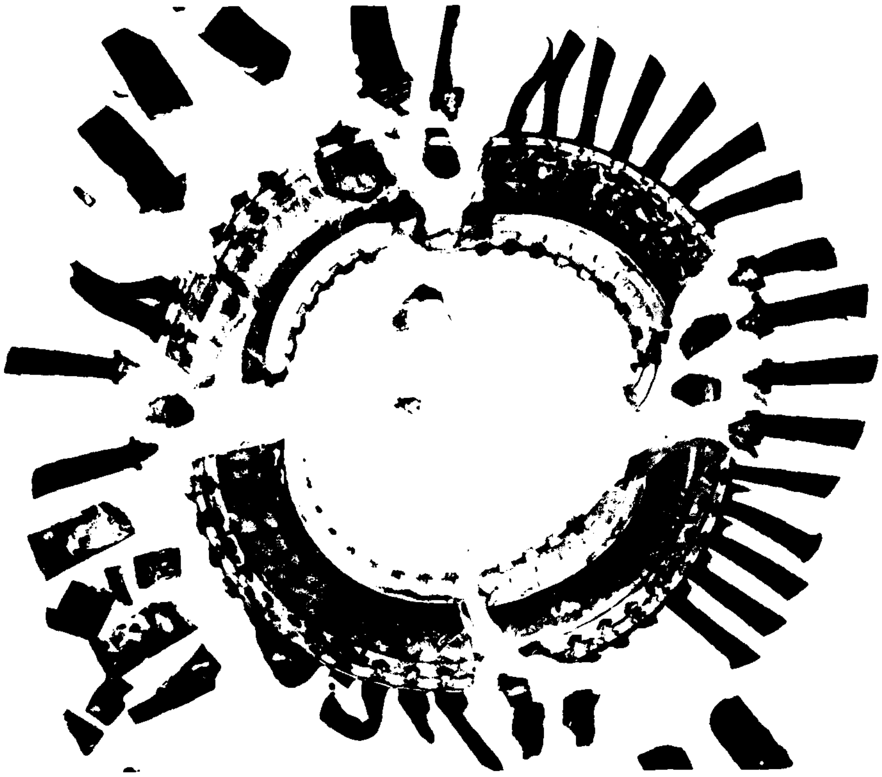
Major advances in the techniques and equipment used on this testing have been made in the past thirty years. These have been necessary in order to keep abreast of the use of new materials and designs and also to satisfy the need for more reliable and accurate test evidence. Further advances can still be made by refining existing techniques and also by developing new ones.

One concern where a relatively new test requirement has led to a developing technique, is that of high cycle fatigue damage. Only low cycle fatigue work has been previously discussed, but high cycle, or vibration, damage can also occur in engines due to aerodynamic excitation. A cross wind situation is a good example of such an excitation relating to fan blades. The current approach for this type of test is to mount, adjacent to the test blades, fixed plates with apertures. As the blades rotate and pass the apertures an aerodynamic excitation occurs and adjustment of the vacuum level and other parameters can be made so as to achieve the desired stress levels. Discs have been similarly tested by arranging discrete jets of air to impinge on the disc rim. This approach has so far proved satisfactory but alternative techniques, such as mechanical excitation methods are under consideration.

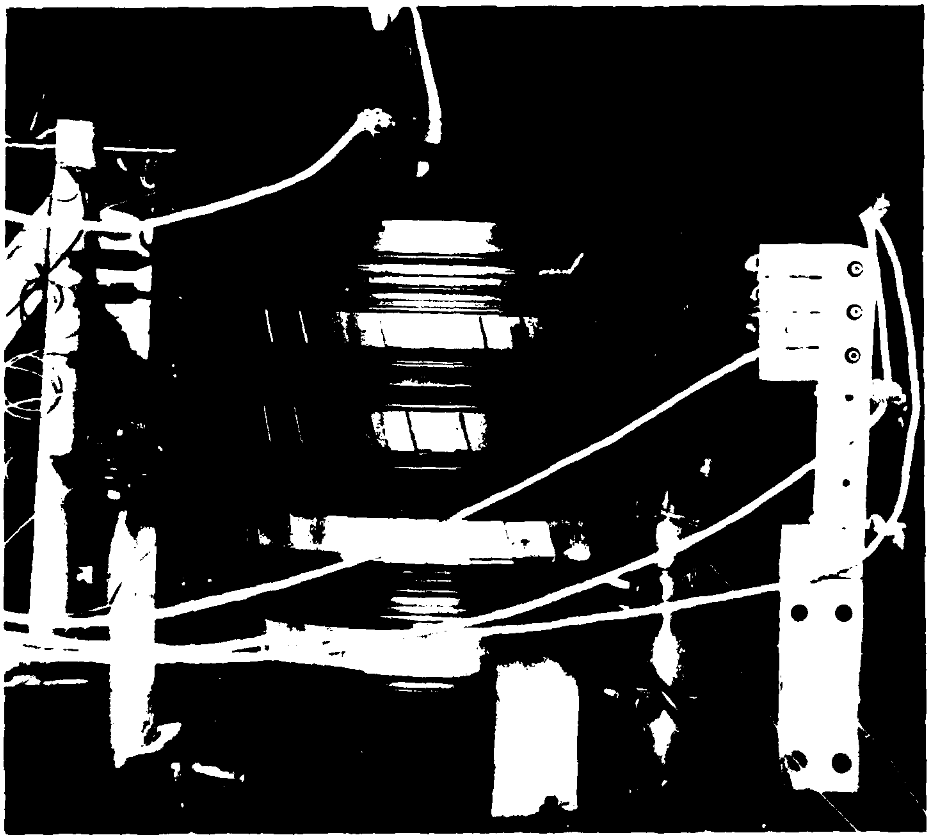
A second important development in this area of testing is likely to be aimed at imposing a better low cycle fatigue test by not only cycling the speed, but also by cycling the temperature of the disc. The full stress range experienced by the disc in an engine would thus be represented. The principal difficulty with this approach is that of cooling the test assembly sufficiently quickly to give a practical test.

Other developments are likely to include changes to the method of temperature measurement by the introduction of pyrometry and fibre optic equipment, and the establishing of acoustic emission as a regular inspection tool. Work on both these techniques is in hand.

In general, mechanical testing will continue to be required, not only in support of theoretical work, but also to help resolve those problems which are not amenable to solution by calculation.

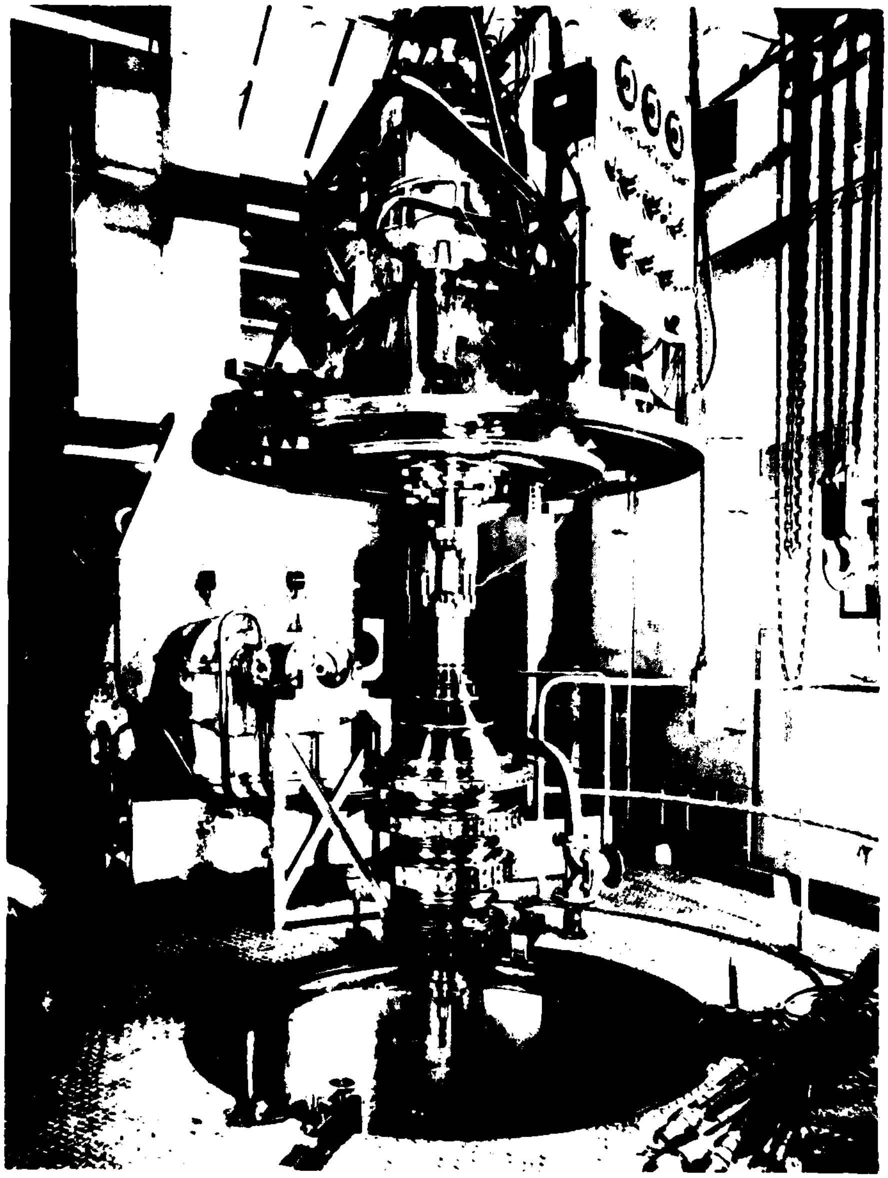


DISC FATIGUE FAILURE

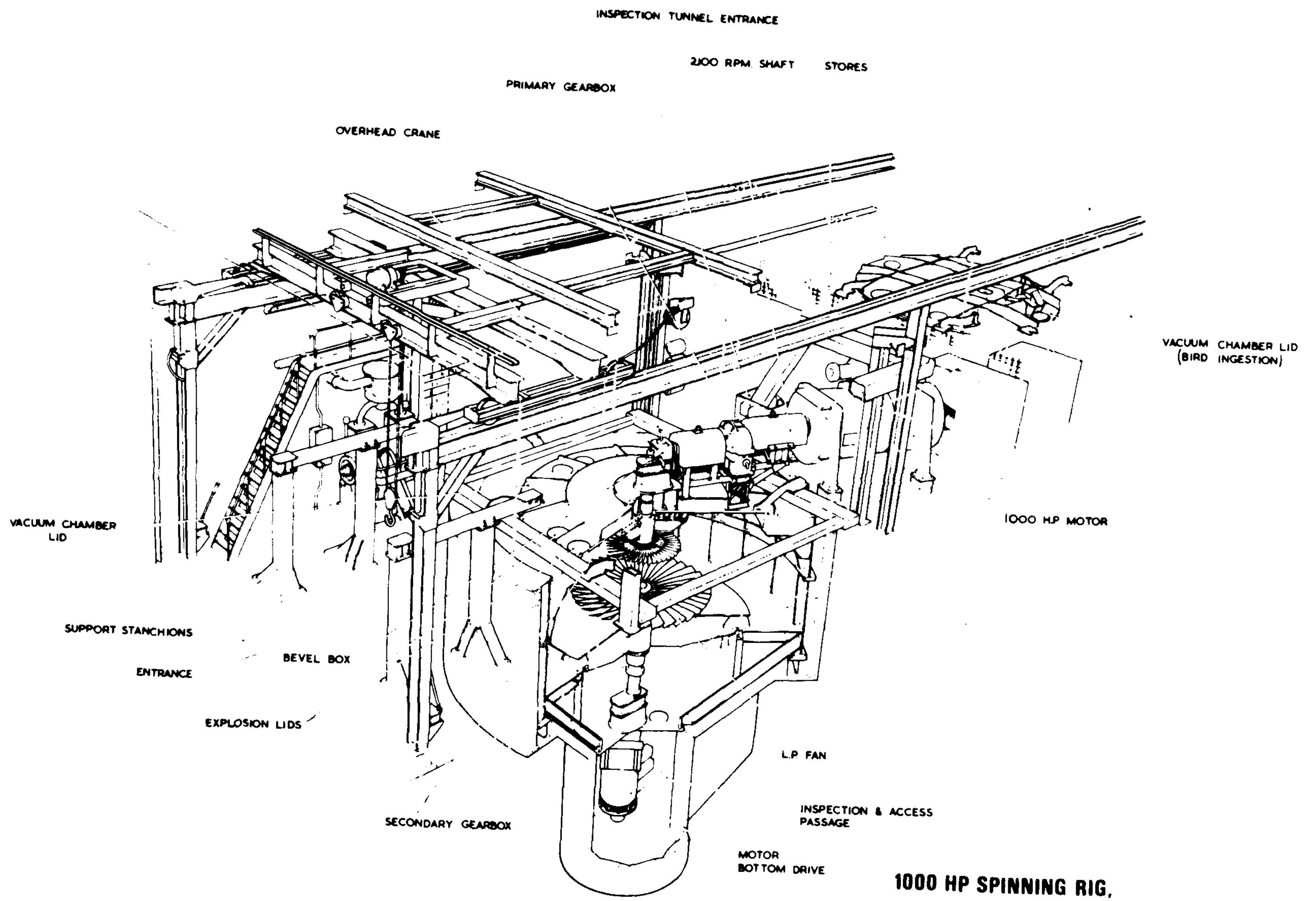


ARRANGEMENT OF HEATERS AND
THERMOCOUPLES AROUND TEST ASSEMBLY

52 10



GENERAL VIEW OF TEST ASSEMBLY
ABOUT TO BE LOWERED INTO TEST
CHAMBER



1000 HP SPINNING RIG,

DISCUSSION

D.K.Hennecke, MTU, Muenchen, Ge

You mentioned that you simulate the thermal loads on the turbine disks. Does that mean that you impose the proper temperature gradients during your tests, and, in particular, do you also run transient, i.e. cyclic thermal tests?

Author's Reply

In some cases it is essential to represent thermal gradients (for example, the bores of nickel alloy turbine discs). It is not practical to represent thermal transients, the cooling time is too long. The effect of the thermal transient can be simulated by adjustment of the cycling speed range, or by correction in the calculations.

HELICOPTER TRANSMISSION QUALIFICATION PROCEDURES AND TESTS

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SUMMARY

This paper will discuss the tests required to qualify helicopter transmission in conformity with the current civil and military requirements, the tests to be conducted for initial development and pre-qualification as the tests essential to guarantee a satisfactory maturity of the product being released for service.

The discussion shall stress the features of each test technique and their significance under the various phases of the program.

Moreover the use of ground test vehicles and intensive flight testing will be pointed out with particular emphasis.

The paper shall also cover the techniques used to assure an adequate degree of confidence to the information obtained from the various experimentations.

INTRODUCTION

The maintenance cost of a helicopter drive system is about 25÷30% of the total maintenance cost. This cost represents a significantly high incidence in the life cycle cost of the helicopter and must be carefully evaluated because in the next years, other parts, such as hubs, blades and controls, will be less expensive because of the application of new and sophisticated technologies.

In addition to new design criterias and new materials, in the future it will be necessary to plan precise test programs designed to improve reliability and thus reduce the drive system maintenance cost.

The purpose of this paper is to describe those tests necessary to obtain both a high level of reliability while satisfying existing requirements.

The test philisophy described in this paper is currently in use at Agusta for the development of new drive systems.

TEST REQUIREMENTS

In Italy the current practice is to apply for the qualification test under the following U.S. regulations :

Military	Civil
MIL-T-8679	FAR 27
MIL-T-5955	FAR 29

Even the latest issues of these documents specify only the demonstration tests and permit the manufacturer freedom in choosing the development tests that may often be restricted both in number and in time because of their costs.

However the AMCP 706/202-203 does suggest some of these kinds of tests.

TEST PROGRAM PLAN

The reliability tests of a system are performed to determine or demonstrate the probability that the system will accomplish a specified task or mission.

The test program will be divided into two main phases : first is for development and the second is for demostration (Fig. 1).

During the development phase the product is tested to determine that both its functional and structural features meet the design requirements, at any time during this phase it is possible to modify parts in order to correct evident malfunctions.

At the completion of this phase the manufacturer can qualitatively predict the reliability level of the system.

The second phase includes the tests established with competent authority which will demonstrate a minimum level of reliability. These tests differ from the preceeding ones because the system configuration may not be changed (Fig. 2).

DEVELOPMENT TESTS

There are two kinds of development tests :
- Initial development tests, to substantiate the design criterias.
- Full development tests, to obtain adequate system reliability information

INITIAL DEVELOPMENT TESTS

Each new technology or new material utilized in the system design will be evaluated by means of suitable research programs.

However, their simul neous applications in a new design can be a risk which must be investigated prior to stating the definitive prototype configuration.

The initial development tests that are designated specifically to minimize the costs incurred from these risks, and are not easily quantified by the design analysis, are the following :
- Elastic tests of the gear blanks
- Lube system tests with and without oil
- Critical points where potential wear problems exist
- Tests to demonstrate producibility.

FULL DEVELOPMENT TESTS

These tests are initiated when either an assemply or the entire transmission system initial development has been completed.

Usually the full development tests are conducted in two different ways:

- Test of one or more assemplies that require specific test benches
- Test of the complete system enmploying :
  - An aircraft secured to the ground
  - A Ground Test Vehicle (G.T.V.)
  - Or a helicopter Bench (Iron bird)

Obviously the maximum reliability is obtained from the second type of tests. However, the disadvantages are that they are very expensive, require too much time, and do not permit the overpower tests which are generally limited by both the rotors and engines.

Those tests which are intended for the purpose of identifying potential malfunctions of components, or subassemplies, thereby requiring correction of the design, should permit to pass qualification by satisfying stated requirements, even though they are not tested in the final definitive configuration.

DEVELOPMENT TESTS ANALYSIS

A test program will assure a successful development of a gear box if it includes:

1) Case static tests
2) Gear box elastic tests
3) No-load Lube-system test
4) Incremental load and efficiency tests
5) Thermal mapping tests
6) Overload tests

1) Case static tests

Static tests of castings are generally of two types :

- Elastic tests at the operative load to measure deflections of bearing housings caused by both internal and external loads.
- Static test at the ultimate loads to prove the adeguancy of the critical case to the required casting factors.

Both of these tests are performed using the same test rig.

2) Gear box elastic tests

These tests are intended to evaluate :

- The gear patterns and design
- The bearings behaviour
- The load sharing in planetary systems
- The maximum deflections of input and output shafts which are critical for seals.

Close-Loop benches are generally used for these tests, which will allow the application of static torque, of external loads and slow rotation.

Another feature of these tests is variable loads and temperatures in order to cover the entire operative range.

3) No-Load Lube-System test

The first assembled transmission will be used to evaluate the behaviour of the Lube-System prior to initiating the load development tests.

For these tests it is useful to have a movable fixture in order to evaluate the windage loss of power and the constant pressure of the oil for different attitudes of the helicopter in normal flight operation.

The normal growth recorded will be a significative mean for the comprehension of hydraulic phenomena because without a load the temperature is a direct function of windage losses.

By means of these tests it is possible to establish the correct shape of oil baffles, the pressure inside the transmission, the vent position, the foaming situation and the preliminary evaluation of rotating seals.

4) Incremental load and efficiency tests

After the no-load lube tests it is useful to perform short functional tests with increasing loads to investigate the actual behaviour of the bearings and the gears.

At the same time it is possible to determine the efficiency of the transmission.

The rigs in this case can be either close-loop mechanical benches or full absorption benches with or without electrical power regeneration.

Modern rigs allow the application of impulsive transient torque to simulate actual function condition for the investigation of the scoring phenomena.

5) Thermal Mapping tests.

Modern transmission of civil and military helicopters are required to demonstrate a fail-safe behaviour in no-lube cordition. By means of special instrumentation such as thermocauples, thermoplackards and thermovision systems, it is possible to create thermal mapping of critical points of the transmission where heat generation could create a risk.

After the examination of the maps, modifications necessary to pass the no-lube tests will be established.

6) Overloads tests

During the prequalification, for a period of 10 ÷ 15% of the total test time, the transmission will be run at an overload torque factor of 1,25 ÷ 1,30 of the correspondent design power. The scope of this experimentation is to produce evidence of the failure modes and the fail-safe behaviour of the rotating parts.

The absence of pitting, scoring, tooth breakage and other catastrophic failure modes shall be demonstrated with the same oil used in service. These tests will become true fatigue tests of rotating parts when overload factors reach the value of 1,3 ÷ 1,4 and are intended to demonstrate the endurance limit of those parts.

The duration will be established to accumulate sufficient number of cycles on the slowest rotating part.

DEVELOPMENT TESTS OF THE WHOLE DRIVE SYSTEM

With the development test of each assembly it is possible to obtain an adequate level of realiability, but there is no information concerning the potential problems of interface with other assemblies.

The development tests of the whole system determine any malfunctions prior to the beginning of the flight tests and tiedown demonstration test.

A tie down helicopter is used to investigate the effect of different operating levels of power and the excursion of flight controls that can cause loads and unpredictable vibrations in the same conditions.

A preculiarity of these tests is the experimental determination that critical flexional and torsional vibration speeds are far from the normal operating speed range and that the behaviour of the engine and transmission mounting and of the hangar tail bearing supports are in agreement with the preliminary vibration survey obtained with the shaking tests.

Dynamic and static balancing tolerance of rotors and shafts are also investigated in a true environment.

The engine installation can be evaluated in order that the air intakes are not effected by the exhaust gas reingestion, which could cause engine stall and damage to the drive system.

A practical substitution of the tiedown helicopter is the "Ground test vehicle" (G.T.V.), that is a helicopter modified for a long and intensive running program on the ground and is operated by remote controls.

The G.T.V. is the most effective device to continue the development on the ground of the transmission system after the qualification tests and to conduct tests in climatic cells to evaluate the behaviour of the helicopter systems in extreme operative conditions.

QUALIFICATION TESTS

Civil and military regulations establish the basic tests to obtain the qualification of a drive system. These basic endurance tests are in particular specified by the military documents such as the MIL-T-8679 and AMCP 706-203.

- A 50 hour test to garantee an adequate level of safety before the beginning of the flight test.
- A 150 hour preproduction test
- A 250 hour test to establish the initial T.B.O. (time between overhaul) and special inspections.

These tests performed on a tie down helicopter or a G.T.V. differ from development tests because major modifications cause the repetition of the test, that is, the configuration is fixed, and time, sequence and policy of the different phases of these tests are governed by requirements.

After this first kind of test the reliability program will be implemented with flight and ground tests to demonstrate a specified M.T.B.F. (Meantine between failure).

GROUND TESTS

The ground tests will continue using regenerative test benches, or a tiedown helicopter or G.T.V. with a qualified transmission. The necessary reliability level will define their duration.

The total duration of this tests will be divided in order to approve different kinds of lubricants and components produced by alternative vendors.

FLIGHT TESTS

The flight tests necessary to complete the qualification program and to develope the load and vibration survey, must be implemented with a program of intensive flights by means of a production helicopter. The purpose of this tests is to collect data about the reliability and maintainability level of the whole dynamic system.

The first flying prototype, instrumented with straingages, accelerometers, thermocouples, etc. will be utilized for the load and vibration survey.

A second one will be utilized for the demonstration tests of handling qualities and functional reliability.

The same helicopter will be used for other activities scheduled as follows :

- Activities recommended for civil helicopters will be performed by the engine manufacturer with a flight test program to develop the efficiency of the engine installation.

  The purpose of these tests is to investigate :

  - Mission profiles
  - Dynamic system torsional stability
  - Decelerations
  - Accelerations
  - Ground starts
  - Engine control system stability

The data furnished by the above tests are useful to eliminate malfunctions in an early stage of the program and to minimize its cost.
After this experimentation the helicopter will be used for the demonstration of the handling qualities by the airframe producer.

- After the flight demonstration tests the helicopter will be employed for intensive flight tests program to accumulate significative flight time in order to garantee sufficient maturity at the commencement of its operative service life.

TEST DATA ANALYSIS

At the end of each test the data concerning visual and analytical inspections will be collected by means of a computer program to elaborate them in order to collate information relating to the frequencies of different kinds of malfunctions and statistical failure rate.

Accuracy in inspection and analysis is very important to assure a true value of test results and to make a correlation with the fault analysis report.

Different kinds of data analysis will be used in each phase of the experimentation program.

A qualitative analysis will be made of each component during the development phase, based upon thorough and varied inspection techniques to determine its "mode of failure" with the conclusion being the identification of the design fault, if any, and its corrective solution. The same analysis criteria will be used also for the initial T.B.O. demonstration.

The reliability demonstration phase and the data accumulated in service, will furnish sufficient data to permit their statistical elaboration.

The "design and testing engineering staff" will directly manage the test program during the development and demonstration phases. But with the great increase of the number of flight hours, after the aircraft has entered service, it's not possible that the same staff can physically handle all the data; for this reason, of extreme importance after the development program are the use of highly skilled mechanics and the proper completion af all forms for thorough review.

CONCLUSIONS

In future test techniques will be implemented to confirm the modes of failure predicted by the design fault analysis rather than discover them. In fact with adequate test program it is possible to garantee the sufficient maturity to a new drive system entering service.

All that will give the customers the economical advantage of low operative costs obtained from the reduction in maintenance incidence.

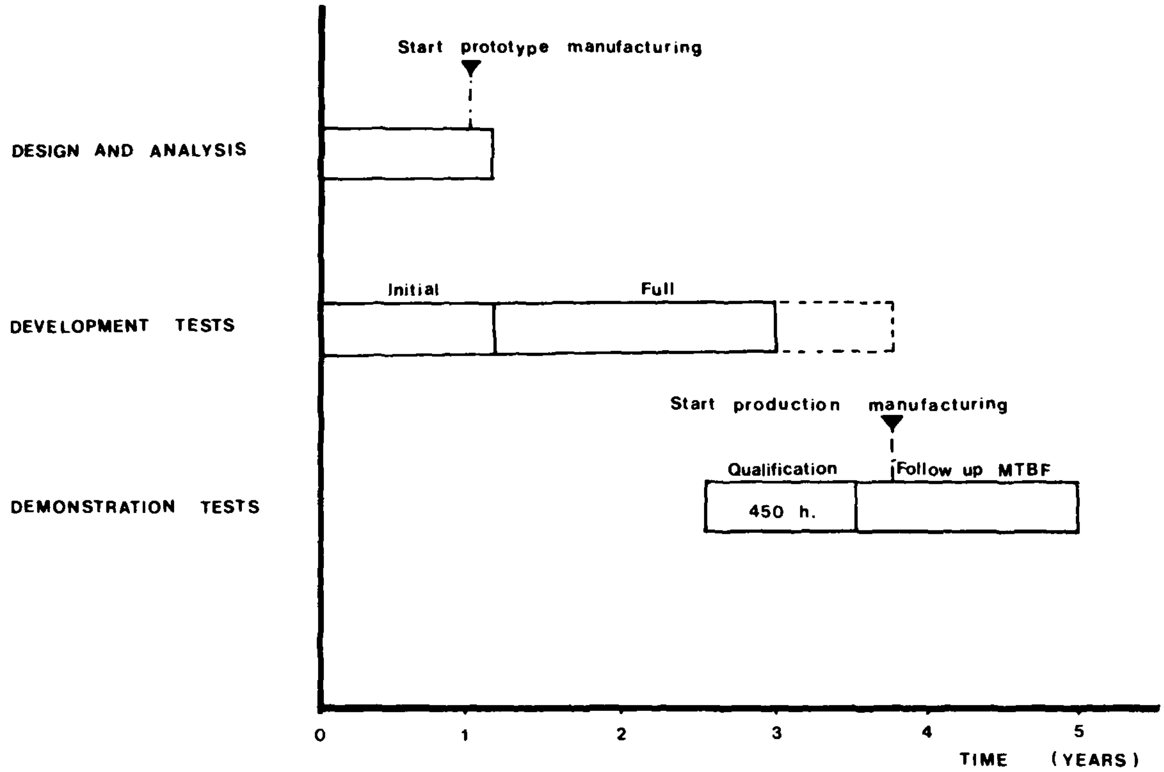

Figure 1. Reliability and testing program.

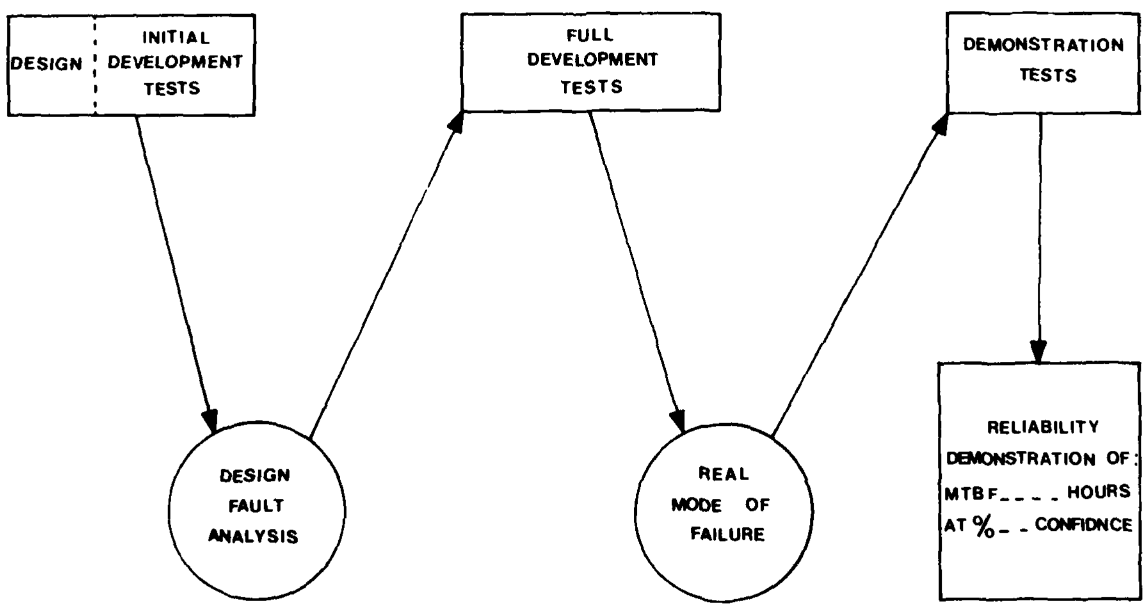

Figure 2. Reliability and testing process.

the risks of explosion, such as could occur following the loss of oil into the test chamber during a test at high temperature,

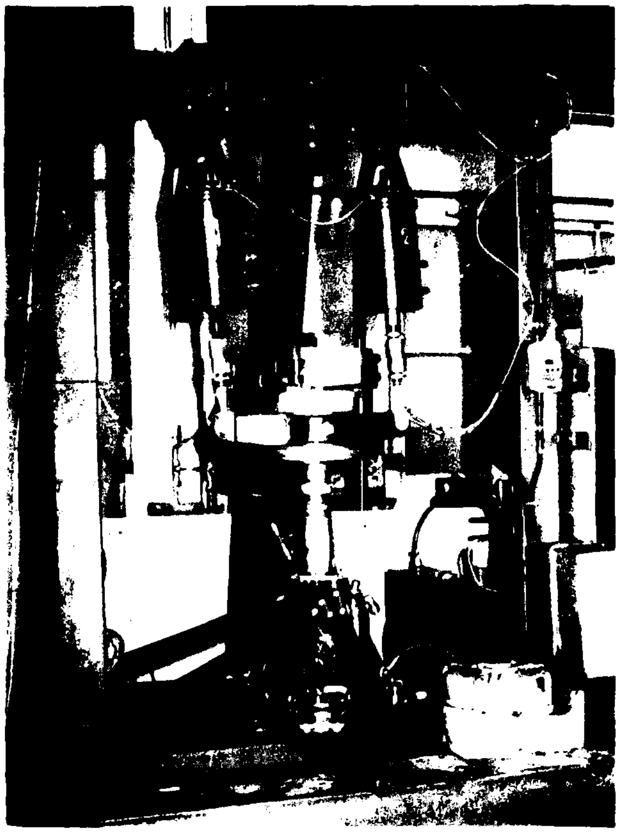

PHOTO 1. A109 MAIN TRANSMISSION MAST BEARINGS TEST BENCH

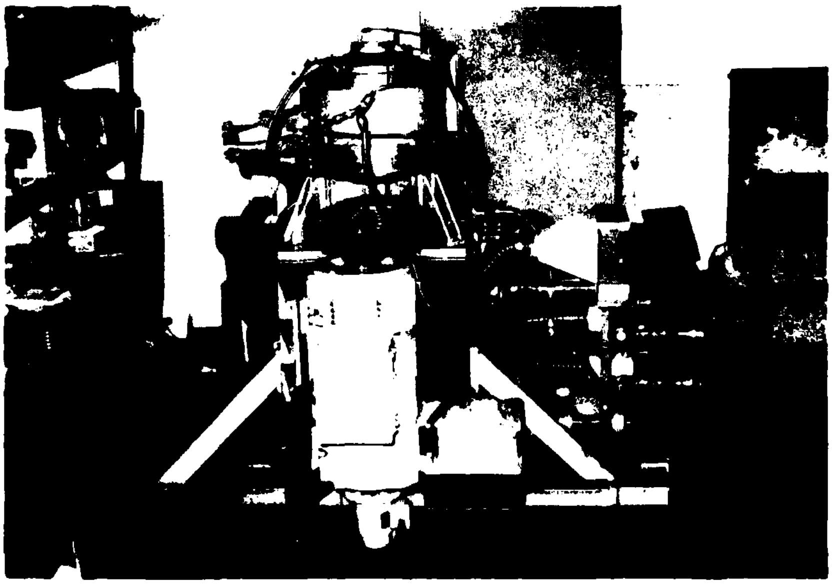

PHOTO 2. A109 PLANETARY SYSTEM TEST BENCH

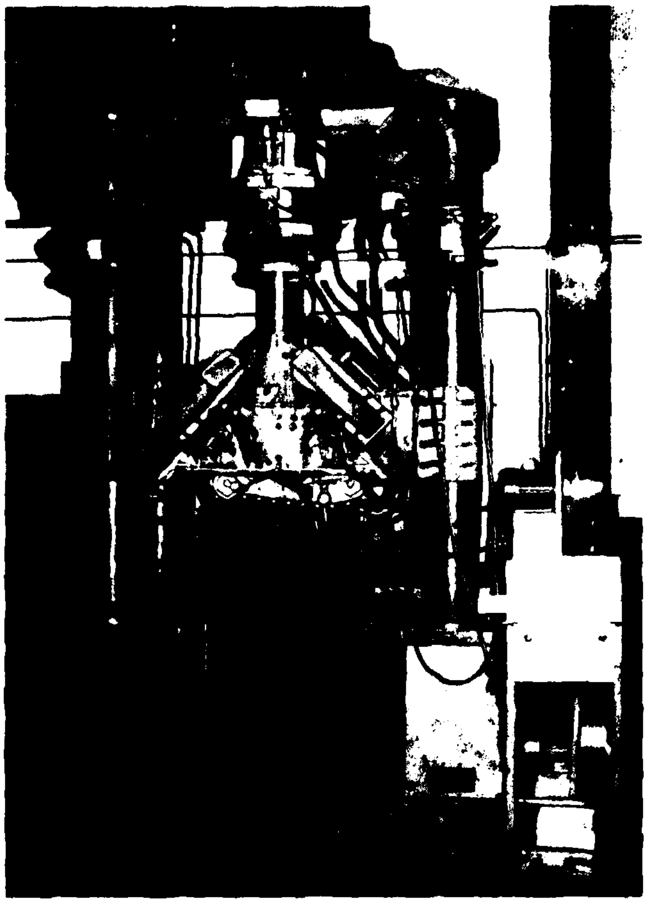

Photo 3. A109 MAIN GEARBOX CLOSE LOOP TEST BENCH

Photo 4. A109 TAIL GEARBOX CLOSE LOOP TEST BENCH

Photo 5. A 109 IRON BIRD

Photo 6. A 109 TIE DOWN HELICOPTER

DISCUSSION

M.D.Paramour, MOD (PE), UK

What tests do you consider are necessary for the acceptance of new production and overhauled transmissions, bearing in mind the range of component and assembly tolerances to be found *in production transmissions?*

Author's Reply

It's current practice at AGUSTA to green-run, with an adequate test cycle, each new or overhauled transmission before delivering it for service

After that, the transmission will be disassembled and visually inspected to assure the absence of malfunctions versus a standard acceptance specification derived from the experience accumulated during the development test.

In fact during the gear pattern and incremental loading tests, at least the most critical tolerances of gears, shafts, seals and bearings have been checked to establish and to confirm tolerance production limits and acceptance test criterias.

HELICOPTER TRANSMISSION TESTING

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SUMMARY

Helicopter propulsion system evaluation testing is conducted on individual components in the early stages of development. The total propulsion system is not operationally tested until the components are installed in the first aircraft. As such, dynamic interface problems are not detected until this stage of the full scale development program which can prove costly. To achieve more development/reliability testing and more meaningful qualification tests of the total system, the Naval Air Propulsion Center, under Naval Air Systems Command sponsorship, developed the only indoor facility in the United States capable of testing a "rotorless" helicopter propulsion system. This paper describes the test facility and presents the rationale and capability of an integrated, dynamic test stand for total system testing.

INTRODUCTION

The performance and design evaluation testing of helicopter propulsion systems has long been accomplished on the individual components, i.e., engines, transmission, gearboxes, etc. Certainly, such engine test stand and transmission gearbox back-to-back test operations are essential in the early stages of development but they fail to provide evaluation of any interactive effects which can exist in the total system. In fact, the total propulsion system is not operationally tested until installed in the tiedown or flight aircraft at which time these interaction or interface problems can cause needless delays and added cost. In an effort to obtain more development and reliability test time and more meaningful qualification tests of the total system, the Naval Air Propulsion Center (NAPC), under Naval Air Systems Command (NAVAIR) sponsorship, designed and installed the only indoor facility in the United States capable of testing an entire helicopter propulsion system except for the rotor and rotor hub assemblies.

The concept of an integrated, dynamic test stand for total system testing was formulated and developed to satisfy a number of test program objectives. Briefly, the types of test programs would include:

- Verification of system designs and performance for new and upgraded aircraft
- Qualification per specification requirements (AS-3694A)
- Endurance/Reliability - Simulated Mission Endurance Test (SMET)
- Pre-flight Rating and Flight Worthiness Demonstration
- Research/Development of system hardware, accessories, and lubricant/lubricant cooling systems
- Fleet service problems

All of these programs can be effective in the early identification of the dynamic interface problems in a propulsion system prior to installation in either the first flight or a modified flight vehicle. NAVAIR acknowledged the capability of an integrated system test facility by specifying its use for pre-flight and qualification testing in a helicopter transmission general specification, AS-3694A (reference 1). The specification covers the general requirement for VTOL-STOL transmission systems and the specific requirements for design, component testing, and integrated system testing of these systems.

FACILITY DESCRIPTION

The facility is an "open end" design wherein the flight powers and certain load parameters are imposed on the main transmission, nose, intermediate, and tail rotor gearboxes, interconnecting shafting, clutches and couplings, and all accessories common to the total propulsion system. The "prime movers" are the actual turboshaft engine(s) used in the flight vehicle.

The "heart" of the facility is the 30:1 ratio step-up gearbox with a capacity of over 5.97 megawatts (8000 horsepower) as shown in Figure 1. The gearbox translates the low speed-high torque output of helicopter main transmission rotor shafts to high speed-low torque values which permits power absorption by the use of multiple waterbrake dynamometers (Figure 2). Power absorption at the tail rotor gearbox is also accomplished with a waterbrake system. All accessory drives are "loaded" with the specific generator, hydraulic pump, and rotor brake common to the particular helicopter propulsion system. Ancillary electrical and hydraulic power absorption systems within the facility provide appropriate generator and hydraulic pump loading throughout the test program.

The integrated, dynamic test stand also includes the means of applying thrust and bending loads to the main rotor shaft of the test transmission. The thrust and bending

loading systems are shown schematically in Figure 3 for an AH-1J transmission. Maximum thrust loads of up to 222,500 newtons (50,000 pound-force) can be applied to the main rotor shaft by means of four gas (nitrogen) operated "jacks" mounted to the four corners of a square loading platform or table. A tapered roller bearing allows rotation of the main rotor shaft and the bearing inner race within the stationary outer race and thrust table. Load cells in the connecting arms to each of the loading "jacks" provide accurate measurement of the thrust loads.

The bending load is applied through a separate housing at the top of the main rotor shaft. As in the thrust loading system, a tapered roller bearing permits main rotor shaft rotation within the stationary housing while the externally applied load is reacted through the bearing to the rotating shaft. The system was designed and procedures were established to assure proper application of the bending load, and to maintain accurate alignment of the entire shafting from the test transmission to the step-up gearbox. If the load were improperly imposed, a severe misalighment could occur and result in high vibratory loading of the test components. Maximum bending loads of up to 22,500 newtons (5000 pound-force) can be applied and accurately measured by means of a load cell. The main rotor shaft loading system imposes realistic load conditions on the transmission components and improves the overall capability of the facility for performance and reliability evaluations. The facility design specifications and operating limits are presented in Table I.

TABLE I

NAVAL AIR PROPULSION CENTER TRANSMISSION TEST FACILITY DESIGN SPECIFICATIONS (TEST CELL 8W)

Input Power Source - 1, 2 or 3 Turboshaft Engines

Transmission Main Rotor Speed - 175 rpm to 325 rpm

Power Absorbed - Main Rotor - 5.97 megawatts (8000 hp) maximum

Rotor Shaft Torque - Maximum 325,400 newton-metres (240,000 pound-force-foot) at 175 rpm

Nominal Engine Airflow - 22.7 kg/sec (50 lb/sec) maximum

Cold Inlet: -54°C (-65°F)   Hot Inlet: +104°C (+220°F)

Sea Level (Ambient Conditions)

Thrust Loading (Main Rotor) - 0 to 222,500 newtons (50,000 pound-force)

Bending Loading (Main Rotor) - 0 to 22,250 newtons (5,000 pound-force)

This facility provides sufficient flexibility to install and test a variety of current and future helicopter propulsion systems up to and including the growth version of the Sikorsky RH-53D helicopter.

An isometric view of the RH-53D propulsion system installation is shown in Figure 4. The test installation includes the two T64-GE-415 engines, two nose gearboxes and interconnecting shafting, main transmission, accessory gearbox, oil cooler/fan, two sections of the drive shaft to the intermediate gearbox, and the tail rotor drive shaft and gearbox. The main rotor shaft output power is absorbed by three 2.2 megawatts (3000 hp) waterbrake dynamometers on top of the step-up gearbox and a tail rotor waterbrake dynamometer capable of absorbing up to 1.5 megawatts (2000 hp).

The control room provides extensive instrumentation, as shown in Figure 5, for the measurement and monitoring of all parameters necessary for proper operation of the engine(s) and all subsystems comprising the total propulsion system. The engine(s), waterbrake dynamometers, and main rotor loading control consoles have direct access to related instrumentation for continuous monitoring and control by the qualified operators. Critical temperatures, pressures, and all chip detectors in the propulsion system components are included in a special panel with warning lights and audible alarms (far right in Figure 5). A critical alarm signal will initiate immediate execution of emergency shutdown procedures to avoid possible catastrophic failures.

The speeds and torques of all rotating subsystems are measured and monitored throughout the test program. In fact, present capability includes direct and simultaneous readout of all input/output powers on command which permits accurate evaluation of the system mechanical efficiency.

The main rotor shaft can also be strain gaged, as shown in Figure 6, to measure the strain induced by the bending and thrust loads, either separately or in combination. The strain signal is telemetered to visicorder and digital readout instruments which permits system calibration with the load cells under static conditions and continuous monitoring under dynamic conditions. The location of the strain measurement and recording instrumentation in the control room is shown in Figure 7. The Visicorders and telemetry receivers can be seen in the center console which is flanked by the vibration monitor console on the right and the generator loadings systems in the console on the left. The console at the far left contains all power supply equipment for the control room. Two, closed-circuit television systems with monitors in the control room (far right console) are used to maintain visual surveillance of the test area throughout the test program.

On a continuing basis, every effort is made to improve and up-date the facility for data accuracy (instrumentation) and effective mission simulation (control/loading).

Future plans include: (a) the tie-in of test instrumentation into the Center's data acquisition and programing systems to reduce the complexity and man-hours in the recording and analyzing of the data; and (b) the installation of a failure monitoring tape system which will continuously record selected parameters for analytical review and diagnosis should a failure occur during test operation.

In addition, NAPC has determined the design concept and the hardware requirements necessary for expansion of the present transmission test facility. The new system would provide basic capacity to 11.94 megawatts (16,000 horsepower) with overload capability to 13.43 megawatts (18,000 horsepower).

DISCUSSION

The integrated, dynamic test stand permits full evaluation of engine/drive train compatibility under simulated service operating conditions prior to installation in an aircraft. The concept of total systems testing offers significant advantages over the regenerative or back-to-back arrangements particularly in assessing interaction effects. The coupled interactions between two or more helicopter dynamic subsystems have often been the source of vibration problems which have required compromising the performance of one or more components. Such interface problems are among the last to be found and the most costly to correct in a development program. The experience of U.S. helicopter manufacturers with dynamic interface development problems are extensively examined and summarized in references 2 through 6. These reviews were performed under government sponsorship and establish a definite need for further analytical and testing efforts to achieve better understanding of the problems and the potential solutions. The method of total propulsion system testing can provide this much needed information in (a) developing the broad data base to improve analytical design techniques, and (b) during the development program to avoid costly modifications and delays.

In general, helicopter mission times are short compared to fixed-wing aircraft and require considerable power cycling of the engine and drive system. The integrated, dynamic test stand permits transient speed and power changes much the same as in the helicopter. In fact, the rigorous mission cycle selected and imposed on the total propulsion system allows an assessment of component performance, torsional vibration instability problems and other potential operating problems associated with power cycling and system response. The "one engine inoperative" (OEI) mode is also accomplished in the test facility by supplying maximum power levels to one input module (clutch engaged) while the opposite input module clutch operates in the overrunning mode. This method applies more realistic clutch operation than the regenerative type test stands which must "lock-out" the overrunning clutch during each cycle. The testing of the total propulsion system under simulated service operation will demonstrate flight worthiness and the fail-safe features of the dynamic components. In addition, the discrepancies cited at the teardown inspection of the test components after qualification tests will indicate specific hardware deficiencies and the most probable long-term failure modes of each propulsion system. The individual testing used extensively to qualify transmission accessories, remote gearboxes, oil coolers and fans can be combined since the facility permits full qualification of these components along with the main transmission. In addition, full operational testing of the main rotor brake(s) can be accomplished with rotor brake actuation to control the time required for full rotor stop from any main rotor speed.

This method of testing provides all the advantages of tie-down testing in a ground test vehicle (GTV) at lower cost and without its associated airframe vibratory stress problems due to rotor ground effects. In addition, the integrated test stand has more extensive and sophisticated instrumentation and equipment which can be "tailored" to meet specific test requirements. The test program dictates the type of data needed and the test cycle establishes the frequency of recording data. All instrumentation and equipment are calibrated as necessary to assure that the required degree of accuracy is maintained throughout the test program. Test conditions and critical operating parameters are more accurately controlled under test cell operation. Unlike GTV tests which are subject to ambient conditions, the facility can control engine inlet temperatures with "conditioned air" to assure that proper power levels are maintained throughout the test program.

A propulsion system test facility can be effectively used in a development program as an integrated, dynamic test for the critical assessment of:

(a) system flight worthiness and safety in a 60-hour pre-flight rating test.

(b) system design and performance in a 150-hour qualification test on production parts.

(c) system reliability/durability in a 500-hour endurance test.

Of course, such total system testing must be predicated on adequate bench testing of all components in the early stages of development. In particular, overstress testing of components in regenerative type test facilities in which the requirement is not a "must pass" test but an evaluation of parts design and integrity. Basically, the test objectives are to determine modes of failure, detectability of failures and the extent of fail-safe features in a program used to "de-bug" the components through redesign, fix and repair. All redesigned and improved parts should be adequately evaluated in these tests prior to installation in the components for total system testing on an integrated, dynamic test stand. In addition, the total system tests will permit qualification and endurance evaluation of all clutches, couplings, shafting, rotor brakes, and accessories

along with the gearboxes.

A brief review of a test program conducted in the NAPC transmission test facility may serve to demonstrate its capabilities and versatility. Often, the demands for new and enlarged operational roles in existing helicopters are satisfied by the increased power available in growth engines which can be "fitted" into existing airframes with few modifications in design to the aircraft and power drive system. One such model modification was accomplished for the Boeing-Vertol CH-46E aircraft with the General Electric T58-GE-16 engine, a growth version of the T58-GE-10 engine used in earlier CH-46 models. A 200-hour qualification test of the uprated (2089 kw/2800 SHP) propulsion system was completed in the NAPC transmission test facility. Figure 8 shows the installation of the aft transmission and mixbox with two, transmission mounted generators. Power absorption of the aft rotor shaft was accomplished by a single waterbrake on the step-up gearbox through the flexible coupling and torquemeter arrangement. Since the gear systems in both the forward and aft transmission are basically the same and the aft transmission is the heavier loaded, the forward transmission was not tested. However, it was simulated in the test installation by a waterbrake power absorption system as shown in Figure 9. Subsequent qualification of the forward transmission was based solely on the performance of the aft transmission. The program also included a qualification test of a second source overrunning clutch to meet projected shortages at the Naval Air Rework Facility level. The program also provided as assessment of the T58 engine Power Management System which maintains "balanced" load sharing between the engines at any output power. The qualification test consisted of 50 cycles at the loading sequence shown in Table II for a four hour cycle. This test cycle provided for a total of 150 hours of dual engine operation and 25 hours (each) of single engine operation. Each load condition in the test cycle was set by imposing the load on the forward (transmission) waterbrake and adjusting engine input powers to the total power requirement. In this manner, the aft/mix transmission power losses were readily assessed. Special instrumentation provided a simultaneous readout of all input and output powers which indicated an aft/mix transmission efficiency of 97.3 percent at 2089 kw (2800 SHP). Single engine operation was accomplished with the "inoperative engine" at ground idle which imposed the most severe differential speed (67 percent) in the overrunning clutch. In addition, each clutch experienced 50 engagements (one per cycle) plus a number of engagements at each engine start. Throughout the test program, the test transmission was periodically monitored by spectrometric oil analysis (SOAP) and continuously by magnetic chip detectors.

TABLE II

CH-46E QUALIFICATION TEST

AFT AND MIX TRANSMISSION

TOTAL Kw/(HP)	TEST CONDITION AT AFT TRANSMISSION	TORQUE N-M/ (IN-LB) EACH ENGINE AT 19,500 RPM	AFT TRANSMISSION AT 264 RPM Kw/(HP)	FORWARD WATERBRAKE AT 2562 RPM kw/(HP)	DURATION HRS:MIN
1. 627 (840)	NRS; 30% N.R. Input Torque	153 (1357)	313 (420)	313 (420)	0:10
2. 2089 (2800)	NRS; 110% N.R. Input Torque	511 (4525)	1149 (1540)	940 (1260)	0:50
3a. 1492 (2000) (NOM) (1481°F T5)	Single (Engine #1)	730 (6464)	746 (1000)	746 (1000)	0:20
3b. 1238 (1660)	Single (Engine #1)	606 (5365)	619 (830)	619 (830)	0:10
4. 2059 (2800)	NRS; 120% N.R. Input Torque	511 (4525)	1253 (1680)	836 (1120)	1:55
5. 2238 (3000)	NRS; 128.6% N.R. Input Torque	548 (4848)	1343 (1800)	895 (1200)	0:05
6a. 1492 (2000) (NOM) (1481°F T5)	Single (Engine #2)	730 (6464)	746 (1000)	746 (1000)	0:20
6b. 1238 (1660)	Single (Engine #2)	606 (5365)	619 (830)	619 (830)	0:10
				TOTAL	4:00 HOURS

NRS = Normal Rated Speed = Power Turbine RPM = 19,500
FWD and AFT XMSN Input RPM = 2562
AFT Rotor (Main) Output RPM = 264
NR = Normal Rating of AFT XMSN = 1044 kw (1400 HP)

#1 Engine - Left-Hand Looking Forward
#2 Engine - Right-Hand Looking Forward

The events described next emphasize the ability of integrated system testing to identify and resolve problems which can result from the coupling of propulsion system components.

During the initial checkout of the test installation, the vibration (acceleration) level at the aft tube support of each engine was well above the limit specified in the aircraft operating manual. The accelerometer pickups were monitored, at the high speed shaft rotational frequency, by the actual aircraft ground test equipment in accordance with normal ground run-up procedures. The high vibration level had reportedly been a source of service problems in the field because of the high operating speed and inherent, complex design of the high speed shaft. The shaft "complexities" were further compounded by the need of a special adapter to permit the use of the T58-GE-10 engine shaft with the T58-GE-16 engine. At the time, an entirely new shaft for the T58-GE-16 engine, with a "phase-shift" torquemeter system, was in the design stage and unavailable for this test program. A vibration spectrum analyzer, with full scanning and tracking capability at all rotational frequencies, was used to assist in the study of vibrational modes in the system. These results were provided to Boeing-Vertol for a critical speed analysis of the system which led to a resolution of the problem through improved assembly fits and rebalancing of shaft components. The total effort on the high speed shaft problem provided a significant amount of operational vibration data which was not readily attainable in other test facilities.

The 200-hour test program was then completed without further incident of major importance. At the end of the test, inspection of the component parts revealed that all parts were in highly satisfactory condition except for the spiral bevel input pinion and gear mesh and one input pinion idler assembly which "housed" the second source test clutch. The spiral bevel gear teeth exhibited a significant amount of surface distress (scuffing) which precluded full qualification of the propulsion system at the uprated power. An extensive investigation indicated that the damage may have been related to "poor" tooth patterning at assembly and a tentative qualification approval was agreed upon. Full qualification approval was ultimately given after a 200-hour penalty run in the first flight aircraft in which the spiral bevel pinion and gear were determined to be in satisfactory condition. The primary damage to the input pinion assembly was described as "flats" and scuffing of the clutch inner race, and, to a lesser degree, to the clutch outer race in the idler hub assembly. The second source test clutch retainers and sprags were not distressed and were considered in good condition for further operation. The damage to the clutch races was attributed to out-of-phase loading of the sprags in the test clutch, i.e., not all sprags were loaded equally at all times. Since the "production" clutch in the other input idler assembly performed well without distress under similar operating conditions, qualification was denied for the second source clutch with specific recommendations for analysis/redesign as necessary and for further full scale testing. As a matter of record, a second source clutch configuration was later qualified following satisfactory demonstration in bench tests and a 200-hour flight test.

A similar effort was employed in the qualification of the Bell Helicopter AH-1J propulsion system at increased power in the NAPC transmission test facility. The results of the AH-1J qualification program were reported with recommendations to incorporate specific hardware modifications for improved system durability. In comparison, another uprated, advanced helicopter model, not tested in the NAPC facility, experienced major component failures in ring gears, main rotor shaft thrust bearings, and rotor brake bearings, over a two-year period of field service. The testing at the contractor's facilities, which included a 200-hour test in the regenerative test stand and two, 200-hour tests in a GTV, had not uncovered the deficiencies experienced in service. From the contractor's records it was determined that the limitations in attainable test conditions may have contributed to the non-failure of components in the three 200-hour tests. For example, limited capacity in the regenerative stand resulted in applied main rotor thrust loading below required values. In the GTV tests, high ambient conditions limited the maximum engine powers attainable and resulted in a lower power spectrum during test operation. The integrated, dynamic test stand is capable of imposing realistic operating conditions throughout the entire operating cycle.

The facility has also been effectively and expeditiously used in the resolution of fleet service problems. In most cases, the inherent capability of the integrated, dynamic test stand permits sufficient flexibility in test operation, use of special instrumentation, and close control of system parameters to extensively investigate service malfunctions/failures through duplication of actual operating characteristics.

FUTURE PLANS

In future helicopter and V/STOL development programs, the U.S. Navy will require compliance with Specification AS-3694A and will continue efforts to provide a coordinated military specification. The integrated system test stand is considered a viable means of performing pre-flight rating and qualification testing in a properly scheduled, well-defined test program which will provide a successfully integrated and qualified propulsion system. In addition, a 500-hour endurance test will demonstrate system durability and add a high degree of functional maturity to the final design configuration.

The transmission facility itself will be improved and updated to assure maximum capability in meeting future test requirements of advanced propulsion systems. Near-term efforts will provide computer system data aquisition and analysis which will improve transient capability and reduce manpower requirements within the facility. Failure analysis and diagnostic capability will be improved with the addition of a multi-channel failure monitoring tape system. In addition to the instrumentation improvements, a parallel effort will be directed to improve the control/loading systems for more effective mission performance simulation in the test programs. Long-term plans are to provide increased main rotor power absorption capacity through a proposed expansion of the present transmission test facility.

The NAPC transmission test facility is available to government agencies and industry contractors in the United States and the NATO nations. The costs and scheduling of programs will be determined upon application to the Naval Air Systems Command.

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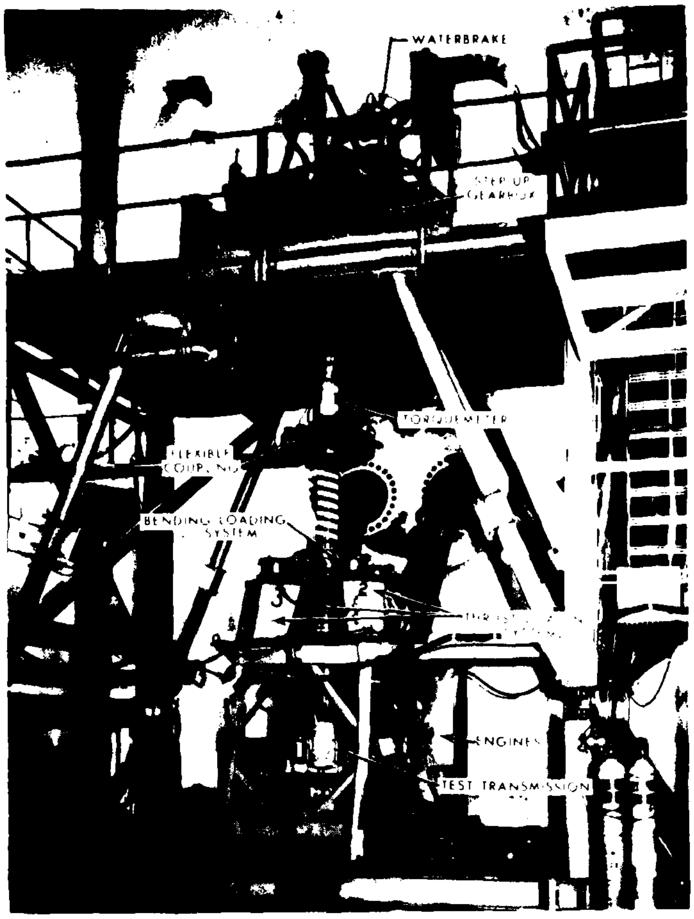

Fig.1 NAPC transmission test facility

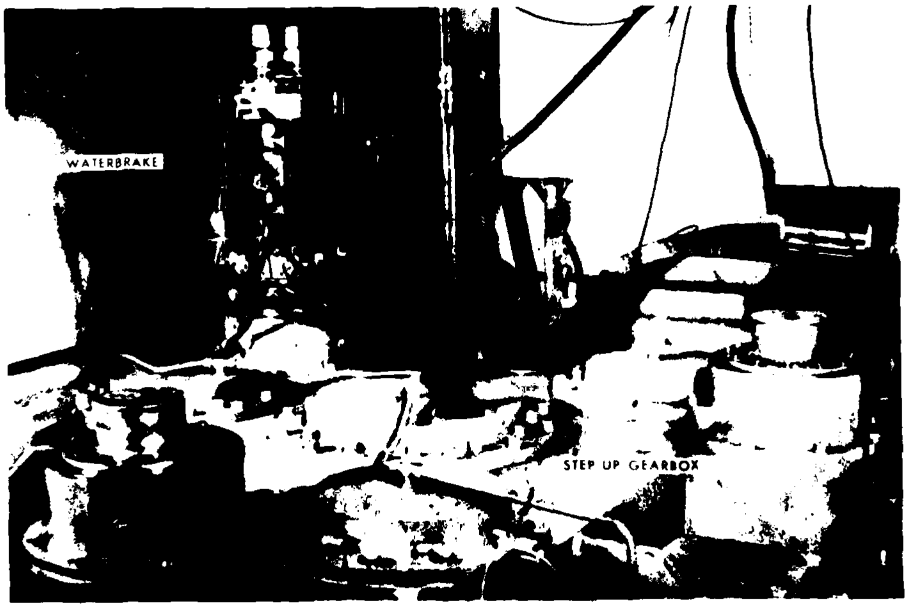

Fig.2 Main rotor shaft power absorber waterbrake on the step-up gearbox

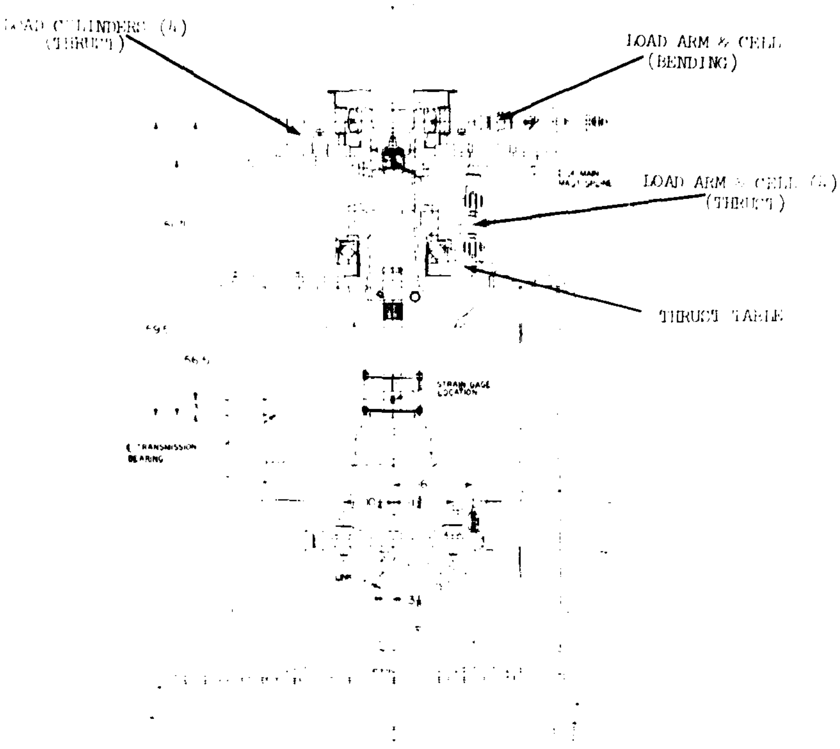

Fig.3 Main rotor thrust and bending loading systems

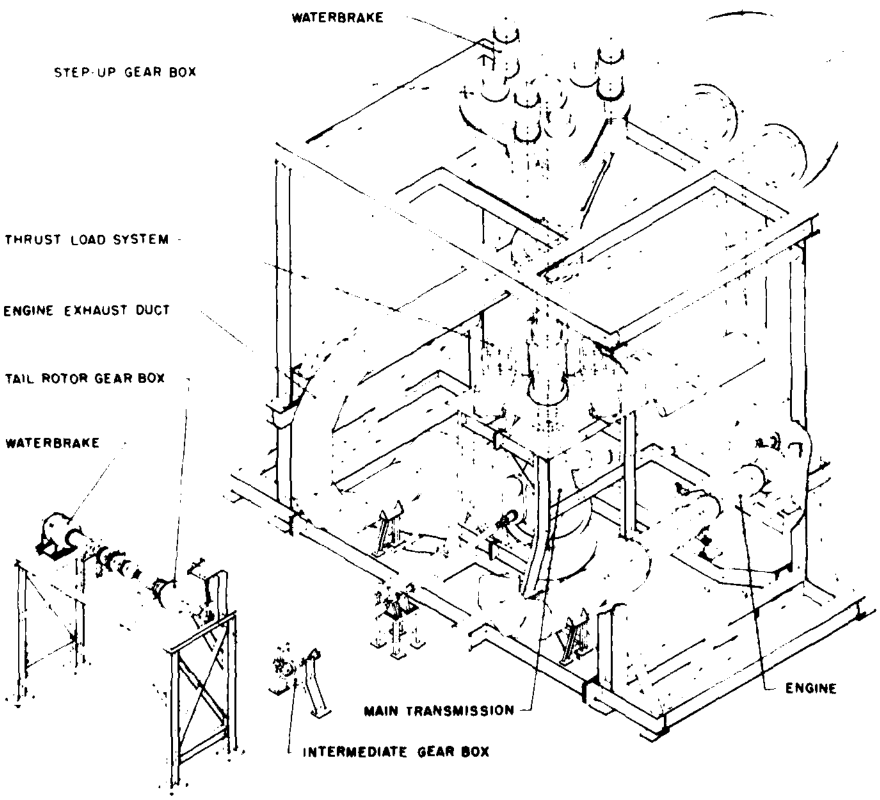

Fig.4 Isometric view of RH-53D transmission test installation

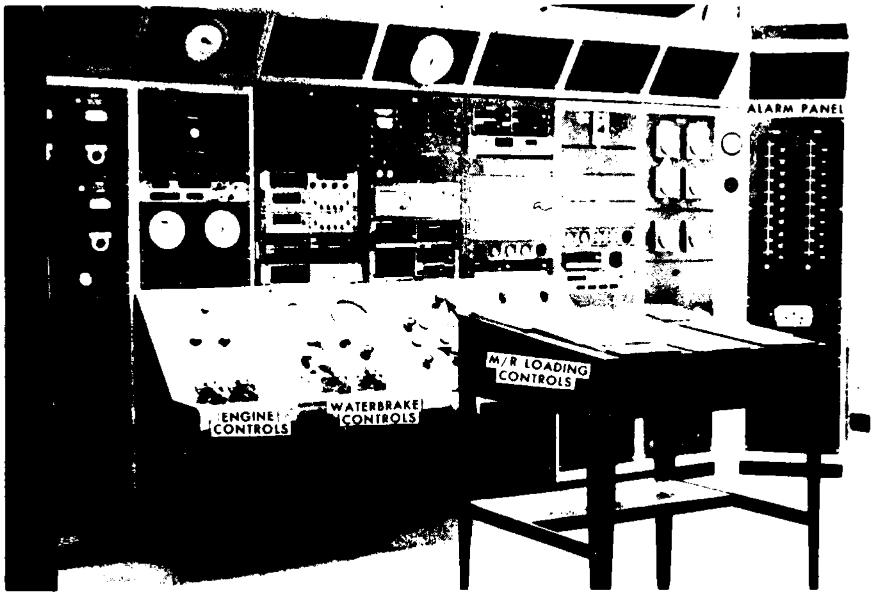

Fig.5 Transmission test facility control room showing control/instrumentation consoles

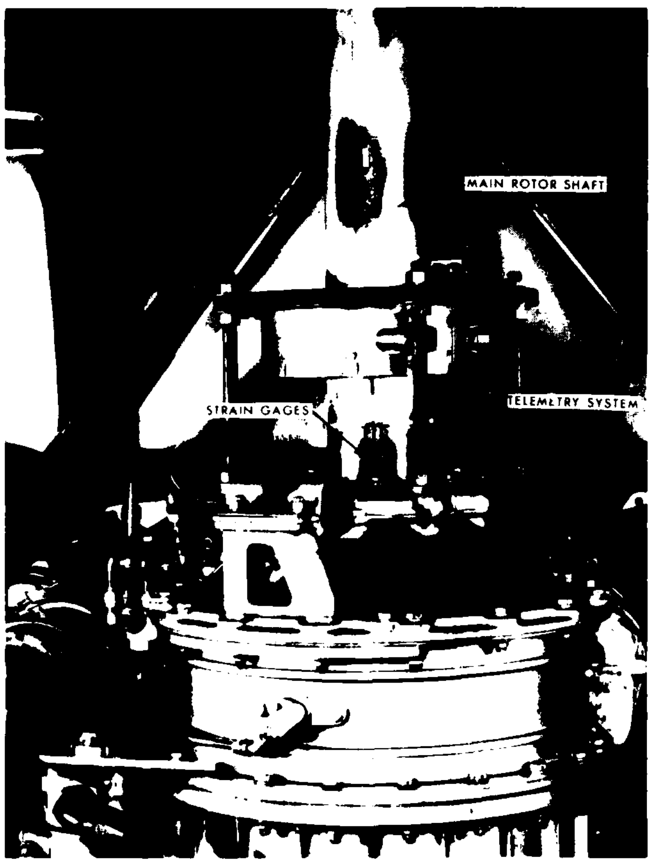

Fig.6 Main rotor shaft strain gage and telemetry installation

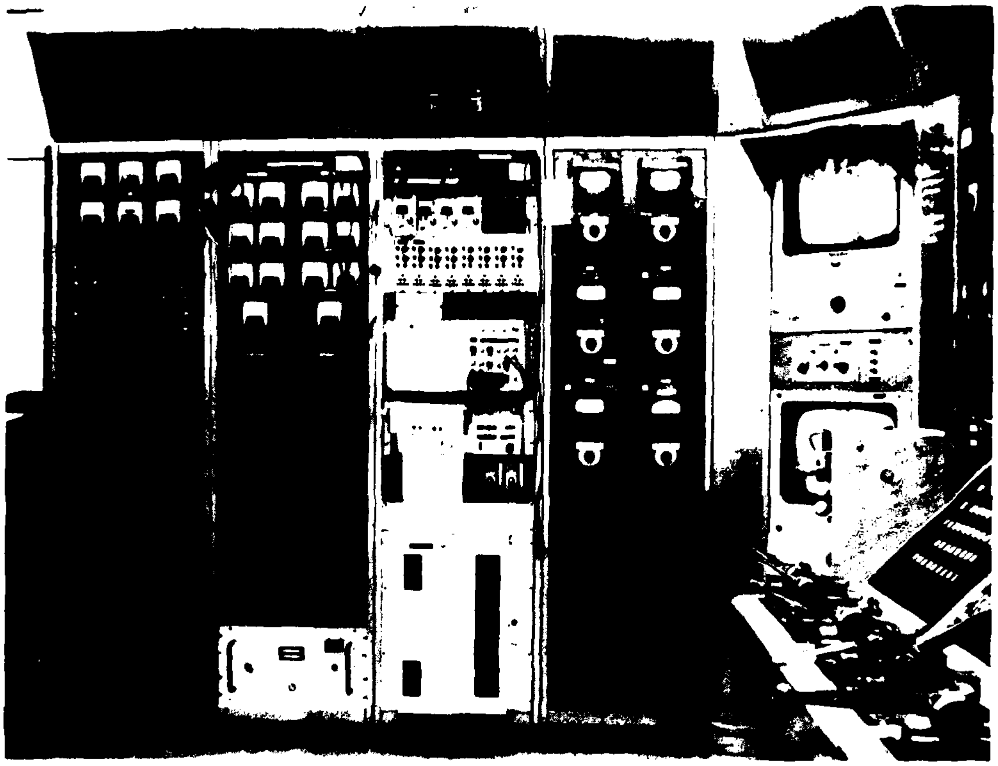

Fig.7 Transmission test facility control room showing instrumentation/monitoring consoles

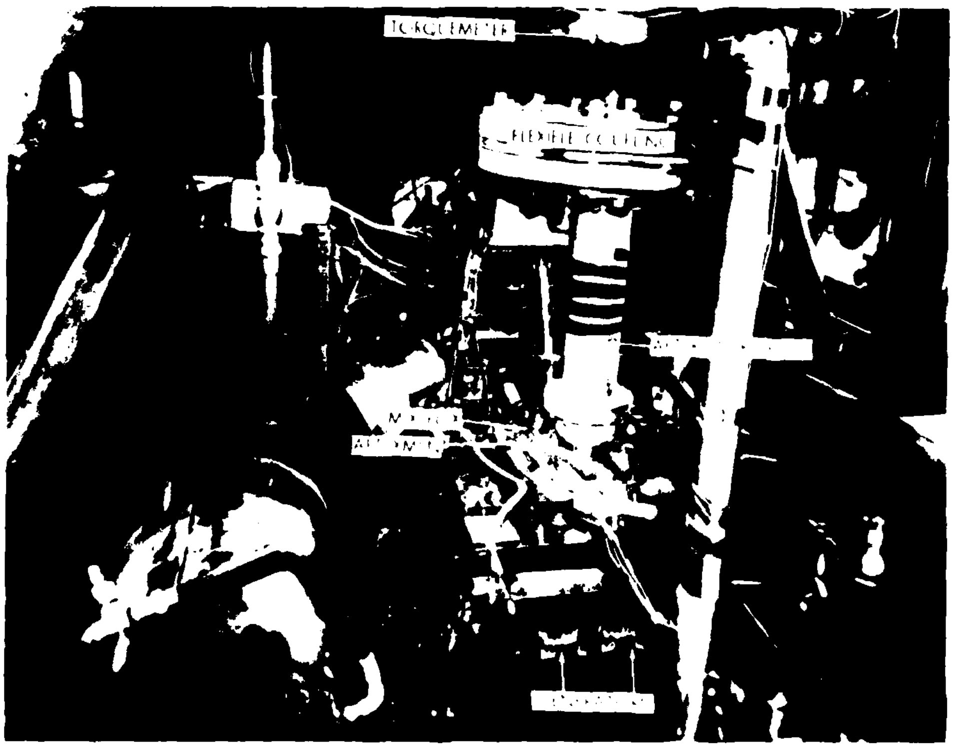

Fig.8 CH-46E aft transmission/mixbox test installation

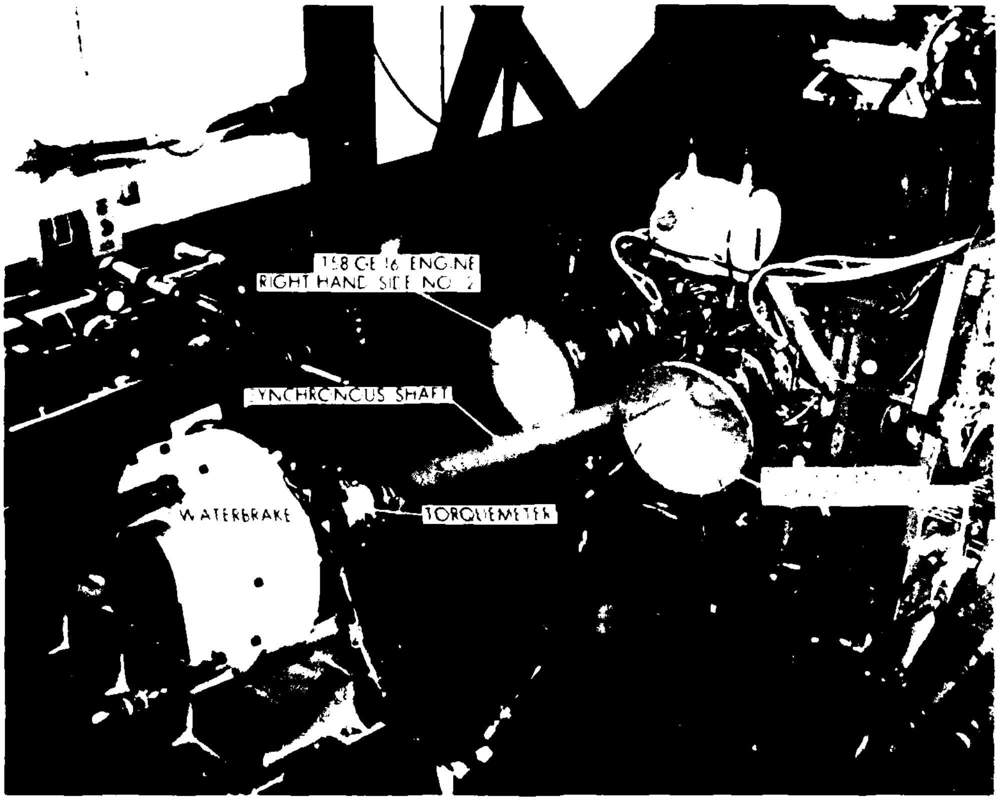

Fig.9 Waterbrake power absorption system simulating CH-46E forward transmission

ESSAIS DE DEVELOPPEMENT DE MOTEUR A «DUREE DE VIE LIMITEE»

par

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RESUME

Un système de propulsion à "durée de vie limitée" doit avoir des poussées massique et surfacique et une consommation spécifique performantes tout en restant simple de conception pour assurer un coût de production faible.

A ces qualités s'ajoutent la facilité d'intégration dans la cellule (coefficient de distorsion d'entrée d'air élevé, équipements concentrés, formes générales simples, ...) et l'accomplissement fiable de la mission impartie (mise en oeuvre et démarrage, régulation précise, bonne tenue aux sollicitations de l'environnement, ...).

Les essais de développement et de qualification de production sont donc très différents de ceux de propulseurs d'aéronefs pilotés : la première partie de la lecture présente les essais principaux réalisés sur le turboréacteur MICROTURBO TRI 60 dans le cadre de quatre programmes d'adaptation à des véhicules sans pilote.

La seconde partie donne quelques indications sur des solutions retenues en vue d'obtenir un faible coût de production.

1. PRESENTATION DU REACTEUR A DUREE DE VIE LIMITEE TRI 60-2

En 1972, la Société MICROTURBO a entrepris le développement d'un réacteur de missile sans pilote conçu dès l'origine pour avoir une durée de vie limitée et un faible coût de production :

- durée de vie : 20 heures et/ou 50 cycles,
- 100 francs par décanewtons (10 dollars par livre de poussée).

Les données techniques, en l'absence de tout programme d'application, étaient simplifiées :

- diamètre maximal .... 330 mm
- masse totale .... 50 kg
- poussée maximale .... 300 daN
- consommation spécifique ... 1,30 kg/daN.h

Il n'existait de même à cette époque aucun réglement dont l'analyse puisse servir de guide dans la conception d'un réacteur : cette liberté a été certainement un élément positif.

L'analyse de différents cycles thermodynamiques et concepts architecturaux a débouché sur le choix d'un turboréacteur à veine axiale dont les principaux composants sont :

- carter d'entrée d'air pouvant contenir l'ensemble des accessoires,
- compresseur axial à 3 étages,
- chambre de combustion annulaire,
- turbine axiale mono-étage.

Hormis les performances excellentes obtenues, bon nombre des qualités recherchées n'ont été révélées que dans le cours des essais de développement et d'application.

La planche 1 décrit le turboréacteur ainsi conçu :

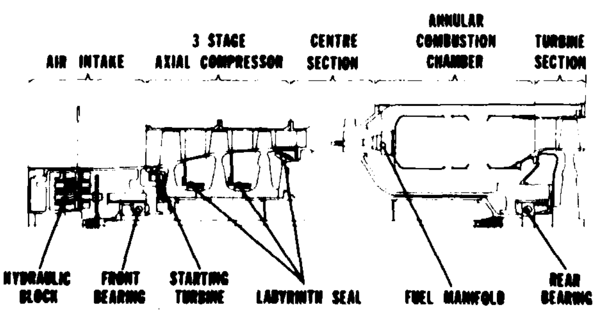


Planche 1

L'amélioration progressive des composants et leur meilleure adaptation réciproque conduisirent aux performances actuelles suivantes :

- diamètre maximal .... 330 mm (maintenu)
- masse totale .... 45 kg (réduction 10 %)
- poussée maximale .... 370 daN (augmentation 23 %)
- consommation spécifique ... 1,30 kg/daN.h (maintenue)

A ce jour, quatre programmes utilisent le réacteur TRI 60, preuve de la justesse des choix initiaux :

- deux programmes concernent des engins-cibles pour lesquels l'adaptation du système global de propulsion, monté en pod à l'extérieur du fuselage, a été particulièrement aisée : la planche 2 démontre la facilité d'intégration des équipements périphériques. Les engins-cibles SNIAS C22 et BEECH VSTT poursuivent leurs essais de développement et d'évaluation.

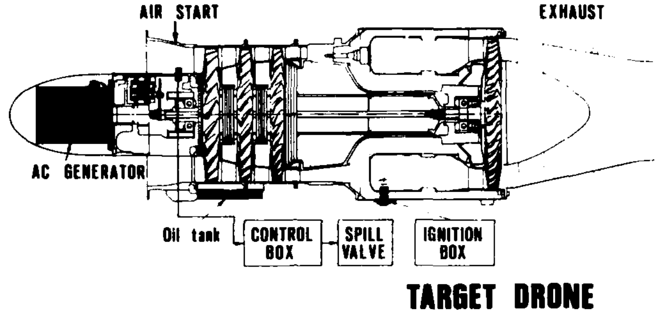


Planche 2

- deux programmes concernent des missiles anti-navires pour lesquels cette intégration est encore plus simple : la planche 3 parle d'elle-même. Ces deux applications sont les missiles BAeDG SEA EAGLE et SAAB RB 15.

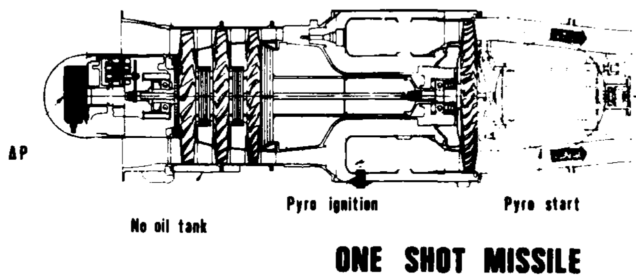


Planche 3

2. ESSAIS DE DEVELOPPEMENT

2.1. Indépendamment d'une quelconque application, de nombreux essais de développement ont eu pour but de caractériser le domaine de vol du réacteur TRI 60. Sous cet aspect, les essais intrinsèques d'un propulseur à durée de vie limitée diffèrent peu de ceux d'un propulseur normal, si ce n'est par la difficulté liée au faible nombre d'heures alloué pour conduire l'essai et à la recherche systématique des limites : la qualification d'un domaine de vol requiert à lui seul la durée normale de vie escomptée compte tenu de son utilisation pendant une bonne partie de ce temps à la limite du niveau maximal des contraintes calculées mécaniques et/ou thermiques. Les bancs d'essais MICROTURBO et CEPr ont été utilisés pour démontrer le domaine de vol initial de la planche 4.

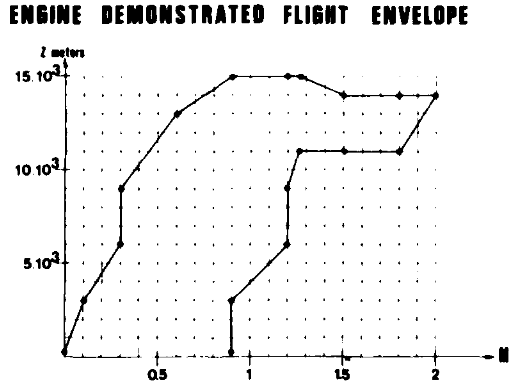


Planche 4

2.2. A cette démonstration des performances s'ajoute la recherche d'une durée de vie initiale grâce à un essai d'endurance accéléré. Le programme utilisé est celui de l'US Air Force (planche 5) particulièrement sévère pour ce qui est du temps d'application de la puissance maximale (TOT) au point fixe au sol, *donc à un niveau* thermique majoré. Ces essais se sont déroulés principalement sur les bancs d'essais CEPr et FORT-EUSTIS (USA). A ce jour, les réacteurs TRI 60 ont accumulé plus de 450 heures et 2 300 démarrages.

ENDURANCE TEST SCHEDULE CYCLE

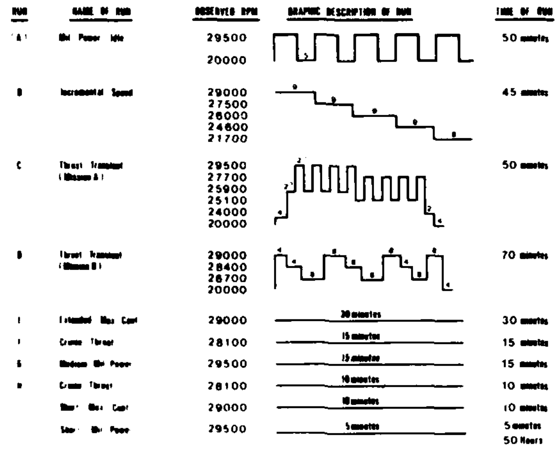


Planche 5

2.3. De nombreux [illegible] annexes ont été conduits pendant le développement du turboréacteur [illegible]trées, température de peau, dilatation, survitesse, surchauffe, [illegible] encore en vue d'étendre les performances et possibilités déjà démontrées.

3. ESSAIS [illegible]

[illegible] réacteur à durée de vie limitée diffère essentiellement du réacteur [illegible]. En effet, les conditions d'intégration sont difficiles à satisfaire [illegible] recherche de fonctions simples et fiables assurant la réussite des missions [illegible]

3.1. Distorsion d'entrée d'air

Chronologiquement, c'est la détermination de cette qualité qui a été effectuée en premier lieu.

La première étape a été la caractérisation au banc du niveau acceptable de distorsion que pouvait supporter le turboréacteur. Le critère choisi est le DC 90 RR obtenu par un obstacle de perméabilité variable, de 90 degrés d'extension périphérique, situé à un diamètre en amont du compresseur. Finalement, il a été démontré qu'une plaque pleine n'empêchait pas le démarrage, l'accélération et le fonctionnement du réacteur, sans décharge du compresseur : la valeur correspondante de la distorsion a été parallèlement mesurée :

DC 90 RR = 1,52

Pour chaque application, cet essai a été répété au banc ou en soufflerie, en utilisant l'entrée réelle du missile, de manière à qualifier les effets d'incidence et de dérapage. Les installations utilisées ont été principalement les bancs du NGTE et de MICROTURBO et la soufflerie de l'ONERA à MODANE.

La planche 6 résume les caractéristiques démontrées.

RESUME
SUMMARY

FONCTIONNEMENT NON PERTURBE
OPERATION UNAFFECTED
DC 90 RR ≥ -0,74

REACTEUR DEMARRE ACCELERE FONCTIONNE
ENGINE STARTS ACCELERATES AND RUNS
DC 90 RR ≥ -1,52

Planche 6

3.2. Démarrage en autorotation

L'utilisation sur les missiles anti-navires impliquait la mise en oeuvre dans un temps minimal : les spécifications initiales demandaient d'atteindre 90 % du régime nominal en 5 secondes à compter de l'éjection du cache d'entrée d'air (largage d'avion) ou du catapultage hors du container (lancement d'un bateau).

Pour satisfaire cette condition, il a été d'abord envisagé d'utiliser un démarreur à cartouche (pyro démarreur) assurant la mise en rotation rapide du rotor. La planche 7 définit les caractéristiques retenues.

SPECIFICATION DU PYRODEMARREUR
PYROSTARTER SPECIFICATION

MASSE DE POUDRE WEIGHT OF PROPELLANT	0,45 kg
MASSE TOTALE DU DEMARREUR TOTAL WEIGHT OF STARTER	7,0 kg
DUREE DE FONCTIONNEMENT OPERATING TIME	3,0 s

Planche 7

Cependant, dès les premiers essais, il est apparu que la conception axiale du réacteur assurait la satisfaction de l'impératif par l'utilisation seule de l'autorotation.

Pour l'application air-mer, la séquence est initiée par l'éjection du cache évitant l'autorotation pendant le vol porté. Dans le même temps, le robinet pyrotechnique de carburant et les deux pyro allumeurs sont mis à feu : plus de 200 cycles de démarrage ont été réalisés sur le banc d'essai NGTE. La planche 8 résume la loi temps de démarrage fonction du nombre de Mach au moment du largage. Les performances de démarrage en autorotation assurent le succès de la mission lors de largages à altitude très faible.

DURATION OF WINDMILLING START AT S.L

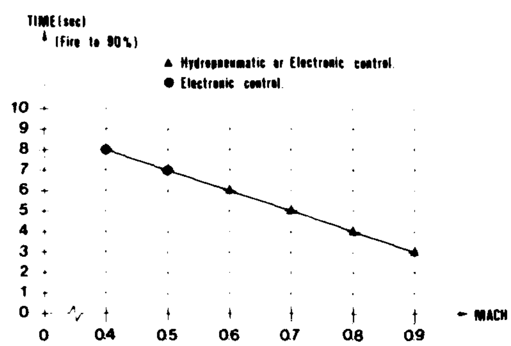


Planche 8

ALTITUDE AND AIRSPEED DURING SURFACE-TO-SURFACE LAUNCH

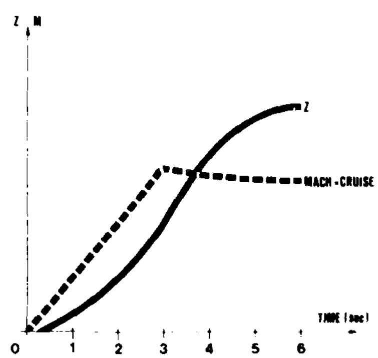


Planche 9

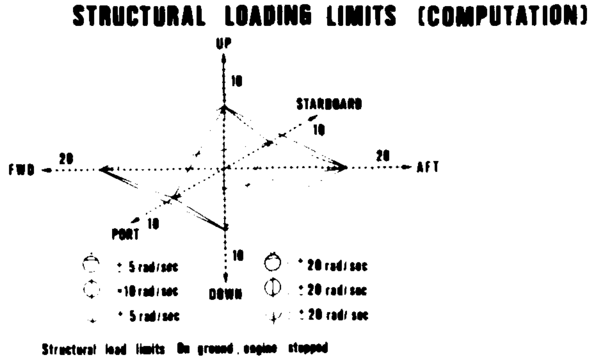
Pour l'application mer-mer, la mise en vitesse est obtenue par la fusée d'accélération du missile : à la fin de cette phase, le missile atteint une altitude de quelques centaines de mètres et une vitesse voisine de la vitesse de croisière.

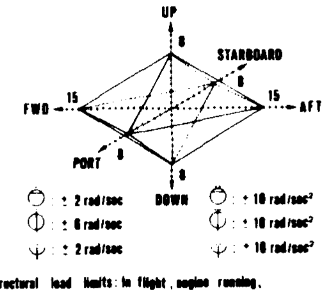
La planche 9 schématise la variation des paramètres altitude et vitesse en fonction du temps. La validation de la possibilité de démarrage en autorotation dans de telles conditions est faite en utilisant un chariot propulsé par fusées, sur un rail, supportant une maquette du missile comportant le réacteur et l'entrée d'air réelle. Cette installation a été réalisée par le CEL. La phase de propulsion accélérée dure moins de 3 secondes sous une accélération pouvant être ajustée à la valeur requise, suivie d'une croisière de 2 secondes au moins à vitesse légèrement décroissante, terminée par un freinage hydraulique.

La satisfaction des spécifications de mise en oeuvre rapide en autorotation impose pour sa part au missilier de fournir au plus tôt le carburant, la durée de transit dans les tuyauteries et la rampe d'injection pouvant atteindre ou dépasser le temps de démarrage proprement dit.

3.3. Accélérations en fonctionnement

L'application du TRI 60 à des engins catapultés a imposé la démonstration de ses capacités de supporter des accélérations longitudinales importantes tant à la mise en vol qu'à la récupération. Ces essais ont été réalisés au CEAT sur une catapulte pneumatique et les résultats obtenus sont résumés à la planche 10.





TRI 60 CATAPULT LAUNCH TEST

ACCELERATION	DECELERATION	QTY LAUNCHES
4g	≥ 6	3
8g	≥ 12	7
10g	≥ 16	10

Planche 10

3.4. Immersion et refourbissage

Il est essentiel que le réacteur d'un engin-cible puisse être remis en oeuvre après une récupération en mer. La planche 11 résume le cycle auquel est soumis en essai un réacteur TRI 60. L'installation sur missile est conçue pour que l'ensemble de ces opérations s'effectue sans dépose du réacteur ; la révision en atelier est envisagée après 10 tels cycles.

CYCLE FONCTIONNEMENT IMMERSION REFOURBISSAGE
OPERATION - IMMERSION - REFURBISHING CYCLE

FONCTIONNEMENT OPERATION	≥ 30 mn / min
IMMERSION DANS L'EAU SALEE IMMERSION IN SALT WATER	= 3 HEURES / HOURS
AIR AMBIANT EXPOSURE TO AMBIENT	= 1 HEURE / HOUR
ARROSAGE EAU DOUCE OU IMMERSION EAU DOUCE WASHING WITH FRESH WATER OR IMMERSION IN FRESH WATER	de 5 à 15 mn from 5 to 15 min
VENTILATION SECHE DRY CRANK	de 1 à 2 mn from 1 to 2 min
TRAITEMENT FINAL END TREATMENT	de 3 à 5 mn from 3 to 5 min

Planche 11

3.5. Vibrations

Tant pendant le vol porté ou le stockage sur un pont de bateau, que pendant le vol du missile, le réacteur est soumis à un environnement vibratoire important.

Cependant, la validation de la tenue du réacteur est étroitement liée à la manière dont est conduit un tel essai et en particulier à la façon dont les vibrations sont transmises par la structure : un tel essai se fait en étroite collaboration avec le missilier.

4. POURSUITE DES PROGRAMMES

Dans le cadre des quatre programmes utilisant le TRI 60, un certain nombre d'essais est en cours ou reste à faire :

- diagramme des vitesses et températures du jet,
- signature infrarouge, naturelle et augmentée,
- ingestion d'eau,
- attitude,
- ...

ainsi que la validation de toute nouvelle solution assurant une meilleure adaptation à la mission impartie.

5. EXPERIENCE DE PRODUCTION

Dès son origine, le programme TRI 60 a été conduit avec le souci principal d'une production à faible coût.

Des solutions originales ont été développées, certaines n'ont pas été retenues, comme par exemple le collage réduisant la démontabilité. Mais la qualité essentielle réside dans la simplicité de l'architecture : le TRI 60 comporte en effet 20 pièces principales (planche 12) :

- alliage léger taillé 6
- alliage léger coulé 5
- acier spécial coulé 2
- mécano-soudure 7

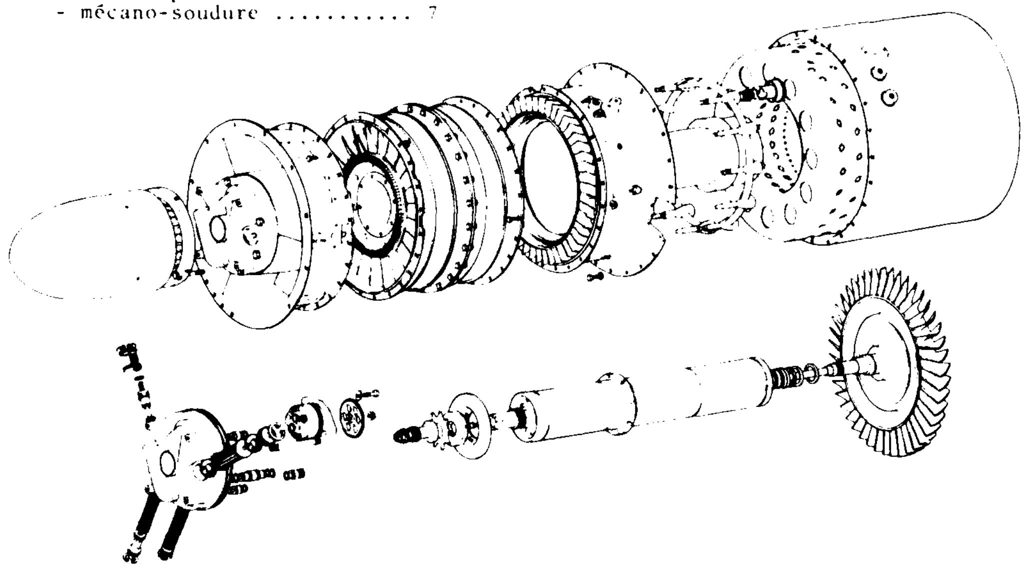


Planche 12

Dans la phase de développement et en l'absence d'outillages importants, certaines pièces ont été obtenues par usinage conventionnel : au fur et à mesure de la création des outillages, les pièces de production ont été incorporées dans le réacteur et soumises à des essais de qualification de performances et d'endurance accélérée.

En conclusion de cet effort de conception, le TRI 60 est de loin le moins cher des réacteurs à "durée de vie limitée" de sa classe en regard des performances et des qualités offertes.

Ramené aux conditions économiques de 1972, le prix au daN du réacteur TRI 60, en production de série, est égal à 102 francs, l'objectif global fixé il y a 8 ans a donc été atteint (planche 13).

CARACTERISTIQUE CHARACTERISTICS	VALEUR PROJETEE TARGETS	VALEUR OBTENUE ACHIEVED	ECART DIFFERENCES
DUREE DE VIE ENGINE LIFE	20 Heures/Hours / 50 Cycles	20 Heures/Hours / 50 Cycles	NEANT NONE
PRIX AU DECANEWTON COST PER DaN OF THRUST	100 Francs	102 Francs	2%
DIAMETRE MAXIMAL MAJOR DIAMETER	330 mm	330 mm	NEANT NONE
MASSE TOTALE TOTAL WEIGHT	50 kg	45 kg	-10%
POUSSEE MAXIMAL MAXIMUM THRUST	300 daN	370 daN	+23%
CONSOMMATION SPECIFIQUE S-F-C	1,30	1,30	NEANT NONE

Planche 13

DISCUSSION

J.C.Ripoll, D.T.C.A., Fr

The engines mounted on vehicles are used after a long period of storage, virtually new, and must demonstrate the reliability of their starting potential in this state. How is this taken into account, if necessary, in the tests which are normally conducted at a fast rate on a single machine?

Author's Reply

During the first part of the development programme, starting tests were conducted, close to each other, with a single engine for a given installation, even when tests were repeated with different engines, this was only demonstrating starting reliability of a new engine.

Starting Reliability of a long-stored engine will be demonstrated with the following programme:

acceptance test of engine as new;
preparation for storage life;
simulation of accelerated storage life (up to 6 years in a period of 6 months);
demonstration of starting reliability in a simulated environment.

A total of 4 engines will be used in 2 main programmes.

Albert A.Martino, N.A.P.C., US

Regarding the "one-shot missile" — how do you plan to lubricate this engine, and how do you plan to experimentally verify this capability?

Author's Reply

TRI 60 is basically designed with a lubrication-loop system which necessitates an oil capacity.

In the case of a target-drone application i.e. a flight duration capability of more than 2 hours, an oil tank is supplied which is wrapped around the compression section and has a useable capacity of 600 cc.

For a "one-shot missile", the flight duration will not exceeed 40 minutes. The air intake casing capacity of 200 cc is quite sufficient as the average oil consumption is 40 cc and piping and bearing capacity is less than 80 cc.

R.H.Blake, Rolls-Royce Ltd, UK

What sort of Production Proof testing is carried out in the light of the short life of the engine?

Author's Reply

At the end of the development and evaluation phases the TRI 60 propulsion system will be frozen for the corresponding application.

This definition will then be launched in production and the first 4 units will be used to conduct a complete programme of production qualification summarizing the main steps of the initial phases as:

performances in the flight envelope;
starting reliability;
vibrations and shocks;
environmental conditions;
long storage life.

PREDICTION OF FUTURE TEST NEEDS, TEST FACILITIES AND PROCEDURES

by

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INTRODUCTION

The requirements for future military turbine engine testing will reflect a more reasonable and rational balance between the types of validation necessary to provide satisfactory operational weapon systems. In the past a large emphasis has been placed on the aerodynamic and thermodynamic aspects of the engine. Durability and reliability have tended to be assigned lesser priority. This situation has led to problems and failures of engines in operational service.

These problems have resulted in a re-examination by the USAF of the type and methods associated with qualification of military gas turbine engines. During this examination it became evident that re-emphasis on the durability aspects of the engine was necessary. It was also recognized that a balanced approach to address all aspects of engine requirements was the best approach to meeting the needs of the USAF operating and support organizations.

TYPES OF TESTING

The development and qualification testing of turbine engines for military applications and indeed for commercial applications can be classified in three broad categories, Performance, Operability, and Durability. Performance is generally considered the thrust, airflow and fuel flow associated with the particular design. These parameters are most critical for system range, payload and maneuverability. Operability is defined here as response to throttle and varying engine inlet conditions. This area is heavily concerned with the engine control equipped with an afterburner or augmentor, the characteristics of that device. Durability involves the rest of the engine requirements such as low cycle fatigue life, stress rupture or creep life, Foreign Object Damage (FOD) resistance, and other mechanical aspects of the design.

As a broad generalization, each of these categories of testing can be considered equally important. However, when a particular weapon system or application is involved, some types of requirements can be emphasized over others. For example, the direct operating costs of transport aircraft are largely driven by the amount of fuel consumed. Thus, a transport engine program will most likely concentrate heavily on the performance aspects during development. A fighter aircraft depends on a smoothly operating and responsive engine over a wide variety of operating conditions. In such a case, the operability side may receive increased emphasis.

However, it is important to recognize the necessity for keeping the durability part of the program on an equal level with the other areas. It makes little sense to have an efficient, smooth responding engine if it is continually in the shop undergoing repairs or maintenance.

PAST PROCEDURES

Performance testing has been relatively successful in the past. While it is a difficult and trying problem to assure the necessary accuracy and repeatability of measurements of engine airflow and thrust, facilities such as USAF Arnold Engineering Development Center and the USN Naval Air Propulsion Center have been able to provide excellent measurements of these parameters to the satisfaction of all concerned.

The area of operability has received increased emphasis in the past ten to fifteen years. It is not coincident that this time period has seen the introduction of augmented turbofan engines. The interaction between the augmentor and the compression system has resulted in increased complexity and potential problem areas. Use of inlet simulators and screens for generation of distortion have been the prime method of evaluating the operability characteristics of the engine.

Typical recent development programs have called for evaluation of the engine's stall margin, augmentor blow-out and relight characteristics and main engine blow-out and airstart envelopes.

The final area of durability has been addressed to a lesser extent in past programs. Heavy reliance was placed on the 150 hour endurance test as the primary indicator of engine life and reliability. As the engine thrust to weight ratio has increased and increasingly sophisticated and exotic materials and manufacturing processes have been introduced into engines the 150 hour test has become less reliable as an indication of the durability of the engine. Failure modes such as low cycle fatigue and rapid crack propagation were not adequately addressed in these types of tests.

The 150 hour endurance test tended to concentrate on the time-at-temperature aspects of the engine, that is creep and stress rupture of failures. This was a recognition that early turbine engines were limited by such failure modes. Therefore, the durability testing was driven to show that the engine hot section (combustor and turbine) could withstand the required times at elevated temperature and only a limited examination of the cyclic loading of the engine was made.

CURRENT PRACTICES

The area of performance testing appears to have reached a plateau. Current facilities have adequate capability to define engine characteristics to a level sufficient for most needs. Additional refinements are necessary and are continuing primarily in data acquisition and reduction. There is also a need to handle the smaller engines that are being used in such systems as the Air Launched Cruise Missile, and New Generation Trainer. The ability to provide comparable measurements on an engine that produces 600 pounds of thrust as opposed to 40,000 pounds of thrust is a difficult task. The primary problem is one of accuracy. A five pound error in thrust is negligible on the large engine but is almost one percent on the small one. The discrepancy increases with altitude to the point where the same absolute level can be a five percent error.

Operability is being addressed via the use of screens and inlet simulators. This type of engine-inlet testing is extremely useful when coupled with the analytical tools to predict engine stability. The use of screens has a drawback since they only represent the inlet at one condition. As the engine airflow changes, the resulting distortion from the screen changes. Thus a series of screens is necessary to cover the extremes of the envelope. The use of an inlet simulator is a partial answer to this problem.

The problem with such a device is a possibility it will not reflect the real aircraft configuration. Another objection to the use of a simulator is its inability to provide varying angle of attack capability where most of the distortion is generated. Finally, since the engine design must start earlier than the aircraft and the inlet may undergo several iterations, the expense of such a simulator may not be warranted.

Another area of concern in operability testing is the capability of the facility to maintain the proper conditions of inlet temperature and pressure while the engine is undergoing a throttle transient. With airflow changes of 500% common in turbofan engines, it is difficult to supply air that represents a constant flight condition or predictable flight path.

For these and other reasons, while operability testing in current facilities is thorough, a heavy dependence on flight testing is necessary. It is in the operability area that the preponderance of propulsion system flight testing is concentrated particularly for fighter and attack aircraft. Flight test is currently the only way the true inlet and environmental conditions can be experienced.

It is in the area of durability testing that the most recent changes in USAF development and qualification testing have occurred. The current series of engines recently developed including the F100, F101, and F404 have adopted a procedure known as the Accelerated Mission Test or AMT. This test replaces the historical 150 hour endurance test. The AMT philosophy is to attempt to identify the structurally damaging events caused by power level changes and time at high temperature and duplicate them. In this type of test, a one for one correspondence with the time at high temperature and major power level changes (near idle to military power and return) from actual or projected use is achieved. Less damaging time such as cruise power, aircraft warm-up and taxi descent time is eliminated. While this time may contribute to wear out and eventual failure of bearings, gears, seals, and the like, it is believed that attempting to evaluate these modes in an expensive engine test is not cost effective. The approach in current use is to conduct specific rig and component tests on these types of devices. This provides the necessary real time information associated with wear out modes.

The use of AMT has increased significantly within the USAF in the recent past. Not only do the current development engines mentioned earlier rely on AMT procedures but also existing older operational engines such as the TF30 and TF41 are using AMTs to investigate service problems and potential solutions. The largest benefit of the AMT is the ability to accumulate large amounts of equivalent time on the high cost and critical parts of an engine. Determining the life of such essential items as discs, turbine airfoils, fan blades and pressure vessels provides the needed confidence for predicting a successful service experience with the engine.

Being able to accelerate the time necessary to verify low cycle fatigue and time at temperature life reduces both the time and cost associated with verification testing. A recent four week test on the USAF development engine resulted in the accumulation of the same number of low cycle fatigue cycles and time at high turbine temperature as the engine

would typically accumulate in three years of service use. This is due to two factors. The test was run on a round the clock schedule with the only interruption for routine servicing and inspection. The second factor was the acceleration or compression of time associated with the AMT philosophy. This results in a 2.5 acceleration for a typical fighter aircraft to 10 or more for a transport engine. In other words, one hour in an AMT program represents 2.5, 10 or more hours in service use.

FUTURE APPROACHES

The area of engine performance testing appears to be on the most solid ground regarding current capabilities vs projected future needs. We expect that with the ever increasing fuel costs, a renewed emphasis will be placed on factors that effect fuel consumption. Items such as active clearance control, more efficient compressors and turbines and reduced cooling system losses will need a thorough evaluation.

This will result in the current testing techniques for performance qualification being generally adequate for determination of the overall engine characteristics. Much more detailed development testing will be required to define and improve specific components of the engine. I expect that the need will exist for advances in instrumentation and measurement techniques to define which area requires additional development and refinement.

The current state-of-the-art is such that it is difficult to determine where performance shortcomings exist. The use of rig tests to define component performance levels is an area of promise that will likely see increasing use. Such testing can result in a sound understanding of the basic capability of the component. The problem is and continues to be one of how the engine as a system responds. Difficulties in analytical predictions of the effects of parasite losses and cooling flows can have a significant impact on the overall performance of the engine. This is where the emphasis will need to be placed in future performance testing.

The problem is especially acute in the smaller engine. It has been said there is one thing wrong with heavily instrumenting engines. The instrumentation disturbs the flow and then it tells the observer what it is. With flow passages in a small engine of below a centimeter in size, it is difficult to obtain accurate measurements to determine whether the design intent is being met. Perhaps the only solution is the use of optical instruments such as laser velocimeters and pyrometers.

Operability testing remains in an evolutionary state. While the use of screens is adequate for go- no go type testing, a need exists to develop techniques and equipment to more realistically simulate the actual environment the engine sees. Two facilities in the USAF that will address this problem are currently being developed, the Aeropropulsion System Test Facility (ASTF) at Arnold Engineering Development Center (AEDC) and the Compressor Research Facility (CRF) at Wright-Patterson AFB.

The ASTF is being designed to provide the capability to more closely follow the transient characteristics of the engine. This should permit a truer determination of the engine's capability to respond to rapid throttle transients and inlet pressure and temperature changes. This facility is also planned to have the ultimate capability to test in a free jet mode. Entire inlet systems can be installed in the test chamber. Variations in angle of attack and sideslip can be evaluated prior to flight test. The facility will also have a sophisticated data reduction system to evaluate on line the results of tests.

The CRF will be capable of testing the compression system with the actual inlet pressure and temperature expected in flight. This is in contrast to the standard practice of operating at reduced inlet conditions in order to meet available power requirements. This facility will thus result in an early definition of the compression system performance, operability, and blade vibration characteristics and thus reduce the risk for a full scale development. It will also evaluate promising research avenues in the most realistic environment short of an actual engine.

Another test technique that will come into prominence is control system rigs. Such rigs permit a complete simulation of the controls, actuators and sensors of a turbine engine. As additional complexity such as variable cycle designs and coupled engine inlet controls, the need will continue to grow to evaluate the total system response. In the past, we have occasionally been surprised by the control integration task and the way various control component and sensors have communicated. The system rig can go far to uncover these pitfalls before they have the chance to cause an engine failure.

Operability testing will continue to rely heavily on flight test. Regardless of the capability to better simulate inlet conditions on the ground, the final answer can only be determined in the actual environment. The advances in ground test will reduce the risk of problems, but it still will be necessary to conduct a thorough flight test program. If the engine is not thoroughly evaluated in flight with the entire aircraft system functional, the developer can be led into a false sense of security and be surprised by operational problems down stream when the solution may be extremely difficult to implement.

The final topic involves the use of durability testing. The major new thrust concerns the application the damage tolerance or fracture mechanics approach to design of critical parts of the engine. This approach, currently being applied

to the F100 engine and planned for future engines, is associated with the requirement to assume that a flaw that is just below the level of inspection detectibility can exist in the part. An analysis is then performed to determine how long the engine can operate with the crack growing due to cyclic loading or temperature loads. Inspections of the parts are then scheduled to prevent the crack growing to a critical size in operational service.

The use of this approach will rely heavily on the AMT and engine internal environment measurements to verify crack growth rates in the engine. To insure the accuracy of the predicted growth rates, a better understanding of the engine environment is clearly necessary. This will require the use of heavily instrumented test engines to measure the temperatures and strains the part undergoes in the engine. The AMT is the primary tool that verifies the actual crack growth rates. Future AMTs will include the actual running of cracked rotating parts such as spacers and discs to verify the crack propagation rates. This will also be increased reliance on component testing in spin pits or "ferris wheels" to load the discs and shafts and determine crack growth rates. A key need is to develop facilities that accurately simulate the rapid thermal changes that an engine part sees in a power cycle. Current pits with few exceptions have relatively slow thermal response characteristics or unable to accurately simulate gradients from bore to rim. They are also required to run under evacuated conditions to limit the drive horsepower requirements. This is an acceptable limitation since the centrifugal load and airfoil gas load can be adequately simulated in current designs. As new and novel blade attachment designs appear, this limitation will require re-examination.

The USAF is considering construction of a new facility to evaluate the impact of maneuvers on engine durability. The Turbine Engine Load Simulator (TELS) proposed for AEDC would subject a full engine to maneuver forces by rotating the engine at the end of an arm while simultaneously spinning the engine. The facility is also planned to have X-Ray capability to check internal motion and clearances. Such a test capability would be used to examine bending loads induced by g forces and gyroscopic moments.

Flight tests also has a role to play in the durability area. AMTs are based on projected missions and the damaging events determined from these missions. As tactics and operational missions evolve, it is occasionally necessary to update the AMT to reflect this fact. Flight test can provide an early verification of the projected mission. One of the current USAF limited development programs has as a prime objective conducting a series of flight tests to validate the mission used to evaluate the engine design. It is essential that such flight test objectives be recognized and completed to improve the insight into the engine design and expected life.

I would like to make a final note about durability testing. We must recognize a turbine engine is a very sophisticated and complex device. This fact should cause us to expect occasional surprises and failures. No matter how hard we try and how sophisticated our techniques become, the unknown unknowns will still be there. We can and will do better but perfection is probably beyond our reach.

CONCLUSION

A fact to remember as we evolve our test techniques is that the test is the proof of the analysis. The two must go hand in hand. Testing for testing sake is not productive. Each test must confirm or deny an analysis. If it does not, then the test becomes meaningless since there is no method to predict what will happen as the situation changes. Tests will become more sophisticated and realistic but unless the requirement of confirmation of analysis is paramount, we can wind up with failed engines with no idea why it happened or what to do about it.

DISCUSSION

J.C.Ripoll, Direction Techniques des Constructions Aéronautiques, France.

My question is related to a remark made by Mr Mihail:

You indicated that the conditions of employment of an engine vary considerably from aircraft to aircraft, and from user to user. Given this, is it possible to codify the longevity tests to produce a norm for certification?

Author's Reply

We have been examining various tests and missions and find, as you may expect, they fall into two general categories, transport or fighter/attack. Transport testing leads to general procedure that is comparable from system to system. Thus it probably can be standardized. The fighter/attack engine is more system peculiar with some general similarity. Thus the fighter/attack test will be more system oriented and less general.

We will be attempting to establish general tests in the transport and fighter/attack areas in the future. It is a difficult problem and will require considerable effort.

DEVELOPMENT FOR NEW LABORATORIES FOR FUTURE TESTING

by
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SUMMARY

The project philosophy of the propellers in general will evolve in the future, as in the past, keeping in mind the three fundamental characteristics of aerospace systems: weight, volume, power. Considering the use of the propeller, (military or civil), two other eminently economic factors must be taken into consideration: the lowest possible obsoleteness for military use, the lowest cost of utilization for civil use. Themes of theoretical and experimental research:

a) about the components relative to only the energetic system: subsonic and supersonic combustion processes; traditional and non-traditional combustibles, interaction between the mechanical and combustible system; with combustion in axial fluxes and deviated fluxes; spark ignition; initial combustion and relative phenomena; transitory of the combustion, the combustion regime;
b) of the fluid-dynamic system components: isolated blades of the compressor and turbine with a small rapport 1/d in subsonic and supersonic fluxes; elevated load blades in subsonic and supersonic fluxes; annular formation of the compressor and turbine;
c) electronically servo-assisted automation during the transitory phase of any process.

1. INTRODUCTION

The definition and the development of the design of aerospace propulsive devices must take into account two factors which are relevant to the general economic context: the lowest possible obsoleteness for military use, the lowest cost of utilization for civil use. These two factors are strictly connected with the characteristics of the aerospace systems: power, weight, reliability and maintenance. The design effort must of course be based on studies and researches which contribute to the development of alternative solutions and technological integration within the context of the objectives of the operative developments and to the forecast of the impact that the new information can exert on the future uses of the various kinds of engines and the new opportunities that may rise in the future, always keeping under control the evolution of the technological and scientific information all around the world.

Starting from the "basic research" it is possible to pick out applicable possibilities from which branches of "orientated research" are developed, with not only the hope - but perhaps it can be claimed - with the certainty, to create innovations for the system project; at this point "finalized researches" can be developed, either sectorial or global, of the system in which Institutions or Industries can be interested.After the development of the research carried out in the way mentioned above, two other phases have to follow: the "diffusion of the research" and the "development of the prototypes". The detailed procedure mentioned above has been presented in Fig. 1-1. Basic and finalized research in the long term plays a primary role in the engineering field and it is of great support for the industrial product. In fact, considering the innovation process and focusing on the industrial product, one can think of a multilevel "hierarchical design"and, passing through various phases, one get to an "optimal design" which, putting in discussion the product in relation to the required goals, widens the field of research to the whole system. The approximative time of the research,of the design of the system and the subsystem, are reported in Fig. 1.2, with specification of the various kind of intervention.

2. FUTURE DEVELOPMENT OF AEROSPACE PROPULSION SYSTEMS

Two general concepts on the fundaments and on the research development, in its various aspects and stages, have been underlined in order to identify the necessary times for realizing the "aerospace system" proper for the use. The system will have to be qualified in the external context in order to be fully efficient and has to be profitably used for 8-12 years after its realization and, therefore, the installations and instrumentations will have to be proper to satisfy the experimental needs for a variable period of time from 0 to 10-15 years. The basic needs requested to carry out these tests are:

a) compressed air plants to drive subsonic transonic supersonic and transonic wind tunnels;
b) compressed air plants to carry out researches on subsonic and supersonic combustion;
c) installations for producing energy of many kinds for various sperimentations not forecasted nowadays;
d) installations for studies and researches of uses and systems not easily forecasted nowadays;
e) instrumentations for the diagnostics of experimental measurements;

f) data acquisition systems.

It is necessary to underline that a series of laboratories, able to solve the experimental needs mentioned above, have such a high cost of basic structures, that the not utilization of them for researches and studies of systems close to the aerospace field, would turn out to be antieconomic.

3. AIRCRAFT PROPULSION

The propulsion which uses atmospheric air as supporter of combustion must adopt different types of propulsive systems, according to vehicle speed. Such systems need different components which themselves require different researches and experimentations. In Fig. 3 various types of aircraft propulsion are represented with the components of each system.

3.1 Turbomachinery for subsonic, transonic and supersonic propulsion

It is easy to foresee that the experimental research on turbomachinery which is useful to integrate and assist computational methods will be one of the most important trend of propulsion laboratories in the future. Such opinion is based on the following considerations:

a) the flow behaviour in a turbomachine is a very complex physical phenomenum; in spite of the recent progress - due to the presence of high speed and capacity computers of the 4th generation - the actual know-how of theoretical research in the field is not so advanced for thinking that very important progress can be reached only theoretically. Very important phenomena as secondary flows, flow unsteadiness, blades interactions, turbulent and inviscid effect, are still studied with the use of very simplified models. For this reason, in the future the experimental work on real machine will give the necessary inputs to mathematical models.

b) the actual world energy situation demands high efficiency in any process which uses any form of energy. In all these processes, except few cases, there are turbomachines whose performances are consequent to reaching good thermodynamic efficiency. Therefore it is necessary that, in the near future, the propulsion laboratories will be able to test every type of turbomachine, also those which are not directly used in the propulsive systems. A reconversion process should be done, in which laboratories specialized in fluid-dynamics carry out also researches on turbomachinery mechanics. As said before, a new turbomachinery laboratory must be as flexible as possible allowing to operate in different fields and on different turbomachines with minimum modifications avoiding large money investments. Therefore, the laboratory should support "service acquisition" rather than the building of unflexible test rigs useful only for particular experiences.

In the future the testing on real turbomachines or on models which reproduce the fluid-dynamic phenomena in its whole complexity will be more and more important, rather than the experimentation on partial test rigs which are able to reproduce only models object of previous theoretical approach (for example see the cascade).

Of course the experimentation on a real machine can be very arduous for many reasons: difficulty of a proper instrumentation of an industrial turbomachine, power requirement too high in the case of driven machines or mass flow rates too high in case of turbines. It is possible to overcome - at least partially - the first difficulty, building a model which already presents the necessary modifications for a correct instrumentation (optical screens, probes into rotating elements, etc.). The similarity theory can help in such a work because it allows to vary the machine dimensions, the pressure and temperature of working fluid reaching values compatible with probe and transducer physical characteristics and, above all, compatible with the stresses of rotating equipment in order to reduce the costs.

As without great fluid-dynamic forces, the stress of a rotating element is function of peripherical velocity u following

$$\sigma = k \, \rho_m \, u^2$$

where k is a function of the dimensions, ρ_m is the rotating element material density, the convenience of reducing the peripherical velocity is evident. In a kinematic similarity this reduction means a decrease in the flow velocity. Following the compressibility laws, i.e. same Mach numbers in the model and in the real machine, the gas sound velocity formula

$$a = \sqrt{\gamma \frac{RT}{M}}$$

shows the possible ways to overcome the problem:
- reduction in temperature (not very used and not very useful due to the low consequent reduction in velocity)
- adoption of fluid with high molecular mass.

This second possibility allows to reach supersonic velocities with reduced mechanical stresses. For example using a fluid with a molecular mass M = 500 at ambient temperature a sound velocity of 70÷80 m/s can be obtained and in consequence mechanical stresses 15÷30 times lower than in the case of turbomachines working with air or steam.

To follow the similarity laws and thus to reproduce the link between the pressure and the specific volume during the thermodynamic transformation, it is necessary to use sometimes fluid mixture of very complex fluids with monoatomic ones. In any case it is necessary to operate in a closed cycle with many problems due to the piping. It should be reminded that if the model works at low revolution speeds, also the problems concerning sealing devices and bearings are greatly reduced.

In addition to the possibility of building a rotor with less stresses, the adoption of a closed cycle allows another very important advantage: the possibility of changing at will the pressure value of the plant. Because of similarity laws, this means a change in the turbomachine power due to

$$\frac{P}{P_o} = \sqrt{\frac{M_o}{M}}\ \sqrt{\frac{T_o}{T}}\ \frac{p}{p_o}$$

where P_o, M_o, T_o, p_o are the power of the real machine and the mass, temperature and pressure of the original fluid, and P,M,T,p are the same parameters of the model. Supposing for example a ratio $p/p_o = 0.1$ and $M/M_o = 0.05$ a power equal to 2,2% of the real machine power can be achieved with a model of same dimensions and operating with the same fluid-dynamic In conclusion, the adoption of a closed cycle and of a working fluid of heavy molecular mass, in respect to similarity laws, allows the building of models not very stressed and thus the use of materials of better machinability, such as aluminum alloys, with cheaper mechanical solutions and a lower input or output power of the turbomachine.

Another experimental possibility is that of maintaining the actual working conditions (fluid, pressure, temperature, velocity and power), but performing the whole experimentation very quickly. This procedure (blow down) allows to fit the laboratory with a power lower than that of testing, using accumulating systems: pneumatic systems with fluid stocking in high pressure cylinders (see Appendix I) or mechanical systems with the use of a flywheel (see Appendix II. The blow-down technique demands a very sophisticated regulating system which allows to minimize the set up of the instrumentation and the timing of the machine (1). Such a system needs also the development of an acquisition data system able to collect and process many signals shortly and at the same time. Moreover, the laboratory should have a high technology in processing computers and efficient equipments for probe and transducer calibration (2).

3.2 Aeronautical propulsion in the subsonic, transonic, supersonic and hypersonic range: internal aero-and gasdynamics

Analysis and design of the components of such propulsion systems must be aided by means of careful experimental tests taking place in blow down type wind tunnels (see Appendix I and II). Test sections must be large enough to avoid wall effects on the model, among the other things also boundary layer suction at the walls can be used.

The experimental apparatus must be able to simulate the flight in all the conditions; Mach number, Reynolds number, heat flux, and, if necessary, dissociation and recombination phenomena must be considered.

Stagnation temperature should be considered in the experiments not only in connection with the aerodynamics phenomena but also because of the thermal shock of the structures. Dissociation phenomena should be considered only if Mach number exceeds 8 ($M>8$), (hypersonic flight) and then more in the field of spatial than of aeronautical propulsion.

In the range $0,5<M<4$ test section cross section must range from 1x1 m. to a minimum of 0,5x0,5 m. with maximum stagnation temperatures of the order of 800°K. Test section must be equipped to analyze the forces (in all directions), pressures, temperatures and aero-and thermoelastics deformations. In the case of $4<M\ 5$, test section size must be not less than $\Phi = 30$ cm; taking into account the axialsimmetry of the propulsion system, only a sector of the component could be tested, allowing in this way the use of a larger model. The stagnation pressure and the stagnation temperature are of the order of 100 bar and 800°K rispectively in these wind tunnels. The air flow must be heated by means of an alloy steel heater, with an aluminum-bronze bed inside.

Beyond these Mach number we are in the field of spatial research; in fact, atmosperic propulsion, till heights of 20.000 ÷ 30.000 m., can not overcome these Mach numbers also because of drag and boundary layer phenomena.

4. EXPERIMENTAL TESTS IN SPATIAL PROPULSION

Spatial propulsion is characterized by hypersonic speeds and by the use of not airbreathing motors.

Equipment to be used to analyze missile performances and connected phenomena are mainly blow-down type wind tunnels but auxiliary systems are required.

4.1 Aero-and gasdynamics of the spacecraft

These tests, also if not strictly of propulsive type, are connected with the behaviour of rocket propelled systems. Phenomena such as the missile or satellite reentry in the atmosphere, gas ionization and then spacecraft control, vacuum effects at high

(1) These basic installations used for such experimentations are used not only for researches on turbomachinery, but also for experimental researches on combustion, intakes and nozzles.

(2) In Appendix III an installation based on the proposed criteria is described. This installation is now being prepared at CNPM.

altitude (80÷400 km) sun radiation and interaction of all such phenomena with spacecraft behaviour and control should be studied.

In the range 6<M<12 an axialsimmetric water cooled wind tunnel of the same type of those employed for M<6 can be used. The stagnation pressure is of the order of 100 bar and the air flow is heated at 1500°K by means of an electric heater equipped with Konta resistances. Of course the wind tunnel working at these Mach numbers must discharge in a closed space with a pressure of 0,5÷1 mmHg, spheres of 9÷20 m diameter, according to the length of the tests, are used.

In the case of M≃20 wind tunnels working operating at the required conditions (speed and stagnation pressure) per interval time in the range 0,1 ÷ 2-4 sec can be used.

The electric arc should be obtained or by means of batteries or following the system of Appendix II. In the case of Mach = 20 test sections should have a diameter of the order of 60÷80 cm, and a stagnation temperature of about 6.000°K. A vacuum system working in the range $10^{-4} \div 10^{-8}$ mm Hg must also be provided. Fig. 4.1 shows the characteristic behaviour of a vacuum system for spheres or other vessels; the degree of vacuum is on the horizontal axis, while the air mass flow at the intake of the vacuum pumps of suitable characteristics is reported on the vertical axis.

The arbitral flight simulation can be obtained in a large cylinder (3÷6 m diameter, 4÷8 m length), cooled at the inside wall by means of liquid nitrogen to reach temperature as low as 70-100°K. The vacuum must be of the order of $10^{-6} \div 10^{-8}$ mm Hg, while sun radiation can be simulated with xenon-hg arc lamps (1).

5. COMBUSTION FOR AERONAUTICAL AND SPATIAL PROPULSION

Fig. 5.1 shows the different types of aeronautical and spatial propulsion also if a sharp division is rather difficult to be made. In fact rocket propulsion is used also in the field of the aeronautical propulsion, in the case of air-air, air-ground,ground-air missiles, or to obtain suitable initial velocities for the ramjets engines (ramjet with liquid rocket engine or hybrid ramjet with solid propellant rocket engine).

The planning of a combustion research facility should take into account not only theoretical and applied research systems, but also of industrial ones. Of course it is necessary to be able to acquire a large number of experimental data to improve knowledge of the thermofluiddynamic phenomena and then to mathematically model them.

Tests with real systems are possible only with large and expensive support (i.e., air mass flow rate, available energy, etc.) but allow to overcome many problems connected with the use not full scale modes, not always possible (see Appendix I and II). Compromising solutions are often necessary (i.e. the use of only one part of an anular combustion chamber), but they must give good results without alteration of the physical phenomenon.

It is now possible, in the combustion field, to obtain measurements of high spatial and temporal resolution as:
- mean velocity (components);
- fluctuating velocity and distribution function;
- fluctuating temperature and distribution function;
- spatial concentrations (also from the air pollution point of view);
- behaviour of the particles and of the sprays (fuel spray, coal particles, etc.);
- spatial and temporal correlations among different quantities.

Now and in the future large experimental efforts should be devoted to the analyses of the turbulent flow field, mainly because of the necessity of obtaining better turbulent models for the numerical simulation.

Besides the classical diagnostics instrumentation (probes, hot wire anemometer, etc.) experimental facilities should be equipped with sophisticated electro-optical instrumentations as:
- laser anemometers (connected with suitable data acquisition and computer systems) able to study the sub-trans and supersonic field measuring at the same time three velocity components;
- diagnostics systems based on Raman scattering (rotational, vibrational, C.A.R.S.);
- other systems based e.g. on Rayleigh scattering, fluorescence, radiation absorption,etc.

Optical diagnostic techniques, in large use in the future, require in many cases high power sources and it seems to be convenient to centralize them and to distribute the light by means of suitable optical ways (optical fibers, mirrors, etc.) to the different experimental test bench.

The analysis of fluctuating and transient phenomena requires the acquisition of a large amount of experimental data to be then worked out; a centralized data acquisition and treatment system seems to represent the best solution for such type of facilities. Fig. 5.2 shows schematically the centralization of these services.

5.1 Air-breathing combustion

Diagnostics techniques of the type previously analyzed can be applied to different combustion phenomena.

(1) In order to reproduce the intensity of radiation spectrum at 400 km of altitude, 20 of these lamps are needed.

a) Periodic combustion (reciprocating engines)

Analysis of the flow field in isothermal and not isothermal conditions, controlling the influence of the combustion chamber configuration, intake and discharge timing etc. It can be analyzed in greater detail with respect to the past, the ignition, the influence of turbulence and the flame propagation, the fuel and air mixing etc. In the case of Diesel engines it is necessary to study the sprays deeply (heating and combustion of the droplets), to analyze the flame propagation and exhaust gas composition. Such an analysis is able to give information allowing e reliable mathematical modelling of engine processes; models can also be used to design the engine (if they are reliable).

b) Continuous combustion (turbojets and gas turbines combustion chambers)

Besides the flow field, it is also possible to analyze droplet combustion, flame propagation and stabilization etc. It is also possible to go deaper into the interaction between droplets and gas movement both in isothermal and not isothermal conditions, and in the mixing processes in primary and secondary regions. Mathematical models of the combustion chamber can be improved starting from the new information obtained.

c) Supersonic combustion (ramjet)

The analysis in this case is more complex because of the necessity of the study of the starting process (liquid rocket engine or solid rocket engine in the hybrid ramjet)(1). Also in this case local and global stability phenomena, instability propagation, etc. should be studied. The matching between internal and external aerodynamic must be carefully analyzed because of the fact that the flight path of the airplane can influence the flow field in the ramjet leading to flow separation phenomena. Mathematical modelling in this case not only contributes to the improvement of ramjet design, but also to the guidance technique and then to the flight mechanics of the ramjet propelled airplane.

5.2 Combustion in rocket engines

Advanced rocket propulsion, particularly in the field of military research, is mainly oriented toward solid propellants, which have now all the characteristics of the liquid propellants and also many advantages (it is our intention to omit to talk about liquid propellants and their sperimentation; solid propellants will be the subject of our attentions).

Forecast and control of combustion limits of condensed matter (ignition, extinction, oscillation, detonation, etc.) are now the main research subjects in this field. This type of information, necessary for all engines using chemical energy, is now lacking also at fundamental level. The main research fields are:
- theoretical combustion models for stationary conditions for traditional propellants with metals or catalysis and for new propellants;
- theoretical analysis and experiments of external controlled multiple ignitions and extinctions;
- experimental information on the flame structure in the plane near the combustion surface;
- experimental information on the particulate coming from the combustion surface;
- experimental information on the low pressure selfoscillating combustion.

But to obtain all the experimental information necessary to control and develop models, special equipment and special use of them is necessary. But not all the necessary information can be obtained or from models or from real rockets; from this point of view theoretical analysis can help experiments, showing the behaviour of the propellants under the action of different external forcing actions. Suitable experimental systems are:
- high power laser sources to obtain directly information on nonlinear combustion instabilities including ignition and extinction processes;
- laser techniques in particular L.D.V. (laser doppler velocimetry) to obtain directly information on the gas phase near the combustion region (velocity and size of the particulate)
- high speed schlieren and holographic films (from 10.000 to 50.000 pictures per second)
- combustion experiments at very low (<1 bar) or very high (>1000 bar) pressure.

These experiments take into account the tendency toward an increasing in the combustion chamber pressure in the future.

6. DIAGNOSTICS AND EXPERIMENTAL EQUIPMENTS

In the future optical techniques based on the coherent light will be used more and more in combustion and fluiddynamics. In fact the use of these techniques allows:
- no alteration of the phenomenon (non intrusive techniques);
- the use in bad environmental conditions where the use of sensors is very difficult (high temperature, corrosion problems, etc.);
- to obtain at the same time more than one information (for example, concentration size and velocity of the particules in the flow);
- to have high spatial and temporal resolution;
- to analyze fluctuating phenomena and to obtain distribution functions.

(1) see combustion not air-breathing

In general, these techniques require highly sophisticated electronic and optical equipment, are then very expensive and,moreover, signal analysis is in many cases rather difficult requiring a large amount of theoretical and numerical work.

A very important problem, involving the configuration of the research facility, is represented by the laser light source to be used. In same cases a very high power source is required. A solution can be the centralization of light sources and the distribution of light beams by means of suitable optical paths, as optical fibers, mirrors, etc., as shown in Fig. 6.1.

All the optical techniques here considered are based on the elastic and anelestic scattering of the radiation; the most important parameter to be considered is the cross section for the particular type of scattering considered, because from it depends the strength of the collected signal, and then it is possible to deduce the type of problems in the analysis and treatment of the signal. In fact, the lower is the cross section the lesser is the scattered signal (the amount of scattered radiation) and then the more complex and sophisticated the signal treatment system must be. Table 6.1 shows the cross sections of some scattering phenomena used in the diagnostics.

Many orders of magnitude divide Mie scattering from vibrational Raman,and from this it can be deduced the different type of problems to be solved in the use of the different processes. While in the case of Mie scattering particules must be present in the flow to have the interaction, in the case of the other scattering phenomena diffusion is made by molecules, seeding is not necessary, and information connected with the molecular properties (concentration, type of molecules, temperature etc.) can be obtained.

6.1 Fluiddynamics measurements

Parameters to be determined for the analysis of non-reacting flow fields are:
- turbulent fluctuations;
- spatial and temporal correlations between fluctuating components.
- the average velocity (or the components of average velocity)

The most suitable system for this type of analysis seems to be the laser doppler anemometer, able to give an instantaneous and simultaneous measurement of the different velocity components. The interaction is based on the Mie scattering and then seeding is necessary. Particle size and concentration should be carefully controlled to obtain reliable information. Fig. 6.2 shows an experimental set up to be used for the flow field analysis.

In spite of the complexity, these systems can be arranged in such a way that it is possible to use them also in large industrial plants without problems of vibrations etc. Velocity measurements are obtained from a frequency and then the limits of the system are connected with the available electronic technology; at the present time using argon-ion laser of some watts power velocities of the order of Mach 3÷4 can be measured. Spatial velocity correlations can be obtained by focusing in different points two different L.D.V. systems; data acquisition and treatment equipments must be used for the practical utilization of such kind of information. Today L.D.V. systems can be easily connected with microprocessors and microcomputers; in this way it is possible to obtain highly automatized systems able to give in a very short time any kind of data treatment and elaboration. Fig. 6.3 shows the interconnection between an L.D.V. and a data acquisition system.

6.2 Measurements in combustion

Optical techniques offer in this case the best opportunities. They are not intrusive and then able to be employed in such a bad environment. Optical techniques are able to give a lot of very useful information about combustion, also if many of them are still in development and probably can be largely improved. In general they are able to determine two fundamental parameters in reacting flow that is temperature and local species concentration. The experimental techniques to be considered are:
- atomic and molecular fluorescence;
- Rayleigh scattering;
- Raman scattering-CARS (Coherent Antistokes Raman Spectroscopy).

Fluorescence can be used for temperature and species measurements; in the case of atomic fluorescence seeding with suitable atoms is necessary. Also exhaust unburned hydrocarbons can be detected with fluorescence. Problems can arise because of quenching, autoabsorption and background radiation.

Rayleigh scattering can be used in density measurements taking into account the fact that scattered radiation is proportional to the number of molecules; this technique can also be used for soot analysis and, in connection with laser anemometry, to determine correlation functions of the type $\rho'u'$. Because of its very small cross section ($10^{-28}cm^2$/str)it can be used only in very clean flames.

Raman scattering can be used in different ways, also according to the experimental situation.It can be used to measure directly velocity and concentration while, in connection with a laser anemometry can furnish velocity-temperature and velocity-concentration correlations. Raman is a molecular scattering and no seeding is necessary; temperature measurements are now mainly obtained from N_2 or OH radical.

Because of the very small cross section ($10^{-30}cm^2$/str) it is necessary to increase the signal, and this may be obtained by means of CARS, in which the emitted radiation is coherent and then not isotropic but undirectional and much more strong. The signal to noise ratio is high in CARS and a large development of this technique in the future can

be foreseen;as it is possible to see in Fig. 6.4 local temperature and concentration can be measured with a single pulse of 10 n.sec duration. Pulsed laser have a repetition interval of the order of 10^{-1}sec. and continuous measurements can not be obtained.

As it is possible to see, the limit is not due to the diagnostic systems but to the background instumentation. It is possible to foresee that laser development will lead in a near future to pulsed lasers with repetition time of the order of $10^{-2} \div 10^{-3}$ sec.

7. LABORATORIES FOR FUTURE STUDIES AND RESEARCH

Looking at the future, taking into account the delay between the project and the industrial production, it is possible to foresee studies and researches on scientific and technical problems to improve the aerospace defense.

7.1 Solid propellant rocket

To develop always greater thrusts is the trend in solid propellant rockets in order to increase the rocket speed and to reduce the arrival time to right altitude for the employment in the range of action of the rocket. This thrust increase can be obtained increasing the combustion chamber pressure and it is expected to reach pressures about 800-1200 bar; this objective implies a research, nearly a base research, about combustion - also with mathematical models - in order to single out the combustion stability limits. Stability is necessary in order to avoid extinction and detonation. Hence it is necessary to acquire a serie of experimental data, for steady and unsteady phenomena, changing temperature and pressure. Temperature must be varied between a few tens of centigrade degrees under zero and a few tens of centrigrade degrees over zero; pressure between limits below atmospheric pressure and those of the order of 1000-1200 bar; that is, it is necessary to determine the static and dynamic stability limits in the heterogeneous combustion.

Dynamic limits, being not a property of the state of a system, can be observed only indirectly; all the experimental techniques, already known, must be matched together in order to observe simultaneously: ignition, combustion rate, pressure, temperature, brightness and doppler signal. It was observed, and analytically proved, that the dynamic stability lower limit is instantaneously valid for monotonically decreasing forcing in time, asymptotically for arbitrary in shape, but growing fainter in time, forcings; in particular pressure impulses and radiation impulses were considered as external forcings. The stability limit is the same for pressure and radiation disturbances, if the condensed phase is optically opaque, as solid propellants often are.

In order to observe the combustion dynamic stability, the implemented experimental technique argued is described. A steady combustion sustained by laser radiation at pressure of interest is considered; dynamic stability is directly bound to the radiant flux intensity and to the fall speed of radiant impulse required in order to extinguish combustion; hence, a radiant flux variation produces the same effect as pressure variation; advantage in experiments is well imaginable.

A laser radiation, appropriately selected, in order to produce a negligible photochemical effect and well controllable, is the ideal external forcing in determining stability limits - static and dynamic - and of acoustic admittance of solid fuels; acoustic admittance driven by radiation derives from its connection with acoustic admittance driven by pressure. Unprofitableness in using pressure like external forcing in this kind of tests is due to following difficulties:
- strong perturbations induced of fluidynamic and kinetic character;
- difficulties in the forcing control.

In Fig. 7.1 the experimental apparatus for tests of combustion driven by monochromatic and appropriately modulated light, at variable pressure from vacuum to tens of atmospheres and with a variable period from millisecond to tens of second.

The instrumentation chain for the required diagnostics must be connected to the experimental bench and with anemometers, laser, high speed cinematography, oscilloscope, image visualization, power meter, etc.

7.2 Interception systems

The detection of shock waves originating from the interaction of fast bodies with the ionosphere and, generally speaking, the transmission of signals through ionized gases (e.g. from and towards aircrafts and spacecrafts at super and/or hypersonic velocities) plays a great role among all the defense problems.

Plasma physics already possesses most of the theoretical and experimental methodologies suitable for studying the ionized gases; at present, such methodologies are widely used in controlled thermonuclear fusion experiments: a very important research field connected with the future availability of energy (e.g. Tokamak and Stellarator machines, Plasma Focus apparatuses, laser fusion systems, etc.).

It is then natural to acquire such methodologies and to make them to tend towards this wide-scope and up-to-date topic; the research on ionized gases has a great operational flexibility being able to be everytime focused on the specific emergent problems. Moreover, the "broad range" professionalism gained by the engaged people guarantiees the continuity at the operational level for the staff personnel.

Aside from the above mentioned very important defense problems, it is then useful to

list some of the fields, most of which possess a technological nature, which have problems very similar to those found in the research on ionized gases:
a) telecommunications; b) shock wave dynamics and explosion phenomena; c) physics of the ionosphere; d) aerospace problems; e) ion and electron beams; f) plasma lasers; g) surface conditioning; h) controlled thermonuclear fusion; i) energy conversion; l) ionic propulsion; m) solid state plasmas; n) combustion and chemical plasmas. In spite of the differences existing among the practical applications of the above mentioned topics, the related research problems need similar theoretical and experimental methods and, often, very similar diagnostic apparatuses.

Here, we propose a "Diffusion Plasma Machine" (DPM) for producing a quiescent and homogeneous plasma. It has two specific characteristics: simplicity and a great operational flexibility. These two requisites have to be considered as essential for a lasting experimental plant, i.e. for a research installation which must satisfy further experimental requirements connected with a today-unknown future.

Fig. 7.2 schematically shows the DPM apparatus for producing the basic quiescent plasma ("environment plasma") (for plasma production see also Appendix II). This plant is a unit-composed system with three structural modules: the central one is connected to the vacuum system. The filling gas is injected through valves. The three modules have basically the same structure: a low permeability metallic jacket encloses a cylindrical grid which is heated up to the thermoionic emission temperature; an accelerating potential is applied between the two structures and the resulting plasma diffuses towards the center of the machine due to particle diffusion processes. The three units can be operated both simultaneously and separately; the electrical insulation between them allows a differential biasing when required. Moreover, the two metallic grids a and a' separate the three moduls for providing the requested boundary values for the plasma voltage between two adjacent sections.

The external jacket and the flanges must be water cooled. For each unit, the electrical insulation between the cylindrical grid and the external jacket is obtained by means of two ceramic rings connected by metallic tension rods supporting the filament winding which constitutes the electrically heated cylindrical grid. A set of flanges in the external jacket provides the required accessibility to the machine for diagnostics purposes, allowing the introduction of probes, electrical devices and the set-up of optics windows.

The proposed machine (DPM) allows the creation and containment of a "quiescent" plasma with an electron density of the order 10^{8} - 10^{10} cm^{-3} and electron temperature of the order 0.5 - 5 eV. More tenuous plasmas with parameters even closer to the ionospheric ones, are obtainable by means of various plasma sources: for this reason the flanges P' must have a diameter broad enough for allowing the connection of these sources to the machine, thus greatly improving the plant flexibility.

The low magnetic permeability of the external jacket allows the use of Helmholtz coils or of permanent magnets for providing a weak magnetic field if required by the experimental conditions. Obviously, the magnetic field configuration must be carefully planned to avoid unwanted effects in the plasma formation and diffusion processes.

Fig. 7.3 shows the cylindrical grid in the inner region of the (external) jacket. The filament material is tantalum (or other similar ones), the filament diameter can be estimated of the order 0.1 mm. The insulation ceramic rings must tolerate a continuous operation at a temperature of the order 2.000°C in their outer hottest region (boron fluoride). The mean distance of the cylindrical grid from the inner surface of the jacket can be estimated of the order 10 mm (tipically: 20 mm).

The flanges perpendicular to the cylindrical axis of the machine prevent some regions of the inner surface from being effective in the plasma formation process: they represent spatial discontinuities affecting the desired homogeneity of the plasma. To lower this effect, the ratio Φ_M/Φ_i cannot be lower than 6-7, where Φ_M and Φ_i are the machine and the flange diameters, respectively (see Fig. 7.3).

In conclusion, it is useful to point some requirements for the related instrumental apparatuses and to list some of the required diagnostic systems: a laser with power of 50 MW (optimum: 20J in 20 nsec, i.e. 1 GW), microwave interferometry, spectroscopy in the visible range, Langmuir probes, r.f. injection systems and spectrum analysis, resonance probes, Faraday's cups, etc.

8. DATA ACQUISITION SYSTEM

In order to design a laboratory for experimentations as close as possible to real conditions but of short duration one has to develop data acquisition systems capable of recording, during the test, all the data necessary to understand the investigated phenomena. In short time duration the maximal amount of data has to be collected i.e. one has to utilize very high rate data acquisition systems. The optimal system configuration of the data acquisition system is depending upon the type of measurements to be operated and chiefly on the sampling frequency needed and on the test duration (data volume). These parameters determine the choise of data acquisition subsystems. Fig. 8.1 shows the actual and projected limits in the next 5 years for the main subsystems dealt with the following paragraphs.

8.1 Converters

The conversion speed for A/D devices is the most significant parameter able to contain the cost of the whole system. In fact, for a fixed number of channels, a higher conversion speed allows more signal channels multiplexed on the same converter and less data channels out of the converter thus simplifying the architecture of the system.Today A/D conversion modules can be found in the market with adequate resolution power at prices ranging in the low thousands USD capable of converting at 50-100 kHz. In the near future and at higher prices conversion systems with frequencies higher than 100 MHz will be available. Costs rise anyway since, at such a high frequencies, subsystems for data storage are needed. For 100 ÷ 200 MHz A/D conversion systems in the next few years prices will range in the low tenths of thousands USD.

8.2 Memories

Data sampled have to be stored in memories from which will be retrieved for computing purposes; the dimensions and speed of the memories are critical parameters for dimensioning the whole system. Mass memories chiefly disks make no problem for the storage of high volumes of data but speed is severely limited, transfer speeds of about 50 kHz are the actual limit. For higher frequencies buffer memories are needed to accomodate temporarly the data; by this way conversion rate can be higher but dimension limits are severe and the volume of data transferable is low. Disks and removable disks chiefly are a very handy way of storing data.

8.3 Control systems

A control system masters the operation of the conversion channels and the storing of data collected during the test; this can be accomplished at different levels. At low speeds for the incoming data a microprocessor with analog interfacing devices can be utilized on real time.

For faster speeds the control has to be distributed in sophisticated peripheral subsystems whose task is the acquisition and storage of data. A central computer with limited computational power, acts as a supervisor and manages the transfer of data, once the test has be completed, from the buffer memories to the mass memories as shown in Fig.8.2.

Today available microprocessors once interfaced with A/D converters are capable of coping with frequencies in the order of 500 kHz. In the near future the HMOS technology will push this limit upwards in the range of 5 MHz.

For still higher speeds dedicated control systems are needed which are constituted by high speed special components by which speeds up to 100 MHz are obtainable as shown in Fig. 8.3.

As for the programming language implemented to ease the operation of software configuration of the control system, interpreters to be easily used by not highly specialized operators are forecastable.

8.4 Electronic laboratory

The optimal configuration for each data acquisition system is depending upon the test requisites and the characteristics of the available subsystems. Some considerations on this point have to be evidentiated; sampling frequency, number of channels, test duration; since these parameters are depending upon the specific test to be carried out the data acquisition system configuration has to be found case by case.

8.5 Considerations on data acquisition systems

Operational limits dealt with the preceeding paragraphs are not coincident with the limits of the devices available in the market; in fact because of economic reason systems on the edge of the technological capabilities are not generally worth to be manufactured.

The ratio between the performance of the systems availables in the market and the inherent potential limits of the components can be grossly estimated to be around 10 and this ratio is probably going to stay in the future. This considerations entails that a research laboratory has to have an electronic department whose tasks are:
- to configurate the data acquisition systems for the specific test by means of utilizing available subsystems and modules;
- to design and realize high performance systems utilizing components at the highest technological limits in order to fill the gap between the potentially realisable systems and the ones available in the market.

8.6 Sensors and advanced diagnostic techniques

These instrumentations and components needed in each test are widely different depending upon the type and modality of the test and therefore general projection are not easily assessable.

Continuosly rising limits in pressure temperature and speed of gases and transient phenomena response impose highly demanding requisites to the sensors and measurements chains along with the more conventional necessities of accuracy and precision. Sensor poisoning by means of the characteristics of the (reacting) gases has to be mentioned.

Progress in the last years has been slow and perhaps new techniques and technologies promise further improvements. This problem area is certainly one of the most critical and demanding and one that shows inadequacies in front of the theoretical and experimental needs. Several manufactures are involved in the field whose number is rising in front of the widening use of microprocessor based control systems which necessitates progress in sensors and acquisition devices. The energetic and aerospace fields are strongly oriented toward more utilization of these systems; solutions for extreme experimental conditions are therefore the leading realizations from which industrial components benefit.

9. CONCLUSIVE REMARKS

Looking at the development of aerospatial propulsion in the near and medium term future in the whole, it can be observed that complex experimentation procedure shall be needed in all the areas of systems and components for propulsive devices in order to develop more and more significant mathematical models to simulate the physical phenomena and therefore to forecast the propulsion system behaviour.

Long term tests, demanding large energy consumption and personnel worktime are less preferable than short burst of energy at high power levels - from 200 10^3 to 10^6 kW - in short durations (10 - 1000s). The laboratory has to be fitted with optical and electronic highly sophisticated instrumentation (while continuously updating in front of new diagnostical techniques) and with data acquisition systems capable of allowing, in very short time durations, to collect and compute data by means of distributor microprocessor power and central processors of high power in order to analyze thoroughly the physical phenomena both in fluid and in machine.

Reliability analysis of single components and of the whole propulsion system may well be performed by means of an accurate sampling of experimental data and by application of analitical and mathematical methods which have been today developed and used for nuclear power plants and relative components.

In case of a strict necessity of carrying out examinations for a long time at speeds up to Mach number 2 ÷ 2.8 aircrafts yet existing can be properly fitted along with experimental flying laboratories like DC-10 and Concorde. Utilization of the abovementioned aircrafts or similar as experimentation centers copes with actual and future economical considerations in the same way as the spatial shuttle.

Having examined research laboratories for aerospace propulsion typical of the past I think to have envisaged with adequate certainty a research laboratory apt for the future, that is yet in the present, in the way synthetically expounded in the present relation.

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- Appendix

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APPENDIX 1

In order to operate a blow-down one has to store large quantities of air at high pressure in large air cylinders whose volume capacity ranks in the 1000 ÷ 2000 liters range and which are capable of sustaining very high pressures 200 ÷ 300 bar (the maximum allowable pressure has to be about 10% higher) and to control by means of a series of pressure regulators the desired pressure and flow to carry out the test.

In this situation cost containement can be looked after by means of volumetric compressors of limited flow value: the delivery pressure has to overcome 300 ÷ 400 bar and the richarge time of course varies depending upon the compressor flow rate and the volume to be filled.

Air compressors can be driven by electric motors or reciprocating diesel engines and compress the air with mono or bicylindric configuration; whenever possible, natural gas has to be chosen as a fuel.

Fig. III-1 shows a detailed scheme of the storage plant with air cylinders fitted for the wind test beds of the CNPM laboratory of the Politechnic of Milan to carry out tests on compressor and turbine cascades as dealt with in Appendix III.

When air inlets and nozzles have to be investigated (by means of air) as components of propulsion systems for supersonic or hypersonic speeds (Mach n. 4-12) in order to substantially rise the air flow speed in the wind bet it is possible to operate the discharge in a vacuum sphere of high capacity, in which depression by means of a proper vacuum pumping set can be as low as 1-0.5 mm Hg.

APPENDIX II

High power values, for reasonably long times, is an uneasy task and the power values to be installed in order to operate subsonic and transonic wind tunnels with large air flows (very large sections) and to operate supersonic wind beds with lower air flows are as high as to necessitate electric motors highly costly, unreliable and difficult to find.

The characteristics of actual electric motors with continuous current excitation for present and future realizations are shown in the plot of Fig. II-1. These characteristics imply limitations in the use of electric motors to actuate compressors in order to drive wind tunnels and therefore blow-down techniques have to be utilized.

In order to make a blow-down the energy storage system may well be mechanical-based on kinetic energy accumulation in large flywheels (this technique has been yet adopted at the CERN laboratories in Geneva for the synchrothrone and at the Max-Plank Institut für Plasma Physik to generate plasma). This type of energy storage is certainly more flexible than the storage of compressed air in large cylinders; in fact it can be used also to drive electric motors, compressors, pumps etc.

Kinetic energy accumulation can be accomplished in a single flywheel for low energy values or in cylinders constituted by the point of several flywheels. Actual technologies lead to different limits depending upon the choise:

- Germany 120-150 t ;
- USA 180-210 t ;
- Japan 220-250 t .

As a fact in order to obtain a higher rotor mass and consequently a higher energy storage the cylinder can be configurated with significant length values which vary from 3 m up to 7 m. Powers to drive the cylinder up to the maximum speed are not very high and anyway lower if a vacuum chamber is fitted around the flywheel to lessen the air friction losses.

An electrical motor can be coupled to the cylinder but the operation time duration of the electric motor is strongly limited by overheating since cooling techniques are uneasy. Fig. II-1 shows the coupling characteristics. Operating time durations vary from 20 s to a minimum of 3.

Coupling by means of an overgear can be easily accomplished as well as driving a pump. These facts demonstrate the inherent versatility of the devise and the diversificated use of this basic energiser in a laboratory for aerospace propulsion and for all the other uses dealing with energetic problems.

Figs. II-2, II-3, II-4 show plots of the thermal power (W), compressed water flow (kg/s) and compressed air flow (kg/s) versus time duration (s). Fig. II-3 and II-4 show different curves as a function of pressure. Curves have been computed for overall system efficiencies of 0.5 and 0.75. The calculus has been carried out basing upon the following assumptions (data have been drown down from the Max-Plank Institut für Plasma Physik)
Cylinder:
length 5.700 m
outer diameter 2.9 m
inner diameter 0.4 m
flywheel weight 223 t
whole weight 323 t
peripheral speed 250 m/s
rotational speed 1650 rpm

Other characteristics are:
driving engine power 5700 kW
maximum available energy 1450 MWs

maximal electric generator power 167 MVA
maximal AC output voltage 2x3,3 kW
maximal AC output 2x22,5 kA

pulse duration 11 s
repetition interval 6 min
time to reach nominal rpm 25 min
time to break down 18 min
time to slow down (without breaking) 3 h

A further consideration is that in order to install electric motors of very high power, omitting cost and realisability problems, an electric power line is needed devoted to the job. Otherwise a black out would be determined in the area served by the power station during the operation. The peak power value is well higher than that of an ordinary nuclear power plant.

APPENDIX III

Inside the limitations of the structures and of the financial possibilities, the Polytechnic of Milan (CNPM) has followed the two criteria above mentioned for its turbomachinery experimentations.

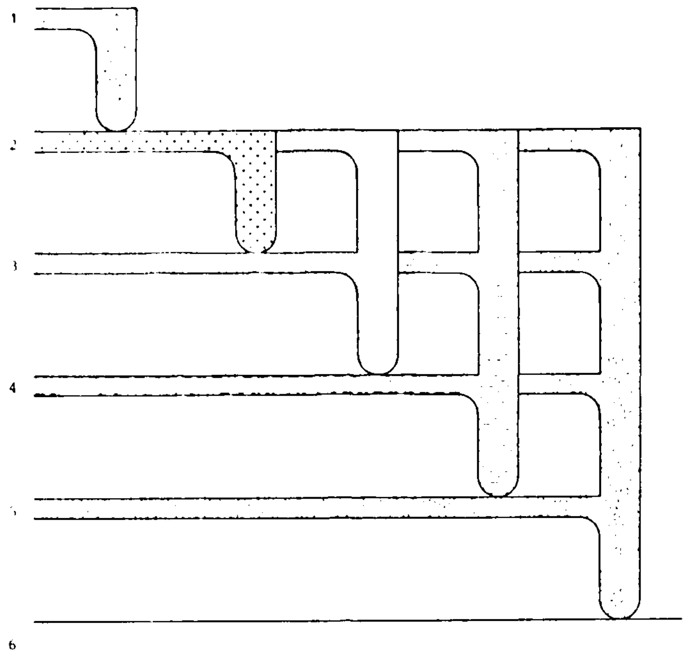
The laboratory has a high pressure (130 bar) air stockage facility at the moment; but a new bigger and higher pressure (200 bar) equipment is now being built which will be able to provide supersonic flows for short periods (from 5 sec to 10 min). The scheme of this installation is represented in Fig. III-1.The laboratory is now developing a closed cycle circuit useful for experimentation of subsonic, transonic, supersonic compressors and turbines working with air or suitable fluid mixtures which allow to reach high Mach numbers.

The technical features of the installation are the following: on the test rig compressors or turbines can be tested with power recovery with the aim of a coupling of a generator/motor in d.c. till 400 kW. Such a system allows to test 800 kW compressors whose half power would be given by the electric net. Other main features are:

compressor max power	800 kW
turbine max power	400 kW
working fluid	air or other fluids (freon, etc.)
revolution speed	0 ÷ 35000 rpm
max rotor diameter	500 mm
max flow rate	15 kg/s for $\beta = 2$
	4 kg/s for $\beta = 10$

The scheme of the turbomachinery test rig and a siplified lay-out are shown in Fig.III-2 and Fig. III-3.

The disegn of the plant was fully carry out and the construction has already started at C.N.P.M. (Centro di Studio per Ricerche sulla Propulsione e sull'Energetica).



1 Free Research: Statement: to improve the knowledge; to amplify the science boundary.

2 Orientated Research: Statement: to explore and investigate the possible applications of the Free Research.

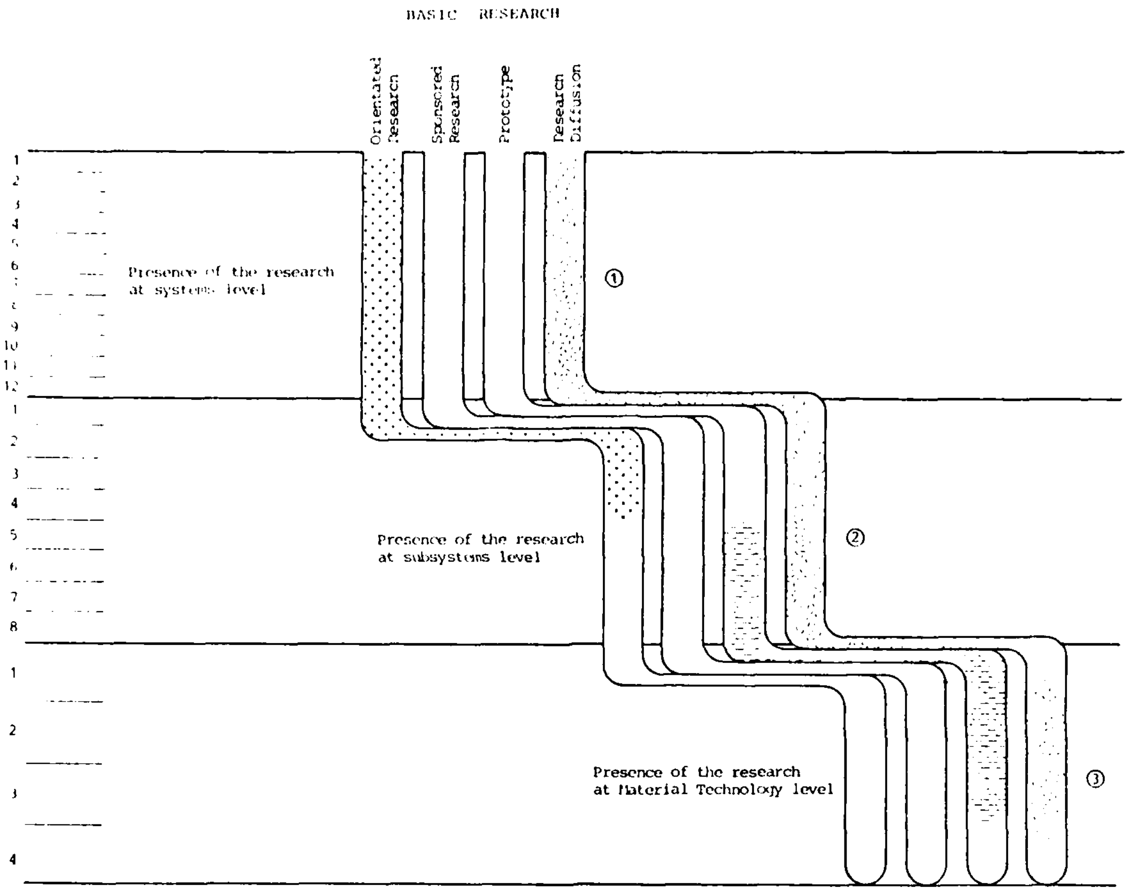
3 Sponsored Research: Statement: to anticipate and define the innovations on systems of country social and political interest.

4 Prototype Realization: Statement: Realization and testing of the whole system using the research results in their variety and complexity.

5 Research Diffusion: Statement: to transfer the studies, the researches and the realizations to other people.

6 Industrial Production: Statement: planning and realization of the system inside an industrial production and therefore inside the production management.

Figure 1.1



① Structural Design:
it is the transfer of use necessities and of scientific options into system engineering techniques, starting from a global inspection of the interactions between the technological inner system and the outer system inherent to the use, also considering all the possible structural changes inside the system.

② System Design:
it is the study of products based upon new technologies which must be developed through subsystems and components, starting from the product characteristics and from the use necessities.

③ Configurational Design:
it is the development of new products with the existing technologies, starting from the product characteristics.

Figure 1.2

- aircraft propulsion
  - subsonic n. Mach<1
    - turbomachinery
      - compressor
      - turbine
    - intake
    - nozzle
    - subsonic continuous combustion
  - transonic 0,9≤n.Mach≤1,05
    - turbomachinery
      - compressor
      - turbine
    - intake
    - nozzle
    - subsonic continuous combustion
  - supersonic 1,05≤n.Mach≤4
    - turbomachinery
      - compressor
      - turbine
    - supersonic intake
    - nozzle
    - continuous combustion
      - subsonic
      - supersonic
  - hypersonic n.Mach>4
    - hypersonic intake
    - nozzle
    - supersonic continuous combustion

Figure 3.1

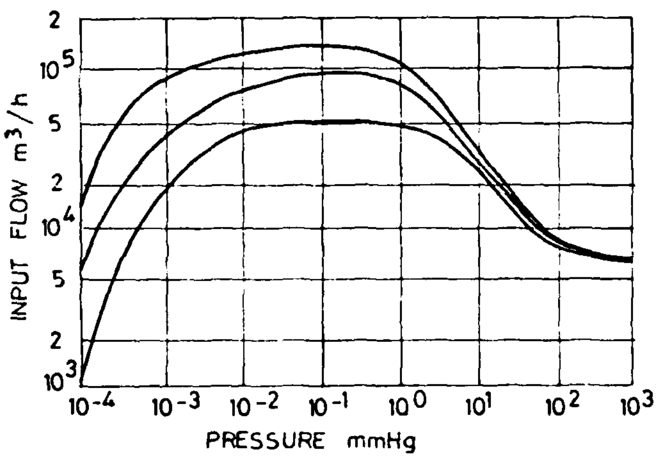


Figure 4.1

- combustion
  - air-breathing
    - subsonic
    - supersonic
  - no-air-breathing
    - liquid propellant rocket
    - liquid-solid propellant rocket
    - solid propellant rocket

Figure 5.1

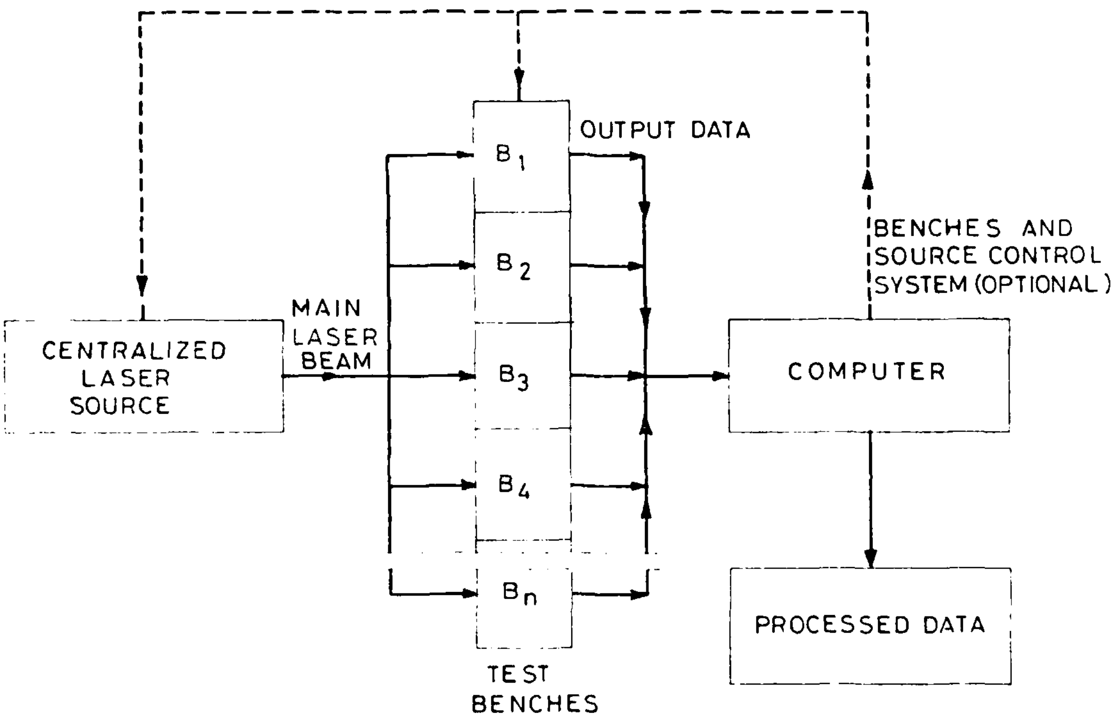


Fig.5.2 The proposed laboratory configuration with the centralized acquisition and data reduction system

Interacting particle	Interaction type	cross section cm^2
Particle Ø 1 μm	Mie scattering	10^{10}
Molecule	Absorption	$10^{17} \div 10^{20}$
Molecule	Fluorescence	$10^{18} \div 10^{21}$
Molecule	Rayleigh scattering	10^{28}
Molecule	Raman rotation	10^{30}
Molecule	Raman vibrational	10^{31}

Tab.6.1 Main characteristics for some optical diagnostics

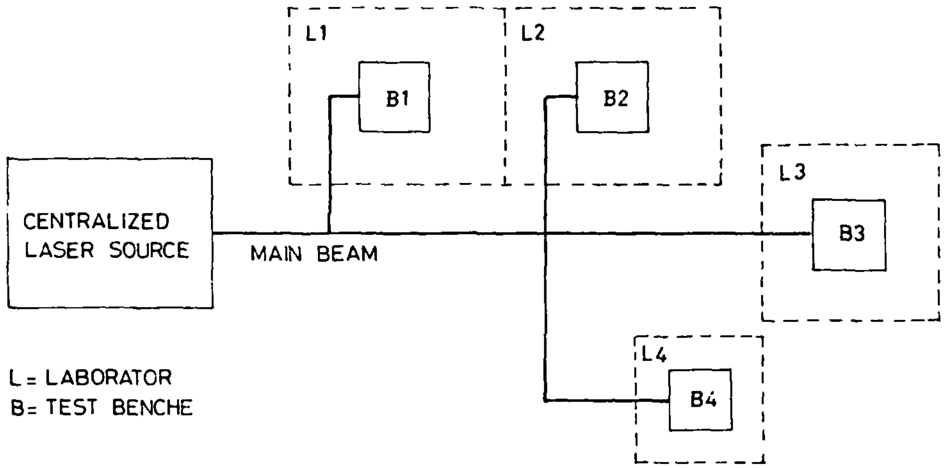


Fig.6.1 The research laboratory with the distribution system of a centralized high power laser beam

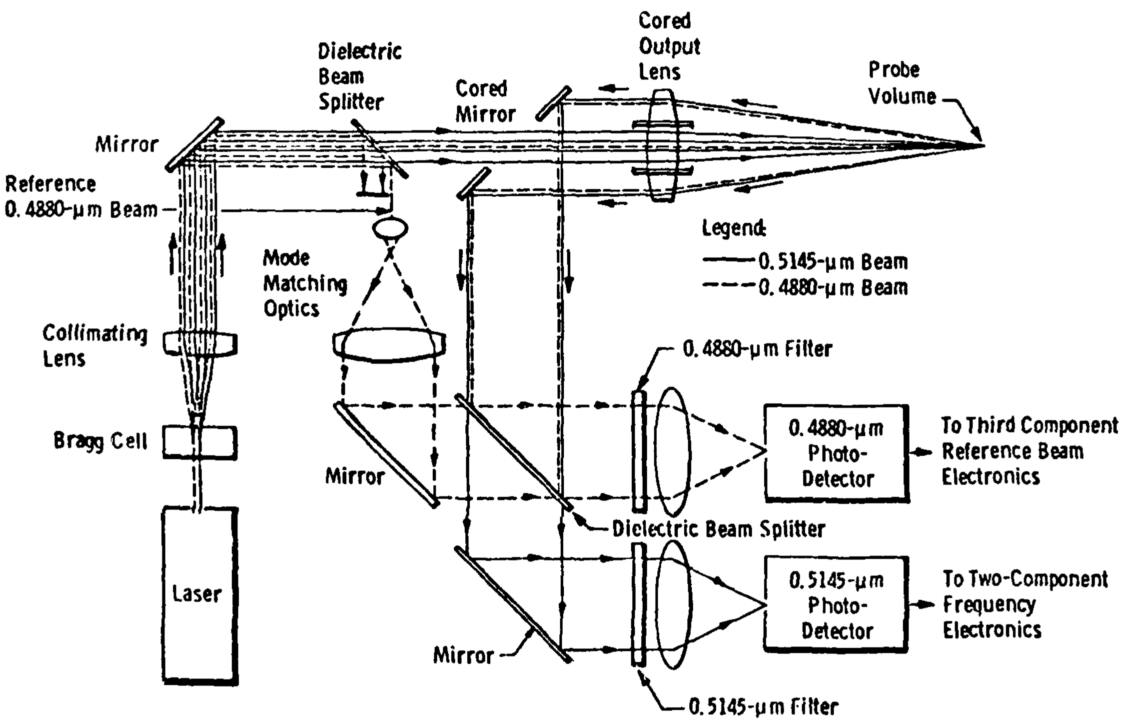


Fig.6.2 A typical three component velocimeter schematic

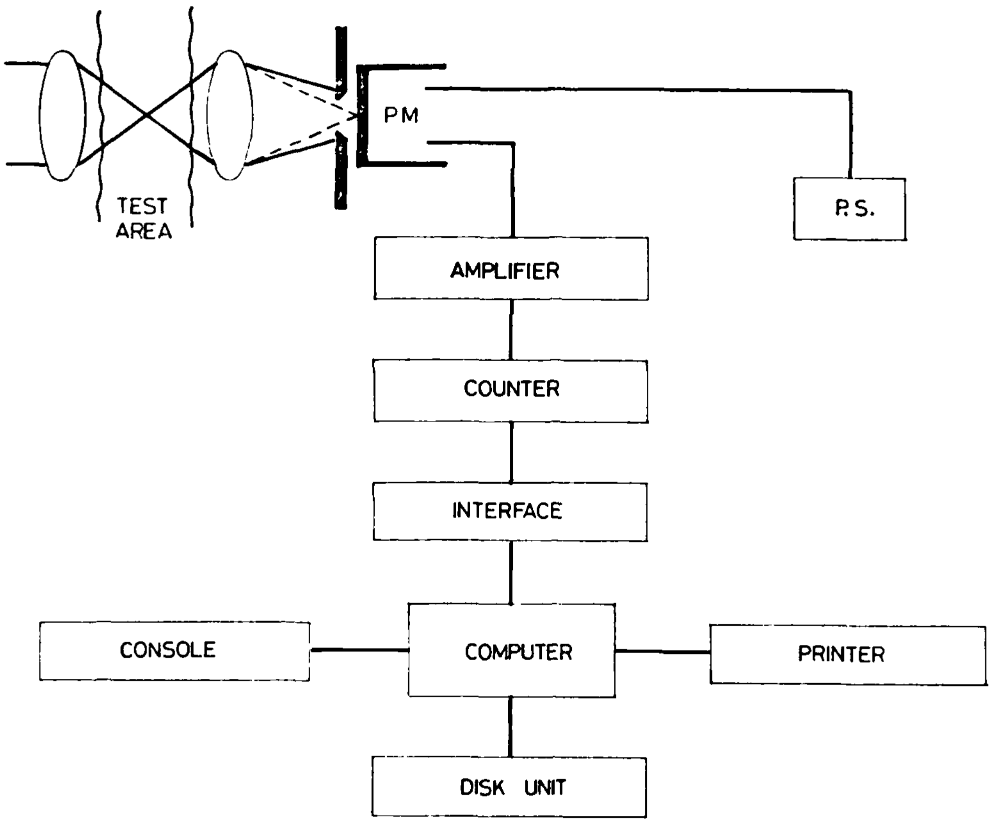


Fig.6.3 A schematic of an LDV system interfaced by a minicomputer

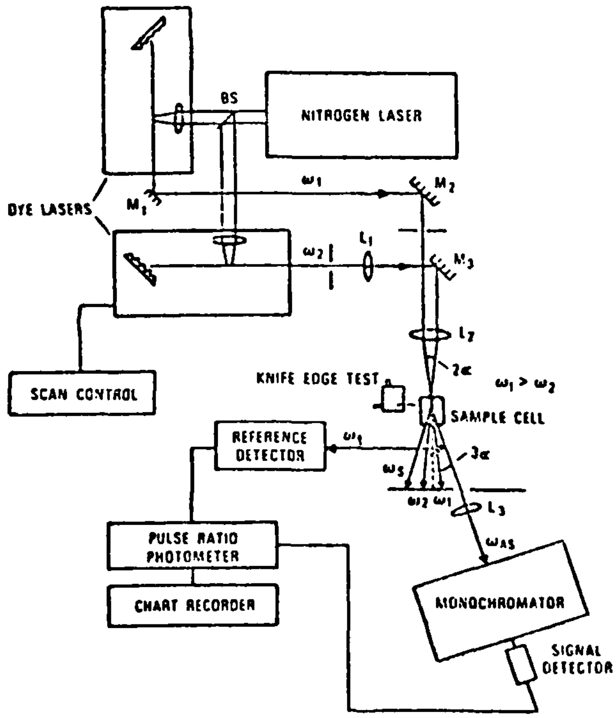
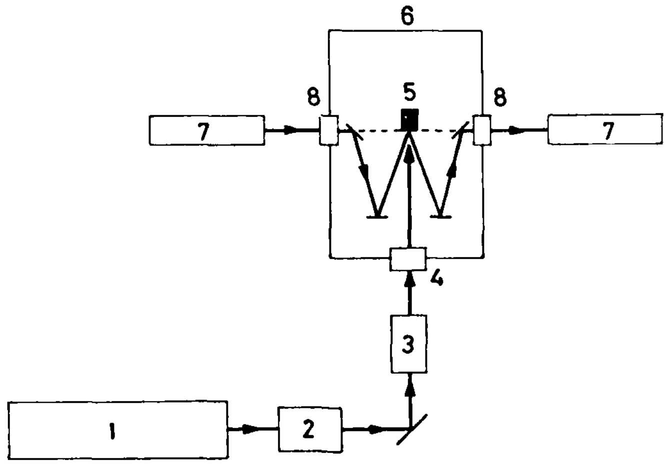


Fig.6.4 The CARS opto-electronic system



Experimental set-up

1. CO_2CW laser
2. Camera shutter
3. Variable attenuator/beam expander/Kaleidoscope system/optic integrator system
4. High pressure I.R. window
5. Solid propellant (target)
6. Combustion chamber
7. Diagnostic system
8. High pressure visible window

Fig.7.1 Test rig for combustion research with power laser

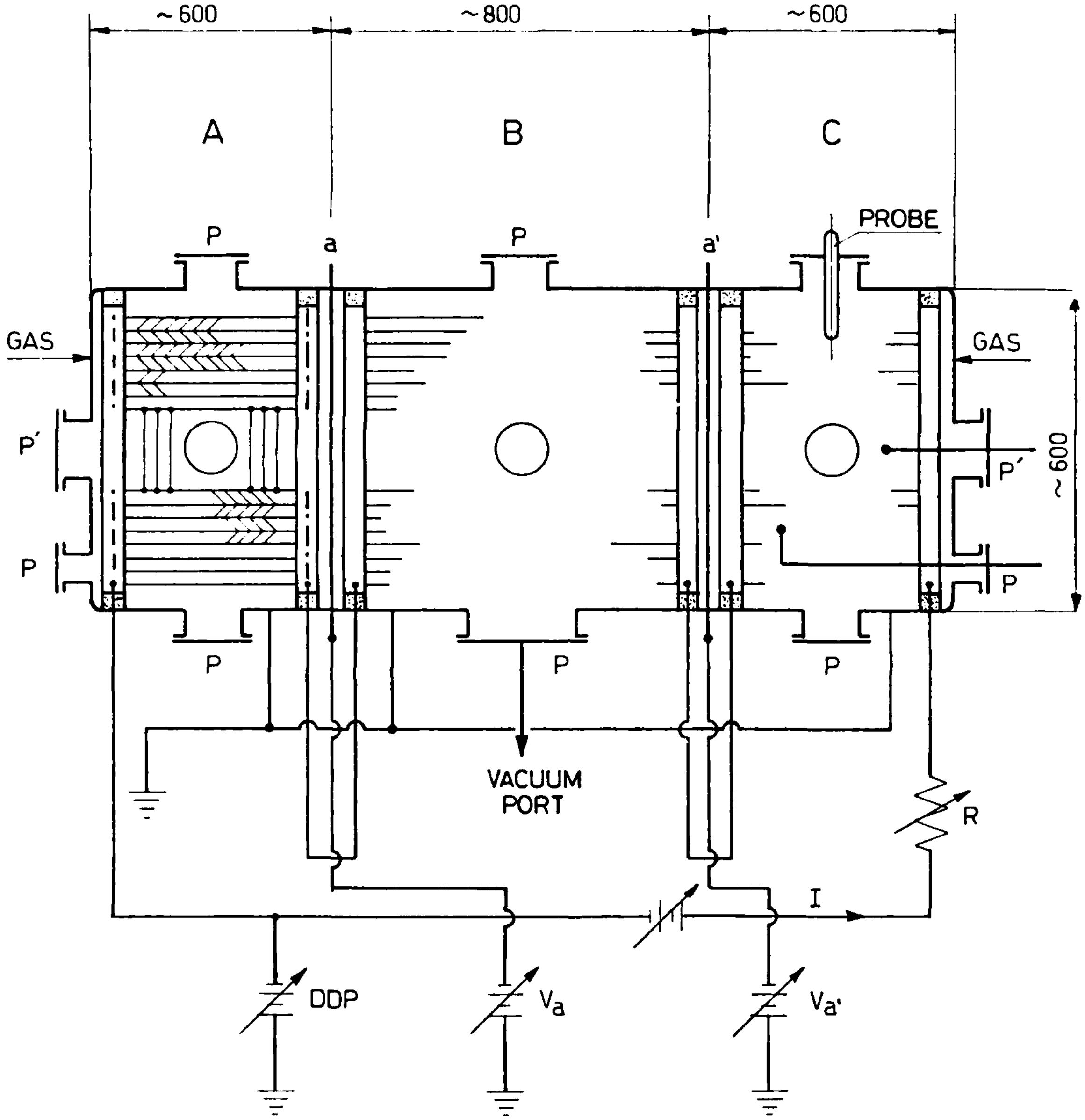

~600
~800
~600
A
B
C
PROBE
P
a
P
a'
GAS
GAS
P'
P'
P
P
~600
P
P
P
VACUUM
PORT
R
I
DDP
Va
Va'

Figure 7.2

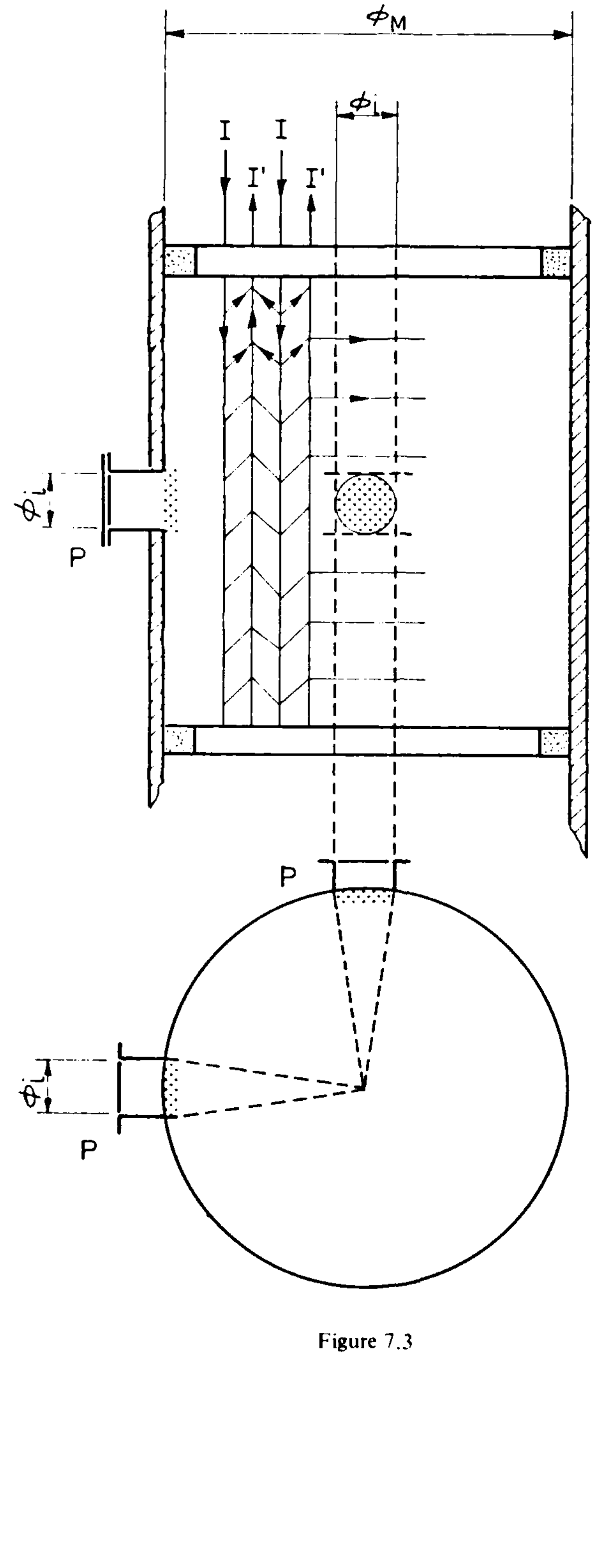


Figure 7.3

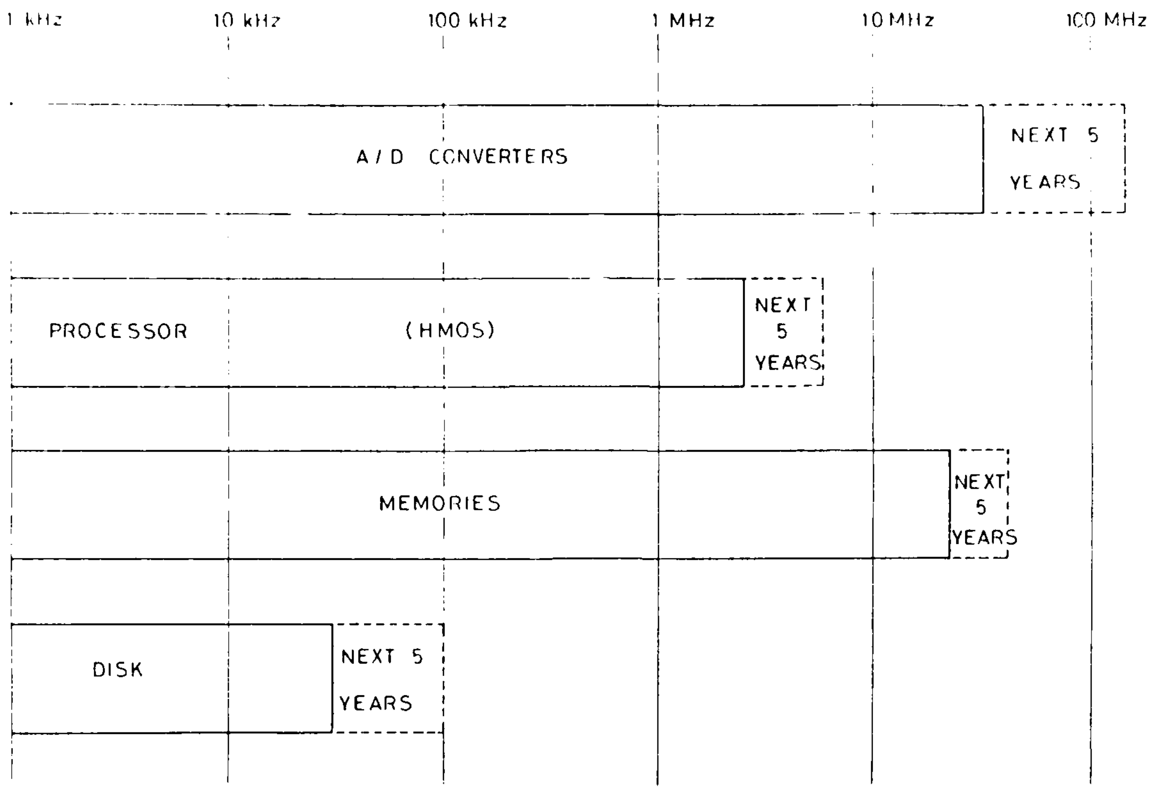
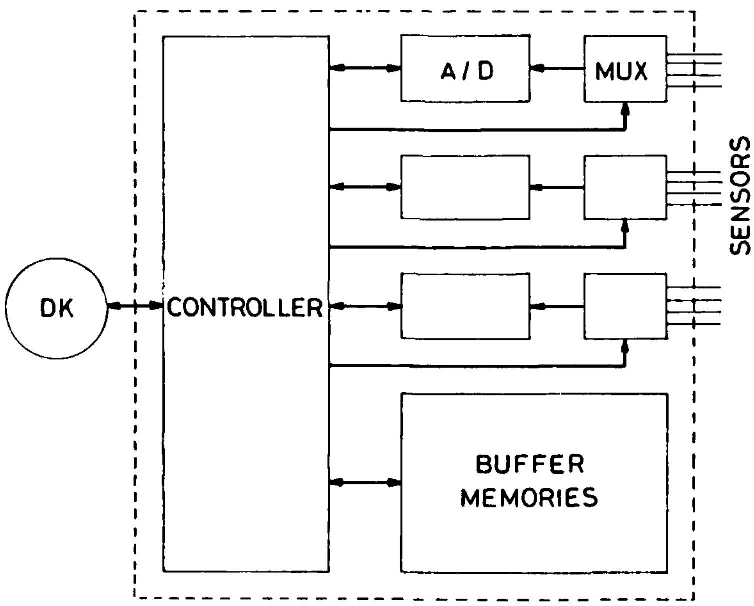
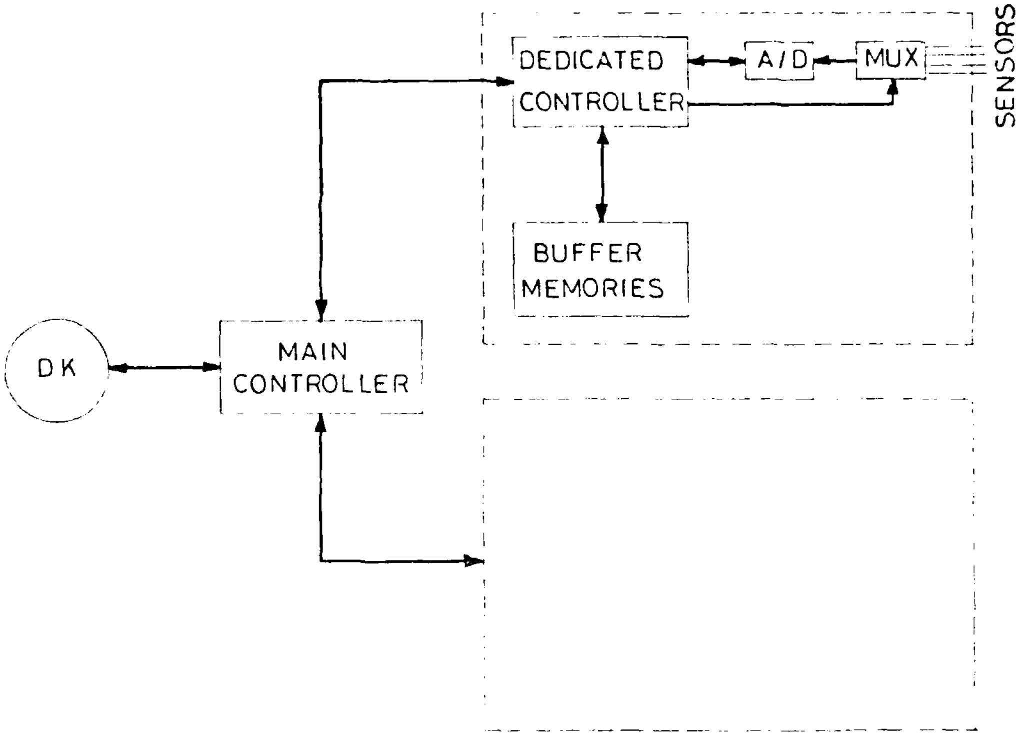


Fig.8.1 Actual and projected speed limits for data acquisition subsystems



A/D = analog to digital converter
MUX = multiplexer
DK = disk memory

Fig.8.2 System configuration for signal acquisition up to 500 kHz



A/D - analog to digital converter
MUX - multiplexer
DK - disk memory

Fig.8.3 System configuration for signal acquisition up to 100 MHz

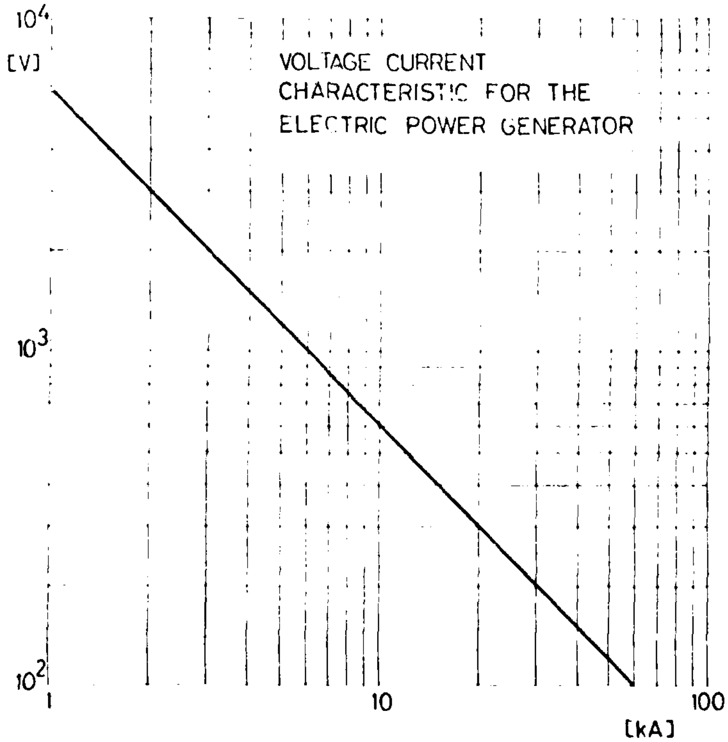


Figure II-1

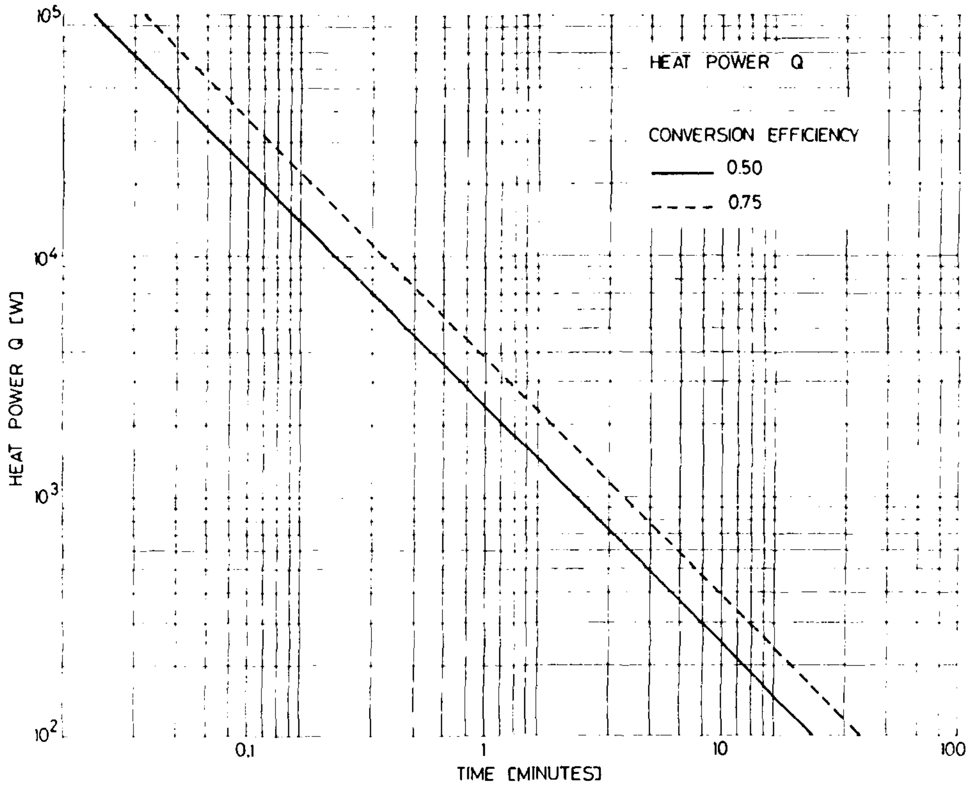


Figure II-2

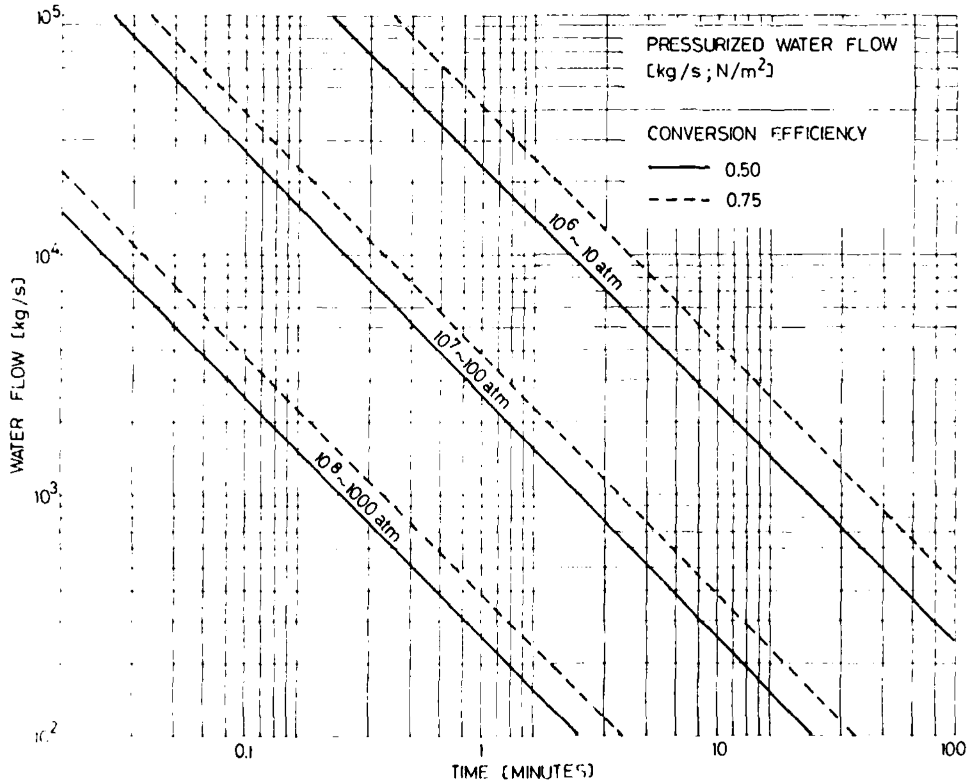


Figure II-3

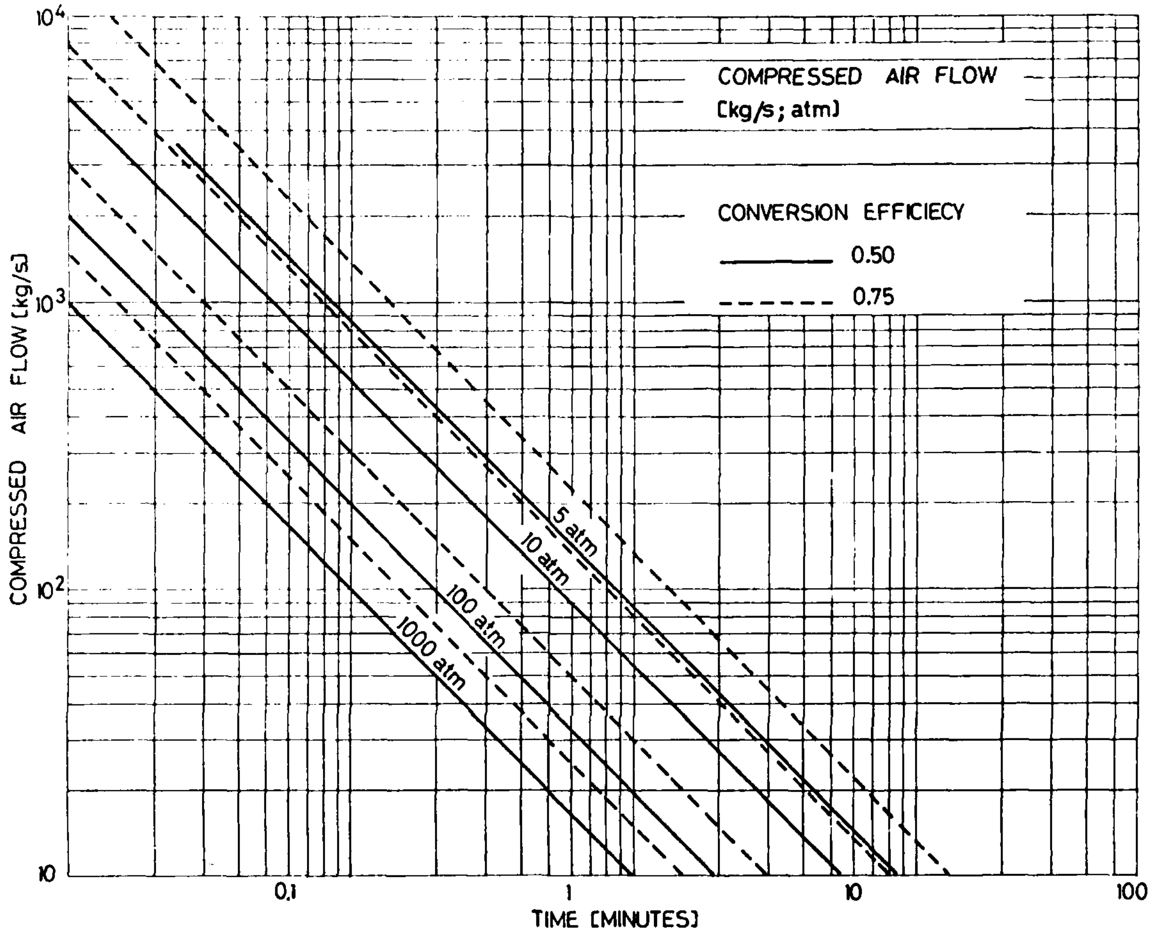

COMPRESSED AIR FLOW
[kg/s; atm]
CONVERSION EFFICIECY
0.50
0.75
5 atm
10 atm
100 atm
1000 atm
10^4
10^3
10^2
10
0.1
1
10
100
TIME [MINUTES]
COMPRESSED AIR FLOW [kg/s]

Figure II-4

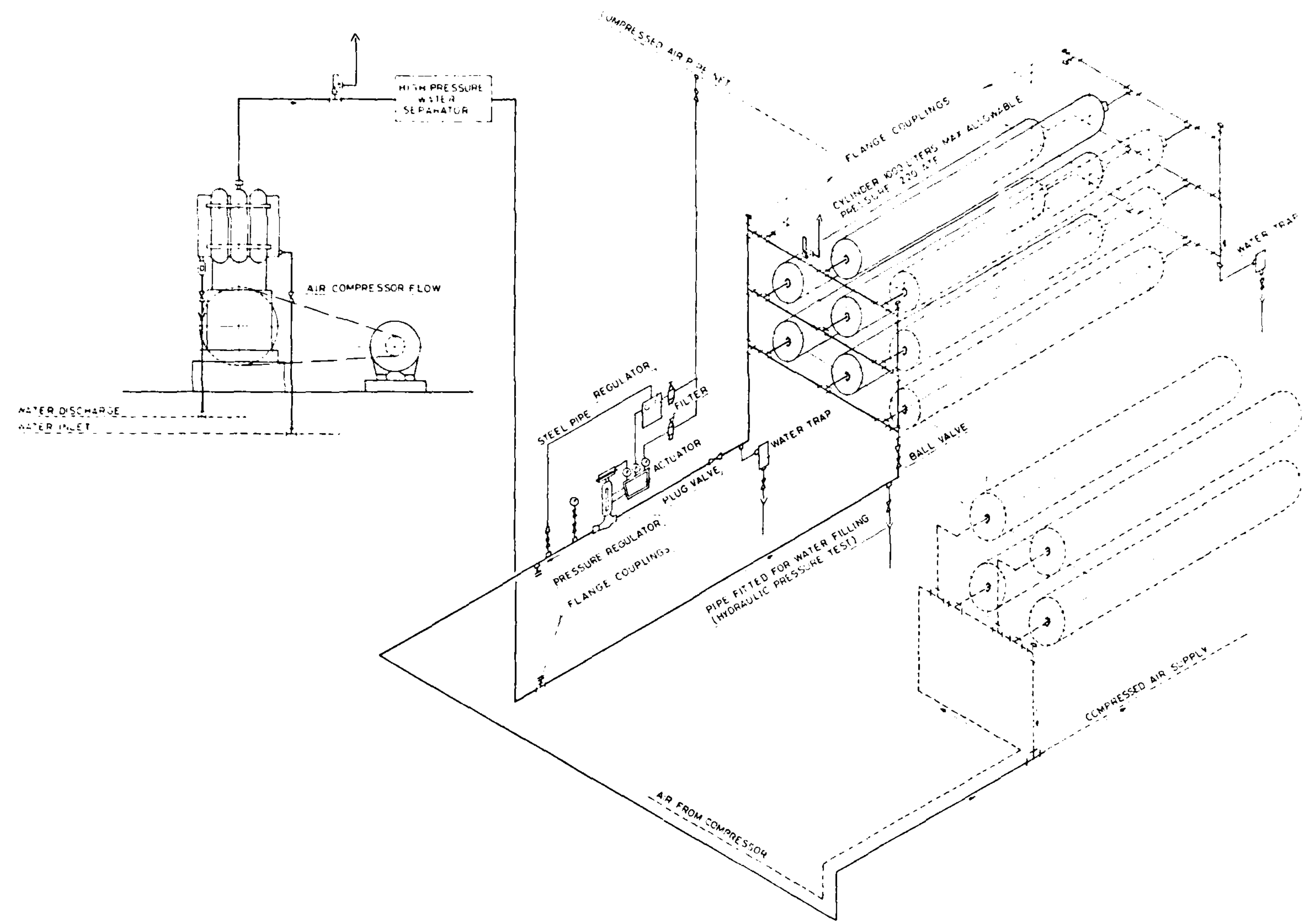

HIGH PRESSURE WATER SEPARATOR
AIR COMPRESSOR FLOW
WATER DISCHARGE
WATER INLET
COMPRESSED AIR PIPE SET
FLANGE COUPLINGS
STEEL PIPE
REGULATOR
FILTER
ACTUATOR
PLUG VALVE
WATER TRAP
BALL VALVE
PRESSURE REGULATOR
FLANGE COUPLINGS
PIPE FITTED FOR WATER FILLING
(HYDRAULIC PRESSURE TEST)
WATER TRAP
COMPRESSED AIR SUPPLY
AIR FROM COMPRESSOR

Figure III-1

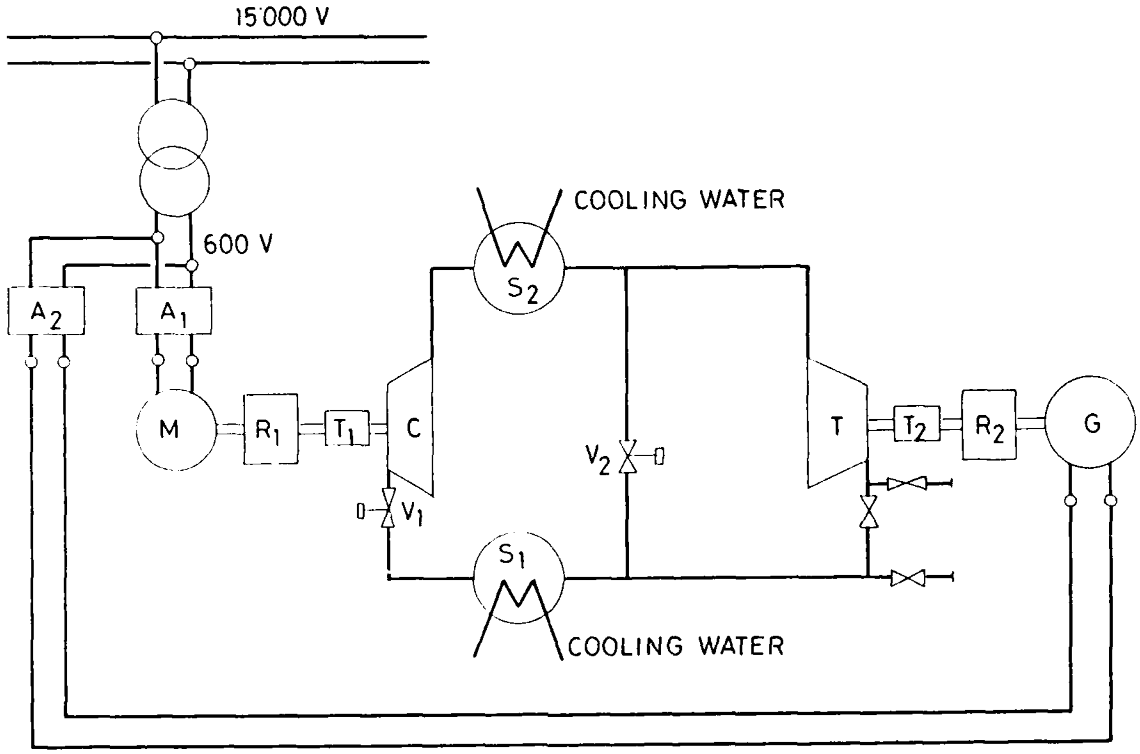


Fig.III-2 Scheme of the turbomachinery test rig

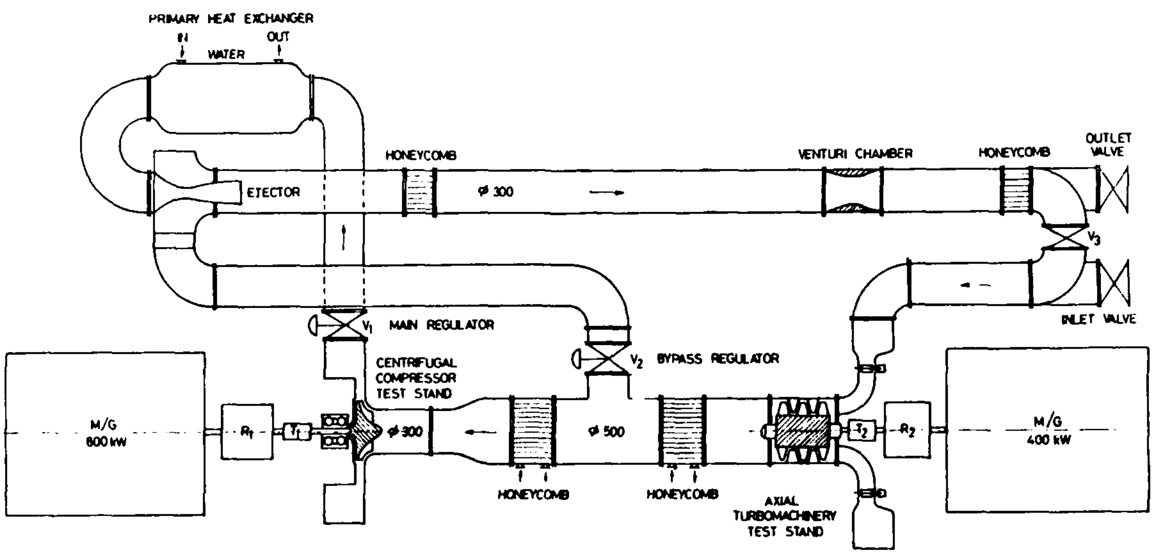


Fig.III-3 Turbomachinery test bed

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The Symposium was organized into 7 sessions: Certification/Demonstration Testing Requirements (8 papers); Development of Test Requirements (5 papers); Engine Life Prediction/Correlation (5 papers); Complete Powerplant Testing (7 papers); Engine Component Testing (7 papers); Development Testing of Gas Turbines for Limited Life Application (1 paper); and Requirements for Future Testing (2 papers).

The aim of the Symposium was to provide better test methods to the engine research and development engineers and to meet the manufacturer's, the buyer's and the user's test requirements for engine delivery, reliability, economy and maintenance. The Symposium created an examination of the various testing types which are proof testing, capability testing, design testing and trouble shooting, and discussed the definition of procedures, instrumentation and test performance for current and future requirements.

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